Ministry of Defence

Defence Standard

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DESIGN AND AIRWORTHINESS REQUIREMENTS FOR SERVICE AIRCRAFT

VOLUME 2 - ROTORCRAFT

DEF STAN 00-970

DESIGN AND AIRWORTHINESS REQUIREMENTS FOR SERVICE AIRCRAFT

VOLUME 2 - ROTORCRAFT

This Defence Standard replaces AvP970 ‘Design Requirements for Service Aircraft’ Volume 3 - Rotorcraft and Volume 3 Memos

1. This Standard has been devised for the use of the Crown and its contractors in the execution of contracts for the Crown, and, subject to the unfair Contract Terms Act 1977, the Crown will not be liable in any way whatever (including but without limitation negligence on the part of the Crown its servants or agents) where the Standard is used for other purposes.

2. Any enquiries regarding this Standard in relation to an invitation to tender or a contract in which it is invoked are to be addressed to the responsible technical or supervising authority named in that invitation to tender or contract.

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AMENDMENT RECORD

Amdt	Incorporated by	Date
1	)	
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5	) Amendment List	
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GENERAL CONTENTS

For details of the Chapter and Leaflet Titles, reference should be made to the Contents in front of each Part.

PREFACE

1 LAYOUT

1.1 The requirements in this Defence Standard are grouped according to their origin or purpose and not on an airframe component basis. The General Contents shows the division into Parts, each Part being allocated one hundred chapter numbers in sequence.

2 INDEX

2.1 To avoid a large number of cross references in the text, the alphabetical index serves as the method of cross referencing requirements for a component.

3 AMENDMENTS

3.1 This Defence Standard will be kept up to date by the issue of amendments.

3.2 In general these amendments will consist of replacement pages, but where only a minor alteration or deletion is necessary instructions for a manuscript amendment may be given. When a manuscript amendment has been made the Amendment number should be added in the margin against the amendment.

3.3 Throughout the Defence Standard a marginal line indicates that the subject matter against the line has been altered or added on the date quoted in the top corner of the page.

3.4 Replacement pages, and reprints of complete chapters not incorporating technical changes have to be issued on occasion, and may include re-numbered or repositioned paragraphs. In these instances marginal lines are not used and the shoulder heading on each page is "Re-issued.....".

3.5 Where a complete chapter or leaflet is issued for the first time marginal lines are not used and the shoulder heading on each page is "Issued.....".

3.6 Dates have not been included on the initial issue of this Defence Standard.

4 RECORD FOLDER

4.1 Since design contracts are based on requirements current at the date of the Aeroplane Specification, it is useful to retain a record of the requirements which have been superseded so far as new types are concerned, but which still apply to existing contracts.

4.2 For this purpose it is recommended that a separate record folder should be kept for filing the pages removed from this publication by amendment action. In amendment instructions the statement "Remove and file" means that the sheets should be filed in the separate folder. For the index and similar sheets the instruction will normally be "Remove and destroy" since these are not needed for record.

4.3 The record folder can be divided by cards, similar to those in this publication, each division corresponding to an amendment number and date. The sheets removed by an amendment can then be placed in the appropriate division.

4.4 It is recommended that the amendment instruction sheets be kept at the beginning of the folder arranged in numerical order.

5 REFERENCES TO THIS DEFENCE STANDARD

5.1 Reference to this Defence Standard in correspondence should quote Defence Standard 00-970 Volume 1 together with the chapter or leaflet number, para number and the amendment number on the shoulder heading.

6 HEALTH AND SAFETY AT WORK ACT

6.1 The Secretary of State for Defence shall not be deemed by virtue of any of these requirements to have assumed any of the responsibilities of the contractor under the Health and Safety at Work Act or any other enactment.

INTRODUCTION

1 GENERAL

1.1 This initial issue of Defence Standard 00-970 Volume 2 replaces AvP970 "Design Requirements for Service Aircraft" Volume 3 (Rotorcraft) and AvP970 Volume 3 Memos.

1.2 The opportunity offered by the reprinting of AvP970 as a Defence Standard, with the Memos incorporated in the text, was taken to revise the publication editorially and technically where time and resources allowed. The new Defence Standard 00-970, therefore, represents the start of a major revision (over the next 5 years) of design and airworthiness requirements for Service aircraft. The objectives of the revision in the longer term are:

(i) To provide modern, state of the art, design and airworthiness requirements for military aircraft projects.

(ii) To reduce the proliferation of design documentation and to implement relevant aspects of NATO and ASCC Standardisation.

(iii) To provide a coherent national view in pursuance of joint European military requirements.

(iv) To maximise commonality with civil airworthiness requirements.

(v) To establish a relationship with USA Mil Specs.

(vi) To pursue the objectives specified in the White Paper (Cmd 8621) - Standards, Quality and International Competitiveness.

1.3 A number of Chapters/Leaflets have been reproduced as they were originally issued or amended in the 1960's and some still quote Imperial Units e.g., Chapter 718 Oxygen Installations.

1.4 Changes will continue to be introduced into Defence Standard 00-970 under the auspices of the Joint Airworthiness Committee.

2 APPLICABILITY OF THE REQUIREMENTS

2.1 The requirements given in the Chapters of Volume 2 of this Defence Standard apply when they are specified in Ministry of Defence Contracts for the new design and conversion of rotorcraft, and are intended to amplify the requirements contained in the Rotorcraft Specifications. In the event of any conflict between the requirements of an Rotorcraft Specification and those contained in this Defence Standard, the requirements in the Rotorcraft Specification shall take precedence.

2.2 Derogation from the requirements of the Chapters (indicated by the words "shall" or "must") is not open to the designer without formal agreement in writing from the MOD Rotorcraft Project Director.

2.3 Information and recommended practices amplifying the Chapter requirements are given in the Leaflets* immediately following the Chapter, and also in a number of Leaflets of Volume 2 of this Standard. The Volume 1 Leaflets applicable are listed on the Reference Pages given at the end of each Chapter (see para 3.2 below). (These Volume 1 Leaflets have in general been written for aeroplanes and hence in some cases may not be wholly applicable to rotorcraft).

2.4 Leaflet contents are not mandatory except where for special reasons the Rotorcraft Specification calls up a particular Leaflet or part thereof. Designers are expected, however, to consult the Leaflets and to incorporate their provisions in their designs unless, in the designer's opinion, there are good reasons for not doing so.

3 SCOPE OF REQUIREMENTS

3.1 The Defence Standard includes within its scope, requirements and recommendations which arise from the operational function of Service Rotorcraft. It covers the following main categories of requirements:

(i) The comfort and safety of the crew, guidance being given in such matters as the placing of controls and seats, the pilot's view and other allied subjects.

(ii) The basic considerations of design, strength and stiffness which are intended to ensure that the rotorcraft can carry out its duties safely and with an acceptably low risk of structural failure.

(iii) The aerodynamic and flying qualities of the rotorcraft.

(iv) The installation of engines, of fuel and oil systems and of the various other rotorcraft services.

(v) The reliability and maintainability of the rotorcraft in order to ensure efficient operation and a reasonable service life.

(vi) The flight tests to be made by the contractor prior to delivery, for acceptance trials, of a new type or converted rotorcraft to an Experimental Establishment.

3.2 Reference Pages contain a list of publications which give useful amplification or explanation of the requirements and recommendations of a Chapter and its Leaflets or which promulgate additional requirements or information of which the designer should be aware. In addition to listing applicable Volume 1 Leaflets (see para 2.3 above), Reference Pages in this Volume will also include references to other publications (e.g., DEF STANs, DERA/RAE (Ex) Reports) which contain useful information applicable to rotorcraft only, together with publications from Volume 1 Reference Pages which may apply wholly or partly to rotorcraft.

* Formerly called Appendices in AvP970 Volume 3

4 INTERPRETATION

4.1 The requirements, with or without explanatory matter, shall not be regarded as constituting a text-book of current aeronautical knowledge. The interpretation of the requirements against a background of such knowledge is essential.

4.2 The MOD Rotorcraft Project Director will advise in cases where doubt exists in requirements expressed qualitatively (e.g., a good view forward, adequately tested).

4.3 Reference to Rotorcraft Project Director shall be taken to mean the MOD Rotorcraft Project Director.

5 STRENGTH REQUIREMENTS

5.1 Many of the strength requirements and recommendations given in this Defence Standard are stated without specific reference to the particular components for which they may be expected to give critical loads. Unless otherwise stated the loads corresponding to the various conditions specified should be traced through the structure far enough to ensure that the rotorcraft has at least the specified factors throughout the whole structure. This does not, however, imply that the whole structure need be stressed for every specified condition; many of the stressing cases overlap, and when it can be shown that any particular case will not give critical loads it will be unnecessary to consider the case further.

6 SPECIFICATIONS AND ALLIED DOCUMENTS

6.1 A reference to a standard, specification or allied document shall be interpreted as implying the latest issue of that document unless the content implies otherwise.

6.2 Documents referred to in this Defence Standard together with the addresses from which they can be obtained are listed in DEF STAN 00-00 (PART 3) SECTION 5 - 'MOD Departmental Standards and Specifications'.

7 REFERENCES TO THE MINISTRY OF DEFENCE (MOD)

7.1 References to the Ministry of Defence (MOD) shall, unless the content implies otherwise be interpreted as references to the appropriate Rotorcraft Project Director, Ministry of Defence or his authorised representative.

8 U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

8.1 A list of Military Specifications (MIL SPECS), Standards (STDS) and Handbooks (HDBKS) which have been identified as covering the design and airworthiness topics of the chapters in DEF STAN 00-970, Volume 2 are now identified in Appendix 2 to each Part. For some of the chapters this comparison is shown at DEF STAN paragraph level, with corresponding paragraphs in the U.S. documents and revision code identified.

8.2 Titles of the U.S. documents referenced in each DEF STAN 00-970 Chapter are given at the head of the chapter.

8.3 U.S. Military Specifications which are known to contain requirements having some equivalence to the chapter, but are under Controlled Distribution, are also listed at the head of each chapter. Included in this list is SD-24: Design and Construction of Aircraft Weapon Systems. Vol II - Rotary Wing Aircraft, produced by the U.S. Navy. Whilst this is not issued as a MIL-STD, it matches the DEF STAN requirements more closely than any MIL-STD or MIL-SPEC, and refers to MIL-STDs and SPECS where appropriate.

8.4 Rarely is there an exact or even close correspondence between a requirement expressed in the DEF STAN and those in the U.S. Military documents, but correspondence has been recorded where requirements address the same or closely similar topic. Headings and sub-headings have not always accurately described the ensuing requirements, particularly in the DEF STAN, and therefore the expressed requirements have been addressed rather than headings.

8.5 Many of the U.S. Military documents are very long and it has therefore been considered useful to reference where possible the para/sub-para numbers of the U.S. document. It has therefore also been necessary to record the revision code of that document. Whilst equivalence has been researched to DEF STAN sub-para level, recording has been limited to para level on the basis that little time would be required by the user to trace the quoted equivalence to sub-para level.

8.6 The listings in the Appendices (No.2) is relevant to Amendment 10 of DEF STAN 00-970 Volume 2 and to U.S. Government Specifications Service Numeric Index of Specifications, Standards and Handbooks. MIL-E-1 to MIL-STD-600010) Issue 93-03 May-June 1993.

8.7 It is now general policy in the U.S. not to produce new Military Specifications, Standards and Handbooks; nor to up-date those extant. However, there may be some high level Standards which will be retained. Therefore reference to the Civil Rotorcraft's Model Specification may be necessary for the listing of airworthiness requirements to which the rotorcraft is designed for assessment of military derivatives.

8.8 A list of all the MIL SPECs, STDs and HDBKs referred to in the Appendices (No. 2) is shown in "computer" numerical order in Table 1.

9 MILITARY DERIVATIVES OF CIVIL ROTORCRAFT

9.1 Future military transport, light communications, patrol, non-combat training rotorcraft and specialist conversions may be derivatives of civil rotorcraft. Foreign military derivatives of civil rotorcraft certificated to civil requirements fall within this category. It is probable that the rotorcraft will be fully certified to its civil operating standard against the requirements of BCARs, FARs of JARs current at the time of design or certification. There are, however, certain basic differences in the civil requirements from these sources and the military requirement of DEF STAN 00-970.

9.2 An acceptance procedure is necessary for such rotorcraft against their proposed military roles, and to determine to what extent certification procedures would be affected by configuration of other changes, e.g., increased operating weights, in the military roles envisaged.

9.3 An Appendix (No. 1) to each of Parts 1 to 8 inclusive sets out design and airworthiness requirements and recommendations for military derivatives. Parts 9 and 10 (Flight Tests) state the test procedures and instrumentation requirements to demonstrate that the Aeroplane can be controlled and the installations and structures function satisfactorily in accordance with the requirements of Parts 1 to 8, reference should be made to Appendix (No 1) of these Parts and the results of the compliance checks/assessments carried out.

9.4 COMPLIANCE CHECKS

9.4.1 Throughout Appendices No 1 to Parts 1 to 10 reference is made to the need for 'Compliance Checks' of the existing rotorcraft against the military requirements of DEF STAN 00-970. It is obviously desirable that the military derivative of a civil rotorcraft as designed and certificated, should be acceptable for its proposed military role(s) without changes other than those needed for operation as a military derivative. However, where there are no corresponding civil requirements or the military and civil requirements conflict or where the military requirements are more severe compliance checks on the existing civil rotorcraft may be needed and will be called for in the derivative specification or by directive from the Rotorcraft Project Director.

9.4.2 The aim of the compliance checks is <u>not</u> to bring the military derivative automatically up to the full military standard in each case, but to provide information to assist in determining whether the derivative:

(i) needs to meet the military requirements

(ii) is acceptable for its military role(s) without change

(iii) is not acceptable without change to meet its military role(s), and

(iv) the extent of any change needed to bring the derivative to a standard at which it will satisfactorily fulfil its military role(s).

9.4.3 Background information to assist in these compliance checks is contained in the DEF STAN Leaflets Parts 1 to 10 and the relevant Chapter - Appendices.

9.4.4 Airworthiness Directives of UK and foreign origin applicable to the type must be taken into account.

9.4.5 The Appendices (No 1) refer to the issue or change number of the requirements listed in 9.5. The derivative may have been designed to different issues/changes of the requirements and these will need to be taken into account.

9.4.6 BCAR 29, FAR 27 and 29 and JAR 27 and 29 have in most cases, identical paragraph numbers for the same requirement, the text of these paragraphs can differ and result in a different interpretation. Due attention must be given to each requirement applicable to the aeroplane under consideration.

9.4.7 On completion of the compliance checks the implementation of any actions required such as modifications will be through the derivative's specification or by a directive from the Rotorcraft Project Director.

9.5 THE CIVIL REQUIREMENTS LISTED IN THE APPENDICES ARE TAKEN INTO ACCOUNT AND ARE RELATED TO DEF STAN 00-970 AMENDMENT 10

(i) British Civil Airworthiness Requirements (BCAR)

Section G Rotorcraft Issue 9
(Replaced by BCAR 29)
Blue Papers 235, G707, G749, G774. G778,
G779, G780 + Corrigendum No. 1, G285, G286,
G865, G809, G811, G825, G826, G832, G839
BCAR 29 Rotorcraft Issue 1
Section J Electrical Issue 3
(being incorporated in G)

BCAR's Sections G and J are included as being relevant to rotorcraft which were designed and certified prior to the dates on which they were superseded.

(ii) Joint Airworthiness Requirements (JAR)

JAR 27 Small Rotorcraft Issue 1

JAR 29 Large Rotorcraft Issue 1

(iii) Federal Aviation Regulations (FAR)

Part 27 Airworthiness Standards: Normal Category Rotorcraft
Revision 27-28 RUSCO (Rules Services Co)
GA (Government Amendment) No. 27-33

Part 29 Airworthiness Standards: Transport Category Rotorcraft
Revision 29-32 RUSCO
GA No. 29-39

10 MANDATORY REFERENCE TO MOD ESTABLISHMENTS/RESEARCH AGENCIES/BRANCHES

10.1 In order to comply with some requirements of this DEF STAN it is necessary for the contractor design authority to consult with, or obtain the approval of MOD Establishments, Research Agencies or Branches. Unless otherwise specified in the particular requirements, the consultation is to take place via the appropriate MOD Rotorcraft Project Director. Also some requirements of the DEF STAN oblige the contractor design authority to consult with, or seek approval of, the MOD Rotorcraft Project Director.

11 ENVIRONMENTAL IMPACT

11.1 The design solutions chosen in compliance with this standard are to take account of environmental implications. Where the design solution requires a substance which could create an adverse environmental impact during the manufacture, use or disposal of the substance, or of the aircraft or its equipment, consideration is to be given to alternative design solutions. When the chosen design solution has an adverse environmental impact the Aircraft Project Director is to be advised.

TABLE 1

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-C-104	Crates, Wood; Lumber and Plywood Sheathed, Nailed and Bolted
MIL-P-116	Preservation Methods of
MIL-M-3171	Magnesium Alloy, Processes for Pretreatment and Prevention of Corrosion on
MIL-F-3541	Fittings, Lubrications, General Specification for
MIL-S-5002	Surfaces Treatments and Inorganic Coatings for Metal Surfaces of Weapon Systems
MIL-W-5013	Wheel and Brake Assemblies, Aircraft, General Specification for
MIL-T-5041	Tires, Pneumatic, Aircraft
MIL-W-5044	Walkway Compound, Nonslip and Walkway Matting, Nonslip
MIL-B-5087	Bonding, Electrical and Lightning Protection, For Aerospace Systems
MIL-W-5088	Wiring, Aerospace Vehicle
MIL-H-5364	Harness, Shoulder Safety, General Specification
MIL-E-5400	Electronic Equipment, Aerospace, General Specification for
MIL-P-5425	Plastic Sheet, Acrylic, Heat Resistant
MIL-H-5440	Hydraulic Systems, Aircraft, Types 1 and II, Design and Installation Requirements for
MIL-G-5485	Glass: Laminated, Flat, Bullet-Resistant
MIL-F-5504	Filters and Filter Elements, Fluid Pressure, Hydraulic, Micronic Type
MIL-P-5518	Pneumatic System, Aircraft, Design, Installation and Data Requirements for
MIL-R-5520	Reservoirs, Aircraft, Hydraulic, Non-Separated Type
MIL-T-5522	Test Requirements and Methods for Aircraft Hydraulic and Emergency Pneumatic Systems
MIL-V-5524	Valves, Check, Hydraulic, Aircraft Type 1 Systems
MIL-T-5578	Tank, Fuel, Aircraft, Self-Sealing
MIL-C-5604	Compass, Magnetic, Pilot’s Standby
MIL-E-5627	Extinguishers, Fire, Carbon Dioxide, Portable
MIL-L-5667	Lighting, Equipment, Aircraft Instrument Panel, General Specification for, Installation of
MIL-C-5778	Covers, Aircraft Components, General Requirements for
MIL-T-5842	Transparent Areas, Anti-Icing, Defrosting and Defogging Systems, General Specification for

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-T-5955	Transmission Systems, VTOL-VSTOL, General Requirements for
MIL-E-6051	Electromagnetic Compatibility Requirements, System
MIL-T-6053	Tests, Impact, Shock Absorber Landing Gear, Aircraft
MIL-I-6115	Instrument Systems, Pitot Tube and Flush Static Port Operated, Installation of
MIL-R-6130	Rubber, Cellular, Chemically Blown
MIL-S-6144	Sound and Thermal Insulation for Aircraft, General Specification for the Installation of
MIL-G-6162	Generators and Starter Generators, Electric Direct Current, Nominal 30 Volts, Aircraft, General Specification for
MIL-T-6396	Tanks, Aircraft Propulsion Fluid System, Internal, Removable, Non-Self-Sealing
MIL-S-6451	Shields, Protective, Aircraft and Missiles
MIL-L-6484	Lights, Cockpit, Utility, Aircraft, General Specification for
MIL-L-6503	Lighting Equipment, Aircraft, General Specification for Installation of
MIL-S-6625	Spray Equipment, Aircraft Windshield Anti-Icing
MIL-L-6723	Lights, Aircraft, General Specification for
MIL-W-006729	Watertightness of Aircraft, Testing, General Specification for
MIL-L-006730	Lighting Equipment, Exterior Aircraft (General Requirements for)
MIL-M-6857	Magnesium Alloy Castings, Heat Treatment of
MIL-I-6866	Inspection, Liquid Penetrant
MIL-I-6870	Inspection Program Requirements, Non-Destructive, for Aircraft and Missile Materials and Parts
MIL-S-6904	Signals, Emergency Warning, General Specification for
MIL-E-7016	Electric Load and Power Source Capacity, Aircraft, Analysis of
MIL-I-7028	Instruments and Instrument Boards, Aircraft, Installation of
MIL-I-7032	Inverters, Aircraft, General Specification for
MIL-E-7080	Electric Equipment, Aircraft, Selection and Installation of
MIL-T-7101	Transmission: Power, Constant Speed, General Specification (Aircraft Use)
MIL-C-7115	Converters, Aircraft, General Specification for
MIL-I-7171	Insulation Blanket, Thermal Acoustical

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-F-7179	Finishes, Coatings, and Sealants for the Protection of Aerospace Weapons Systems
MIL-C-7188	Compass, Pilots Standby, Installation of
MIL-F-7190	Forgings, Steel, for Aircraft/Aerospace Equipment and Special Ordnance Applications
MIL-C-7413	Couplings, Quick Disconnect, Automatic Shutoff, General Specification for
MIL-S-7470	Shaft, Power Transmission, Aircraft Accessory, General Specification for
MIL-I-7566	Indicator System
MIL-R-7705	Radomes, General Specification for
MIL-R-7726	Repair and Rebuilding of Used Aircraft Pneumatic Tires
MIL-T-7743	Testing, Store Suspension and Release Equipment, General Specification for
MIL-C-7762	Compasses, Installation of
MIL-P-7788	Panels, Information, Integrally Illuminated
MIL-F-7872	Fire and Overheat Warning Systems, Continuous Aircraft: Test and Installation of
MIL-C-7905	Cylinder, Steel, Compressed Gas, Nonshatterable
MIL-P-8045	Plastic, Self-Sealing and Non-Self Sealing Tank Backing Material
MIL-B-8075	Brake Control Systems, Anti-Skid, Aircraft Wheels, General Specification for
MIL-M-8090	Mobility, Towed Aerospace Ground Equipment, General Requirements for
MIL-D-8181	Detector, Ice, Air Intake Duct, Aircraft Engines and Airframe Systems, General Specification for
MIL-P-8184	Plastic Sheet, Acrylic, Modified
MIL-R-8236	Reel, Shoulder Harness, Inertia Lock
MIL-F-8490	Fastener, Case, for Equipment Rack System in Aircraft
MIL-I-8500	Interchangeability and Replaceability of Component Parts for Aerospace Vehicles
MIL-H-8501	Helicopter Flying and Ground Qualities, General Requirements for
MIL-S-8512	Support Equipment, Aeronautical, Special, General Specification for the Design of
MIL-S-8552	Landing Gear, Aircraft Shock Absorber (Air-Oil Type)
MIL-P-8564	Pneumatic System Components, Aeronautical, General Specification for

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-B-8565	Battery, Storage, Aircraft, General Specification for
MIL-B-8584	Brake Systems, Wheel, Aircraft, Design of
MIL-A-8591	Airborne Stores, Suspension Equipment and Aircraft-Store Interface (Carriage Phase); General Design Criteria for
MIL-E-008593	Engines, Aircraft, Turboshaft and Turboprop, General Specification for
MIL-F-8615	Fuel System Components, General Specification for
MIL-M-8650	Mock ups, General Specification for
MIL-P-8651	Plates: Identification and Modification (for Aircraft) Installation of
MIL-I-8670	Installation of Field Guns and Associated Equipment in Naval Aircraft
MIL-I-8671	Installation of Droppable Stores and Associated Release Systems
MIL-I-8672	Installation of Test and Aircraft Pyrotechnic Equipment in Aircraft, General Specification for
MIL-I-8673	Installation and Test of Aircraft Flexible Weapon Systems
MIL-I-8675	Installation,. Aircraft Armor
MIL-I-8677	Installation of Armament Control Systems and Associated Equipment in Naval Aircraft
MIL-C-8678	Cooling Requirements of Power Plant Installations
MIL-T-8679	Test Requirements, Ground, Helicopter
MIL-D-8683	Design and Installation of Gaseous Oxygen Systems in Aircraft, General Specification for
MIL-P-8686	Power Units; Aircraft Auxiliary, Gas Turbine Type, General Specification for
MIL-S-8698	Structural Design Requirements, Helicopters
MIL-I-8700	Installation and Test of Electronic Equipment in Aircraft, General Specification for
MIL-D-8708	Demonstration, Aircraft Weapon Systems, General Specification for
MIL-H-8755	Hydraulic System Components, Aircraft and Missiles, General Specification for
MIL-C-8779	Colours, Interior, Aircraft, Requirements for
MIL-A-8806	Sound Pressure Levels in Aircraft, General Specification for
MIL-D-8804	De-Icing System, Pneumatic Boot, Aircraft, General Specification for
MIL-S-8812	Steering System, Aircraft, General Requirements for

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-F-8815	Filter and Filter Elements, Fluid Pressure, Hydraulic Line, 15 Micron Absolute and 5 Micron Absolute, Type II Systems
MIL-S-8860	Airplane Strength and Rigidity, General Specification for
MIL-A-8861	Airplane Strength and Rigidity, Flight Loads
MIL-A-8863	Airplane Strength and Rigidity, Ground Loads for Navy Acquired Airplanes
MIL-A-8865	Airplane Strength and Rigidity, Miscellaneous Loads
MIL-A-8867	Airplane Strength and Rigidity, Ground Tests
MIL-A-8868	Airplane Strength and Rigidity, Data and Reports
MIL-A-8869	Airplane Strength and Rigidity, Nuclear Weapon Effects
MIL-A-8870	Airplane Strength and Rigidity, Vibration, Flutter and Divergence
MIL-H-8890	Hydraulic Components, Type III, General Specification for
MIL-H-8891	Hydraulic Systems, Manned Flight Vehicles, Type II Design. Installation and Data Requirements for, General Specification for
MIL-F-8905	Fittings and Cargo Rings, Tie Down, Aircraft Floor
MIL-R-8931	Reservoirs: Aircraft and Missile, Hydraulic, Separated Type
MIL-E-8970	Engines and Related Propulsion and Power Equipment, Aircraft, Acceptance Tests of, Sampling Plan for, Statistical
MIL-P-9024	Packaging, Handling and Transportability in System/Equipment Acquisition
MIL-A-9094	Arrester, Lightning, General Specification, for Design of
MIL-D-9129	Dischargers, Aircraft, Eletrostatic, General Specification for
MIL-T-9166	Tie Down, Cargo Aircraft, Cable Net, 10,000 LB capacity, Type A-2
MIL-P-9400	Plastic Laminate and Sandwich Construction Parts, Aircraft Structural, Process Specification Requirements
MIL-A-9482	Anti-Icing Equipment for Aircraft, Heated Surface Type, General Specification for
MIL-F-9490	Flight Control Systems-Design, Installation and Test of Piloted Aircraft, General Specification for
MIL-Q-9858	Quality Program Requirements
MIL-C-11866	Casting, Precision, Non Ferrous
MIL-L-11992	Launchers for Guided Missiles, Ground and Airborne, General Specification for
MIL-M-13231	Marking of Electronic Items

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MI-P-15024	Plates, Tags and Bands for Identification of Equipment
MIL-C-16310	Cylinder, Compressed Gas (Compressed Air and Carbon Dioxide), Nonshatterable and Non-Magnetic
MIL-L-16462	Litter, Folding, Rigid Pole
MIL-F-17874	Fuel Systems: Aircraft, Installation and Test of
MIL-M-18012	Markings for Aircrew Station Displays, Design and Configuration of
MIL-C-18244	Control and Stabilisation Systems, Automatic, Piloted Aircraft, General Specification for
MIL-I-18259	Installation of Window Anti-Icing, Degreasing and Washing Systems, General Specification for
MIL-L-18276	Lighting, Aircraft Interior, Installation of
MIL-D-18300	Design Data Requirements for Avionic Equipment
MIL-N-18307	Nomenclature and Identification for Aeronautical Systems Including Joint Electronics Type Designated Systems and Associated Support Systems
MIL-H-18325	Heating and Ventilating Systems, Aircraft, General Specification for
MIL-R-18370	Release Systems for the Ejection of Life Rafts from Naval Patrol and Transport Series Aircraft, Design Requirements for
MIL-F-18372	Flight Control Systems: Design, Installation and Test of, Aircraft (General Specification for)
MIL-I-18373	Instruments and Navigation Equipment, Aircraft: Installation of
MIL-T-18606	Test Procedures for Aircraft Environmental Systems
MIL-T-18607	Thermal Anti-Icing Equipment, Wing and Empennage
MIL-I-18802	Fuel and Oil Lines, Aircraft, Installation of
MIL-T-18847	Tanks, Fuel, Aircraft, Auxiliary Eternal, Design and Installation of
MIL-E-18927	Environmental Control Systems, Aircraft, General Requirements for
MIL-A-19531	Aircraft: Maintenance and Engineering Inspection Requirements
MIL-A-19736	Air Refuelling Systems, General Specification for
MIL-O-19838	Oil Systems, Aircraft, Installation and Test of
MIL-A-21180	Aluminium-Alloy Castings, High Strength
MIL-G-21480	Generator System, 400 Hertz Alternating Current, Aircraft, General Specification for
MIL-D-21625	Design and Evaluation of Cartridges for Cartridge Actuated Devices

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-E-21981	Electronics Equipment, Nomenclature, Serial Numbers and Identification Plates: Requirements for
MIL-E-22285	Extinguishing System, Fire, Aircraft, High-Rate-Discharge Type, Installation and Test of
MIL-L-22769	Launcher, Weapons, Airborne and Associated Equipment, General Specification for
MIL-T-23103	Thermal Performance Evaluation, Airborne Electronic Equipment and Systems, General Specification for
MIL-D-2322	Demonstration Requirements for Helicopters
MIL-F-23447	Fire Warning Systems, Aircraft, Radiation Sensing Type: Test and Installation of
MIL-H-23599	Hook, Helicopter Rescue
MIL-I-23659	Initiators, Electric, General Design Specification for
MIL-R-23761	Regulators, Voltage, and Control Panels, Aircraft Direct Current Generator, General Specification for
MIL-E-24021	Electric Power Monitors, Eternal, Aircraft
MIL-K-25049	Knobs, Control, Electronic Equipment, General Specification for
MIL-C-25050	Colors, Aeronautical Lights and Lighting Equipment, General Specification for
MIL-S-25073	Seat, Aircraft
MIL-W-25140	Weight and Balance Control System (for Aircraft and Rotorcraft)
MIL-F-25173	Fastener Control Panel, Aircraft Equipment
MIL-A-25175	Air Transport, Nontactical, Packing for
MIL-F-25381	Flight Testing, Electric System, Piloted Aircraft and Guided Missile, General Requirements for
MIL-C-25427	Coupling Assembly, Hydraulic, Self-Sealing, Quick Disconnect
MIL-L-25467	Lighting, Integral, Red, Aircraft Instrument, General Specification for
MIL-E-25499	Electrical Systems, Aircraft, Design and Installation of, General Specification for
MIL-M-25500	Mockup Testing, Electric System, Piloted Aircraft and Guided Missile, General Requirements for

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-P-25690	Plastic, Sheets and Parts, Modified Acrylic Base, Monolithic, Crack Propagation Resistant
MIL-A-25730	Antenna Subsystem for Airborne Identification and Navigation Equipments, General Specification for
MIL-G-25871	Glass, Laminated, Aircraft Glazing
MIL-T-25959	Tie Down, Cargo, Aircraft
MIL-B-26220	Batteries, Storage, Aircraft, Nickel-Cadmium, General Specification for
MIL-P-26292	Pitot and Static Pressure Systems, Installation and Inspection of
MIL-F-26301	Flasher, Solid State, Aircraft Navigational Light, General Specification for
MIL-L-27160	Lighting, Instrument, Integral, White, General Specification for
MIL-S-27174	Seat, Troop, Variable Seating Width
MIL-T-27260	Tie Down Cargo, Aircraft, CGU-1/B
MIL-T-27422	Tank, Fuel, Crash-Resistant, Aircraft
MIL-T-28800	Test Equipment for use with Electrical and Electronic Equipment, General Specification for
MIL-T-31000	Technical Data Packages, General Specification for
MIL-A-46146	Adhesive Sealants, Silicone, RTV, Non-Corrosive (for use with Sensitive Metals and Materials)
MIL-S-38039	Systems, Illuminated, warning, Caution, and Advisory, General Specification for
MIL-C-38214	Compass, Magnetic, Mounted
MIL-C-38373	Cap, Fluid Tank Filler
MIL-L-38779	Lavatories and Accessories Aircraft
MIL-M-46062	Magnesium Alloy Castings, High Strength
MIL-A-46108	Armor, Transparent, Glass: Glass/Plastic; Plastic Laminates (General Specification)
MIL-P-46111	Plastic Foam, Polyurethane (for use in Aircraft)
MIL-P-46112	Plastic Sheet and Strip, Polyimide
MIL-A-46146	Adhesive Sealants, Silicone, RTV, Non-Corrosive (for use with Sensitive Metals and Materials)
MIL-H-46855	Human Engineering Requirements for Military Systems, Equipment and Facilities

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-S-58095	Seat System; Crash Resistant, Non-Ejection, Aircrew, General Specification for
MIL-T-81259	Tie-Downs, Airframe Design, Requirements for
MIL-A-81264	Asbestos Felt or Mat, Resin Impregnated
MIL-B-81365	Bleed Air Systems, General Specification for
MIL-R-81367	Rain Removal Systems, Aircraft Windshield, Jet Air Blast
MIL-T-81571	Thermal Protective System, Aircraft Cockpit, General Specification for
MIL-R-81589	Rain Repellent Fluid Application system, Aircraft Windshield
MIL-P-81655	Pitot-Static Tube, L-Shaped, Compensated, Electrically Heated, General Specification for
MIL-R-81729	Restraint Systems, Aircrewman's
MIL-B-81757	Batteries and Cells, Storage, Nickel-Cadmium, Aircraft, General Specification for
MIL-S-81771	Seats; Aircrew, Adjustable; Aircraft, General Specification for
MIL-C-81774	Control Panel, Aircraft, General Requirements for
MIL-D-81980	Design and Evaluation of Signal Transmission Subsystems, General Specification for
MIL-E-82513	Explosive Block, Airframe Separation
MIL-R-83283	Rubber, Silicone, High Strength, Cabin Pressure Seal Material, Diaphragm Type
MIL-F-83300	Flying Qualities of Piloted V/STOL Aircraft
MIL-P-83310	Plastic Sheet, Polycarbonate, Transparent
MIL-A-83376	Adhesive Bonded Metal Faced, Sandwich Structures, Acceptance Criteria
MIL-A-83377	Adhesive Bonding (Structural) for Aerospace and Other Systems, Requirements for
MIL-C-83413	Connectors and Assemblies, Electrical, Aircraft Grounding, General Specification
MIL-I-83456	Installation of Segmented Lightning Diverter Strips on Aircraft Radomes, General Specification for
MIL-C-83467	Cloth, Fire Retardent and Fire Retardent Treated, Aircraft, Upholstery
MIL-F-83660	Filter Element, Fluid Pressure, Hydraulic, Aircraft Disposable, 5 Micron Absolute

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY SPECIFICATIONS

MIL-F-83861	Filter Elements, Noncleanable, Fluid Pressure, Hydraulic Line, 5 and 15 Micron Absolute, Type II Systems, General Specification for
MIL-F-83870	Filter Element, Fluid Pressure, Transmission, Aircraft, Disposable, 20 Micron Absolute
MIL-P-85034	Power Package, Emergency AC/DC Generator Ran Air Turbine Driven
MIL-I-85071	Inverters, Aircraft, DC to AC, General Specification for
MIL-T-85075	Tie Down Assembly, Helicopter Blade, General Specification for
MIL-V-85245	Valve, Relief, Hydraulic, High Response, Type II Systems, General Specification for
MIL-L-85314	Light Systems, Aircraft, Anti-Collision, Strobe, General Specification for
MIL-S-85510	Seats, Helicopter Cabin, Crashworthy. General Specification for
MIL-P-95573	Power Unit, Aircraft, Auxiliary, Gas Turbine, General Specification for
MIL-E-85583	Electric Power Generating Channel, Variable Input Speed, Alternating Current, 400 Hz, Aircraft; General Specification for
MIL-L-87139 CD	Landing Gear Systems
MIL-P-87141	Parachutes
MIL-E-87145 CD	Environmental Control, Airborne
MIL-F-87154 CD	Fuel Systems
MIL-F-87168 CD	Fire and Explosion Hazard Protection Systems, General Specification for
MIL-P-87210 CD	Pneumatic Power Systems, High Pressure
MIL-E-87219 CD	Electrical Power Systems, Aircraft
AFGS-A-87221 CD	Aircraft Structures - General Specification for
AFGS-E-87235	Emergency Escape, Aircraft
AFGS-S-87328 CD	Survival and Flotation System, Airborne, Specification for
AS-3694 CD	Transmission Systems, VTOL-STOL, General Requirements for
SD-24 CD	Design and Construction of Aircraft Weapon Systems. Vol II - Rotary Wing Aircraft
SD-8706 CD	General Specification for Design Examinations, Engineering, Aircraft Weapon Systems

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY HANDBOOKS

MIL-HDBK-5	Metallic Materials and Elements for Aerospace Vehicle Structures
MIL-HDBK-131	Identification Markings for Fasteners
MIL-HDBK-132	Protective Finishes for Metals and Wood Surfaces
MIL-HDBK-221	Fire Protection Design Handbook for U.S. Navy Aircraft Powered by Turbine Engines
MIL-HDBK-235	Electromagnetic (radiated) Environment Consideration for Design and Procurement of Electrical and Electronic Equipment, Subsystems and Systems
MIL-HDBK-237	Electromagnetic Compatibility Management Guide for Platforms, Systems and Equipment
MIL-HDBK-241	Design Guide for Electromagnetic Interference (EMI) Reduction in Power Supplies
MIL-HDBK-244	Guide for Aircraft/Stores Compatibility
MIL-HDBK-253	Guidance for the Design and Test of Systems Protected against the effects of Electromagnetic Energy
MIL-HDBK-274	Electrical Grounding for Aircraft Safety
MIL-HDBK-275	Guide for Selection of Lubricant Fluids and Compounds for use in Flight Vehicles and Components
MIL-HDBK-336-1	Survivability, Aircraft, Non-Nuclear, General Criteria - Volume 1
MIL-HDBK-336-2	Survivability, Aircraft, Non-Nuclear, Airframe - Volume 2
MIL-HDBK-336-3	Survivability, Aircraft, Non-Nuclear, Engine - Volume 3
MIL-HDBK-337	Adhesive Bonded Aerospace Structure Repair
MIL-HDBK-419	Ground Bonding, and Shielding for Electronic Equipments and Facilities Basic Theory
MIL-HDBK-472	Maintainability Prediction
MIL-HDBK-691	Adhesive Bonding
MIL-HDBK-693	Magnesium and Magnesium Alloys
MIL-HDBK-694	Aluminium and Aluminium Alloys
MIL-HDBK-697	Titanium and Titanium Alloys
MIL-HDBK-698	Copper and Copper Alloys
MIL-HDBK-705	Generator Sets, Electrical, Measurements and Instrumentations
MIL-HDBK-723	Steel and Wrought Iron Products

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY HANDBOOKS

MIL-HDBK-728	Nondestructive Testing
MIL-HDBK-729	Corrosion and Corrosion Prevention, Metals
MIL-HDBK-772	Military Packaging Criteria
MIL-HDBK-1553	Multiplex Application Handbook

MILITARY STANDARDS

MIL-STD-12	Abbreviations for use on Drawings, Specifications, Standards and in Technical Documents
MIL-STD-22	Welded Joint Design
MIL-STD-100	Engineering Drawing Practices
MIL-STD-101	Colour Code for Pipelines and Compressed Gas Cylinders
MIL-STD-129	Marking for Shipment and Storage
MIL-STD-130	Identification Marking of U.S. Military Property
MIL-STD-210	Climatic Information to Determine Design and Test Requirements for Military Systems and Equipment
MIL-STD-250	Aircrew Station Controls and Displays for Rotary Wing Aircraft
MIL-STD-280	Definitions of Item Levels, Item Exchangeability, Models and Related Terms
MIL-STD-322	Explosive Components, Electrically Initiated, Basic Evaluation Tests for
MIL-STD-411	Aircrew Station Signals
MIL-STD-417	Classification System and Tests for Solid Elastomeric Materials
MIL-STD-453	Inspection, Radiographic
MIL-STD-454	Electronic Equipment, Standard General Requirements for
MIL-STD-461	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462	Electromagnetic Interference, Characteristics, Measurement of
MIL-STD-469	Radar Engineering Design Requirements, Electromagnetic Compatibility
MIL-STD-470	Maintainability Program for Systems and Equipment
MIL-STD-471	Maintainability Verification/Demonstration/Evaluation
MIL-STD-490	Specification Practices
MIL-STD-680	Contractor Standardisation Program Requirements
MIL-STD-681	Identification Coding and Application of Hook Up and Lead Wire

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY STANDARDS

MIL-STD-704	Aircraft Electric Power Characteristics
MIL-STD-705	Generator Sets, Engine Driven, Methods of Test and Instructions
MIL-STD-765	Compass Swinging, Aircraft, General Requirements for
MIL-STD-781	Reliability Testing for Engineering Development, Qualification and Production
MIL-STD-783	Legends for Use in Aircrew Stations and on Airborne Equipment
MIL-STD-785	Reliability Program for Systems and Equipment Development and Production
MIL-STD-794	Parts and Equipment, Procedures for Packaging
MIL-STD-805	Towing Fittings and Provisions for Military Aircraft, Design Requirements for
MIL-STD-809	Adapter, Aircraft, Jacking Point, Design and Installation of
MIL-STD-810	Environmental Test Methods and Engineering Guidelines
MIL-STD-838	Lubrication of Military Equipment
MIL-STD-850	Aircrew Station Vision Requirements for Military Aircraft
MIL-STD-872	Test Requirements and Procedures for Aircraft Emergency Ground and Ditching Escape Provisions
MIL-STD-875	Type Designation System for Aeronautical and Support Equipment
MIL-STD-877	Antenna Subsystems, Airborne Criteria for Design and Location of
MIL-STD-878	Method of Dimensioning and Determining Clearance for Aircraft Tires and Rims
MIL-STD-882	System Safety Program Requirements
MIL-STD-889	Dissimilar Metals
MIL-STD-961	Military Specifications and Associated Documents
MIL-STD-962	Military Standards, Handbooks, and Bulletins, Preparation of
MIL-STD-965	Parts Control Program
MIL-STD-970	Standards and Specifications, Order of Preference for the Selection of
MIL-STD-973	Configuration Management
MIL-STD-1186	Cushioning, Anchoring, Bracing, Blocking and Waterproofing with Appropriate Methods
MIL-STD-1247	Markings Functions and Hazard Designations of Hose, Pipe, and Tube Lines for Aircraft, Missile, and Space Systems

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY STANDARDS

MIL-STD-1285	Marking of Electrical and Electronic Parts
MIL-STD-1288	Aircrew Protection Requirements, Non-Nuclear Weapons Threat
MIL-STD-1289	Airborne Stores, Ground Fit and Compatibility Test for
MIL-STD-1290	Light Fixed and Rotary Wing Aircraft Crash Resistance
MIL-STD-1333	Aircrew Station Geometry for Military Aircraft
MIL-STD-1353	Electrical Connectors, Plug-In Sockets and Associated Hardware, Selection and use of
MIL-STD-1365	General Design Criteria for Holding Equipment Associated with Weapons and Weapon Systems
MIL-STD-1366	Transportability Criteria
MIL-STD-1367	Package, Handling, Storage and Transportability Program, Requirements for Systems and Equipment
MIL-STD-1385	Preclusion of Ordnance Hazards in Electromagnetic Fields, General Requirements for
MIL-STD-1472	Human Engineering Design Criteria for Military Systems, Equipment and Facilities
MIL-STD-1476	Metric System, Application in New Design
MIL-STD-1512	Electro-Explosive Subsystems, Electrically Initiated, Design Requirements, and Test Methods
MIL-STD-1521	Technical Reviews and Audits for Systems, Equipments, and Computer Software
MIL-STD-1523	Age Controls of Age-Sensitive Elastomeric Material (For Aerospace Applications)
MIL-STD-1530	Aircraft Structural Integrity Program, Airplane Requirements
MIL-STD-1553	Aircraft Internal Time Division Command./Response Multiplex Data Bus
MIL-STD-1629	Procedures for Performing a Failure Mode, Effects and Criticality Analysis
MIL-STD-1754	Fastening Devices Preferred for Design, Listing of
MIL-STD-1757 CD	Lightning Qualification Test Techniques for Aerospace Vehicles and Hardware
MIL-STD-1760	Aircraft/Stores Electrical Interconnection System
MIL-STD-1763	Aircraft/Stores Certification Procedures
MIL-STD-1776 CD	Aircrew Stations and Passenger Accommodations

TABLE 1 (continued)

TABULAR LISTING OF MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS REFERRED TO IN APPENDICES No. 2

MILITARY STANDARDS

MIL-STD-1788	Avionics Interface Design Standard
MIL-STD-1789 CD	Sound Pressure Levels in Aircraft
MIL-STD-1791 CD	Designing for Internal Aerial Delivery in Fixed Wing Aircraft
MIL-STD-1843	Reliability-Centered Maintenance for Aircraft, Engines, and Equipment
MIL-STD-1949	Inspection, Magnetic Particle
MIL-STD-2069	Requirements for Aircraft Nonnuclear Survivability Program
MIL-STD-2072	Survivability, Aircraft; Establishment and Conduct of Programs for
MIL-STD-2073	DOD Material Procedures for Development and Application of Packaging Requirements
MIL-STD-2074	Failure Classification for Reliability Testing
MIL-STD-2076	Unit Under Test, Compatibility with Automatic Test Equipment, General Requirements for
MIL-STD-2077	Test Program Sets, General Requirements for
MIL-STD-2084	Maintainability of Avionic and Electronic Systems and Equipment, General Requirements for
MIL-STD-2089	Aircraft, Nonnuclear Survivability Terms
MIL-STD-2124	Flight Data Recorder, Functional Standards for
MIL-STD-2131	Launcher, Ejection, Guided Missile, Aircraft, General Design Criteria for
MIL-STD-2154	Inspection, Ultrasonic, Wrought Metals, Process for
MIL-STD-2156	Launcher, Rail, Guided Missile, Aircraft, General Design Criteria for
MIL-STD-2161	Paint Schemes and Exterior Markings for U.S. Navy and Marine Corps Aircraft
MIL-STD-2164	Environmental Stress Screening Process for Electronic Equipment
MIL-STD-2165	Testability Program for Electronic Systems and Equipment
MIL-STD-2173	Reliability-Centered Maintenance Requirements for Naval Aircraft, Weapons Systems and Support Equipment
MIL-STD-2175	Castings, Classification and Inspection of
MIL-STD-6866	Inspection Liquid Penetrant

ALPHABETICAL INDEX

Note: This index will be updated when the modernisation exercise has been completed

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FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 12

1 Crash Landing, Ditching and Precautionary Alighting on Water: Revised requirements proposed by JAC Paper No. 1236 introduced - Chapter 307

2 Electrical Installations: Revised requirements proposed by JAC Paper No. 1323 introduced - Chapter 706.

3 General Flight Test Requirements - Engines: Revised requirements proposed by JAC Paper No. 1239 introduced - Chapter 1001.

4 General Flight Test Requirements - Fuel Systems: Revised requirements proposed by JAC Paper No. 1243 introduced - Chapter 1005.

5 General Flight Test Requirements - Ice Protection Systems: Revised requirements proposed by JAC Paper No.1244 introduced - Chapter 1006.

6 General Flight Test Requirements - Escape Systems and Flotation Gear: Revised requirements proposed by JAC Paper No.1253 introduced - Chapter 1012.

LIST OF MINOR CHANGES INTRODUCED

1 Aircrew Anthropometry: Revised requirements agreed by JAC ex-committee circulation introduced - Leaflet 105/3.

2 Fuel System Lines and Fittings: Revised requirements for V-flange couplings proposed by CE(RAF) and agreed by ex-committee JAC circulation introduced - Chapter 702.

3. Pneumatic Systems: Revised requirements for V-flange couplings proposed by CE(RAF) and agreed by ex-committee JAC circulation introduced - Chapter 703.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 11

1. Introduction Paragraph 11: Revised requirements proposed by JAC Paper No. 1281.

2. Emergency Escape: Revised requirements proposed by JAC Paper No. 1313 introduced - Chapter 102.

3. General Detail Design: Revised requirements proposed by JAC Paper No. 1309 and 1310 introduced - Chapter 400.

4. The Effect of Machining Abuse on Aluminum Alloys: Revised requirements proposed by JAC Paper No. 1317 introduced - Leaflet 402/9.

5. Air Launched Weapons Installations: Revised requirements proposed by JAC Paper No. 1269 introduced - Chapter 720.

6. Strength of Pressurised Air Ducts and Pipes: Revised requirements proposed by JAC No. 1229 introduced - Chapter 730.

7. Rotor Systems: Revised requirements proposed by JAC Paper No. 1278 introduced - Chapter 732.

8. General Test Requirements: Systems and Structures: Revised requirements proposed by JAC Paper No. 1238 introduced - Chapter 1000.

9. Electrical Systems: Revised requirements proposed by JAC Paper No. 1241 - introduced Chapter 1003.

10. Rescue Hoists External Cargo and Pole Equipment: Revised requirements proposed by JAC Paper No. 1285 - introduced Chapter 1017.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 11

1. Appendices No. 2. US Military Specifications, Standards and Handbooks. Existing Appendices No. 1 replaced by new Appendices No. 2. in Parts 1-10.

2. Part 1 new Appendix 1 General and Operational Requirements Military Derivatives of Civil Rotorcraft introduced.

3. Part 2 new Appendix 1 Structural Strength and Design for Flight Military Derivatives of Civil Rotorcraft introduced.

4. Part 3 new Appendix 1 Structural Strength and Design for Operation on Specified Surfaces Military Derivatives of Civil Rotorcraft.

NOTE: This amendment incorporates those JAC (Rotorcraft) Papers which were approved at, or prior to the 145th meeting of this Ministry/Industry Joint Airworthiness Committee.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 10

1 Use of Magnesium Alloys: Revised requirements proposed by JAC Paper No. 1233 introduced - Chapter 400, Leaflet 400/2 and 400/4, Chapter 407, Leaflet 407/3 and Chapter 702.

2 Pressurised Gas Storage vessels: Revised requirements proposed by JAC Paper No. 1264 introduced - Chapter 719, Leaflet 719/2.

3 Alighting Gear: Revised requirements proposed by JAC Paper No. 1245 introduced - Chapter 1008 and Leaflet 1008/1.

4 Protection from the Effects of Nuclear Explosions, Laser Weapons, Chemical and Biological Warfare Agents: Revised requirements proposed by JAC Paper No. 1248 introduced - Chapter 717 and Leaflets 717/0, 717/1, 717/2, and 717/3.

5 Deck Securing Systems: Revised requirements proposed by JAC Paper No. 1252 introduced - Chapter 1009 and Leaflet 1009/1.

6 Control Systems - Mechanical Components: Revised requirements proposed by JAC Paper No. 1260 introduced - Chapter 203 and Leaflets 203/0, 203/1, 203/2, 203/3, 203/4 and 203/5.

7 Powered Flying Controls: Revised requirements proposed by JAC Paper No. 1261 introduced - Chapter 1010 and Leaflet 1010/1.

8 Manoeuvres: Revised requirements proposed by JAC Paper No. 1263 introduced - Chapter 202 and Leaflet 202/1.

9 Gust Loads: Revised requirements proposed by JAC Paper No. 1267 introduced - Chapter 208.

10 Ice Protection Installations: Revised requirements proposed by JAC Paper No. 1274 introduced - Chapter 711 and Leaflets 711/0, 711/1, 711/2 and 711/3.

11 Strength Considerations for Automatic Control Systems: Revised requirements proposed by JAC Paper No. 1275 introduced - Chapter 204 and Leaflet 204/1.

12 Optical Transparent Components: Revised requirements proposed by JAC Paper No. 1277 introduced - Chapter 715 and Leaflets 715/0, 715/1, 715/2, 715/3, 715/4.

13 Waterproofing: Revised requirements proposed by JAC Paper No. 1307 introduced - Chapter 1013

LIST OF MINOR CHANGES INTRODUCED

1 Dehumidification Equipment: Revised requirements proposed by JAC Paper No. 1212 introduced - Chapter 407, Leaflet 407/7.

2 MoD Quality Assurance Authority: Mandatory involvement of MoD QA reduced as proposed by JAC Paper No. 1289.

3 Accident Data Recorders: New requirement for Accident Data Recorders introduced as proposed by JAC Paper No. 1299.

NOTE: This Amendment incorporates those JAC (Rotorcraft) Papers which were approved at, or prior to, the 138th meeting of this Ministry/Industry Joint Airworthiness Committee.

FOR INFORMATION ONLY

LIST OF CHANGES INTRODUCED BY AMENDMENT 9

1 Canopy or Hatch Controls: New requirements proposed by JAC Paper No.1258 introduced - Chapter 103.

2 Colour Standards at Crew Stations: Revised requirements proposed by JAC Paper No.1257 introduced - Chapter 103.

3 Automatic Flight Control System (AFCS) Controls: Revised requirements proposed by JAC Paper No.1235 introduced - Chapter 107.

4 Location and Arrangement of Engine Displays: Revised requirements proposed by JAC Paper No.1259 introduced - Chapter 107.

5 Pilots Cockpit - Location, Actuation and Shape of Airframe Controls: Revised requirements proposed by JAC Paper No.1256 introduced - Chapter 107.

6 Active Control Systems: New requirements proposed by JAC Paper No.1234 introduced - Chapter 207 and Leaflets 207/1 to 207/7.

7 Adhesive Bonding: Revised information proposed by JAC Paper No.1247 introduced Leaflets 402/0, 402/2 and 402/3.

8 Serial Numbers for Repair Control Purposes: Revised requirements proposed by JAC Paper No.1223 introduced - Chapter 404.

9 Flight Tests: Auxiliary Power Systems: Revised requirements proposed by JAC Paper No.1240 introduced - Chapter 1002 and Leaflet 1002/1.

10 Flight Tests: Hydraulic Systems: Revised requirements proposed by JAC Paper No.1242 introduced - Chapter 1004 and Leaflet 1004/1.

11 Flight Tests - Conditioning Systems: Revised requirements proposed by JAC Paper No.1251 introduced - Chapter 1007 and Leaflet 1007/1.

NOTE: Items 9, 10 and 11 have been extensively revised, therefore each page has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 8

1 Guidelines for the Design of Crew Station Lighting and Displays: New information proposed by JAC Paper No.1133 introduced - Leaflet 105/4.

2 Radomes: New requirements proposed by JAC Paper No.1222 introduced - Chapter 209 and Leaflet 209/0.

3 Transmission Systems: Revised requirements proposed by JAC Paper No.1232 introduced - Chapter 705 and Leaflets 705/1 and /2.

4 Conditioning Systems: New requirements proposed by JAC Paper No.1217 introduced - Chapter 731 and Leaflets 731/0 to /4.

NOTE: Item 4 has been extensively revised, therefore each page has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 8

1 Sonar Locator Beacons: Revised requirements proposed by JAC Paper No.1228 introduced - Chapter 100.

2 Transient Voltage Spikes: New requirements proposed by JAC Paper No.1183 introduced - Chapter 706.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 7

1 U.S. Military Specifications, Standards and Handbooks: Lists of U.S. Military Specifications, Standards and Handbooks introduced - Introduction and Appendices No.1 to each Part.

2 Fuel Systems: New requirements proposed by JAC Paper No.1194 introduced - Chapter 702 and Leaflets 702/0 to /6.

3 Health and Usage Monitoring Systems: New requirements proposed by JAC Paper No.1204 introduced - Chapter 727 and Leaflets 727/1 to /5.

4 lnterchangeability: Revised requirements proposed by JAC Paper No.1205 introduced - Chapter 805.

NOTE: Item 2 has been extensively revised, therefore each page has been annotated Issued' not 'Amended' and marginal lines have not been used.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 7

1 Sonar Locator Beacons: Revised requirements proposed by JAC Paper No.1915 introduced - Chapter 100.

2 Control Characteristics: Revised requirements proposed by JAC Paper No.1211 introduced - Chapter 600 and Leaflet 600/6.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 6

1 Design to Resist Birdstrike Damage: New requirements proposed by JAC Paper No.1153 introduced - Chapter 206.

2 Flight and Ground handling qualities: New requirements proposed by JAC Paper No.1186 introduced - Chapter 605 to 607 and associated Leaflets.

3 Propulsion System Installations: New requirements proposed by JAC Paper No. 1188 introduced - Chapter 700, Leaflet 700/0 and Chapter 705.

4 Refuelling and Defuelling Systems: New requirements proposed by JAC Paper No. 1176 introduced - Chapter 701 and Leaflets 701/0 to /4.

5 Hydraulic Systems: New requirements proposed by JAC Paper No.1193 introduced - Chapter 704 and Leaflets 704/0 to /2.

6 Electrical Installations: New requirements proposed by JAC Paper No.1191 introduced - Chapter 706 and Leaflets 706/0 to /3.

7 Radio and Radar Installations: New requirements proposed by JAC Paper No.1192 introduced - Chapter 707 and Leaflets 707/0 to /3.

8 Rescue Hoist and Sonar Hoist Installations: New requirements proposed by JAC Paper No.1179 introduced - Chapter 723.

9 Instrument/Display Installations: New requirements proposed by JAC Paper No. 1189 introduced - Chapter 724, Leaflets 724/0 and /1.

10 Avionic Equipment Installations: New requirements proposed by JAC Paper No. 1190 introduced - Chapter 725 and Leaflets 725/0 to /3.

NOTE: Items 1, 3, 4, 5, 6 and 7 have been extensively revised, therefore each page has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 6

1 Health and Safety at Work Act: Disclaimer introduced - Preface.

2 Loose Article Hazards: New requirements proposed by JAC Paper No.1160 introduced - Chapter 100, para 14 and 15.

3 Rotorcraft Wheel Burst Protection: New requirements proposed by JAC Paper No. 1168 introduced - Chapter 310, para 3.2.8.

4 Asbestos and Asbestos Related Materials: Prohibition of these materials introduced - Chapter 400, para 11.

5 Flutter Clearance Programme for Non-rotating Surfaces: Calculations and tests proposed by JAC Paper No.1196 introduced - Leaflet 500/3.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 5

1 Static strength and deformation: New requirements proposed by JAC Paper No. 1148 introduced - Chapter 200, Leaflets 200/1 and 200/2.

2 Carriage and underslung loads: New requirements proposed by JAC Paper No. 1158 introduced - Chapter 205 and Leaflet 205/1.

3 Flight and ground handling qualities: New requirements proposed by JAC Paper No.1128 introduced - Chapters 600 to 603 and associated Leaflets.

4 Emergency liferaft installations: New requirements proposed by JAC Paper No. 1171 introduced - Chapter 721, Leaflets 721/0 and 721/1.

NOTE: Items 1 and 3 have been extensively revised, therefore each page has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 5

1 Nomenclature: New requirements (STANAG 3647 Edition 3) proposed by JAC Paper No.1166 introduced - Leaflet 105/1.

2 Aircrew anthropometry: New requirements proposed by JAC Paper No.1177 introduced - Leaflet 105/3.

3 Pilot's cockpit - controls and instruments: New requirements proposed by JAC Paper No.1164 introduced - Chapter 107.

4 Use of lead seals on locking wire: New requirements proposed by JAC Paper No. 1167 introduced - Chapter 400, para 8.

5 Allowable values for static strength certification: New requirements proposed by JAC Paper No.1163 introduced - Chapter 401.

NOTE: Item 5 has been extensively revised, therefore it has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 4

1 Pneumatic systems: New requirements proposed by JAC Paper No.1127 introduced - Chapter 703.

2 Static and pitot pressure systems: New requirements proposed by JAC Paper No. 1138 introduced - Chapter 716.

3 Demonstration of limits of flight and manoeuvre envelopes: New requirements proposed by JAC Paper No.1139 introduced - Chapter 905.

4 Engine handling and rotor governing: New requirements proposed by JAC Paper No.1140 introduced - Chapter 906.

5 Autorotation, partially powered flight and engine off landing: New requirements proposed by JAC Paper No.1141 introduced - Chapter 907.

6 Deck landing: New requirements proposed by JAC Paper No.1142 introduced - Chapter 908.

7 Automatic flight control systems: New requirements proposed by JAC Paper No. 1143 introduced - Chapter 909.

NOTE: Items 1 and 2 have been extensively revised, therefore each page has been annotated 'Issued' not 'Amended' and marginal lines have not been used.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 4

1 Sonar Locator Beacons: New requirements proposed by JAC Paper No.1118 introduced - Chapter 100.

2 Tell-tale devices for emergency and standby systems: New requirements proposed by JAC Paper No.1150 introduced - Chapter 105.

3 Detail design and strength of materials: New requirements proposed by JAC Paper No.1149 introduced - Chapter 400 and 402.

4 Locking of threaded fasteners: New requirements proposed by JAC Paper No.1112 introduced - Leaflet 400/4.

5 Master Armament Safety Switch: New requirements proposed by JAC Paper No.1094 introduced - Chapter 710.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 3

1 Internal noise: Revised requirements proposed by JAC Paper No.1121 introduced - Chapter 108, Leaflets 108/1 and 108/2.

2 Design to resist birdstrike damage: New requirements proposed by JAC Paper No. 1120 introduced - Chapter 206.

3 Sealants and sealing: New requirements proposed by JAC Paper No.1066 introduced - Chapter 402, para 7 and Leaflet 402/7.

4 Plastics materials: New requirements proposed by JAC Paper No.1065 introduced - Chapter 408 and Leaflet 408/1.

5 Rubbers: New requirements proposed by JAC Paper No.1067 introduced - Chapter 409.

6 Fire precautions: Revised requirements proposed by JAC Paper No.1064 introduced - Chapter 712, Leaflets 712/0, 712/1, 712/2, 712/3 and 712/4.

7 General handling flight test requirements: New requirements proposed by JAC Paper No.1113 introduced - Chapter 900, Leaflets 900/1, 900/2, 900/3 and 900/4.

8 Ground handling: New requirements proposed by JAC Paper No.1115 introduced - Chapter 901 and Leaflet 901/1.

9 Take-off, hover, low speed manoeuvres and landing: New requirements proposed by JAC Paper No.1116 introduced - Chapter 902 and Leaflet 902/1.

10 Longitudinal trim, stability and control: New requirements proposed by JAC Paper No.1123 introduced - Chapter 903 and Leaflet 903/1.

11 Lateral and directional trim, stability and control: New requirements proposed by JAC Paper No.1125 introduced - Chapter 904 and Leaflet 904/1.

LIST OF MINOR CHANGES INTRODUCED BY AMENDMENT 3

1 Aircrew overnight kit stowages: New requirements proposed by JAC Paper 1104 introduced - Chapter 105.

2 Grading of Rotorcraft parts and assemblies: Deletion of requirements proposed by JAC Paper No.1119 introduced - Leaflet 400/1.

3 Marking of Rotorcraft parts: Revised requirements proposed by JAC Paper No.1144 introduced - Chapter 404.

4 Safety harness components: Revised requirements proposed by JAC Paper No.1099 - Chapter 407.

5 Pressurised gas storage vessels: Revised requirements proposed by JAC Paper No.1111 introduced - Chapter 719.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 2

1 Reduction of Vulnerability to Battle Damage: New design requirements and recommendations proposed by JAC Paper No.1044 introduced - see Chapter 112 and Leaflet 112/1.

2 Protection of Aircrew against Conventional Weapons: New design requirements and recommendations proposed by JAC Paper No.1088 introduced - see Chapter 113 and Leaflet 113/1.

3 Emergency Escape: Revised design requirements and recommendations proposed by JAC Paper No.1068 introduced - see Chapter 102 and Leaflets 102/1 and 102/2.

4 Aero-Elasticity: Revised design requirements and new recommendations proposed by JAC Paper No.1101 introduced - see Chapter 500 and Leaflets 500/1 and 500/2.

5 Vibration and Internal Noise: Revised design requirements and recommendations proposed by JAC Paper No.1O85 introduced - see Chapter 501 and Leaflets 501/1, 501/2 and 501/3.

6 Fragmentation Test Requirements for Gas/Oil Hydraulic Accumulators: New test requirements proposed by JAC Paper No.1096 introduced - see Leaflet 719/4.

7 Folding Components: New design requirements proposed by JAC Paper No.1102 introduced - see Chapter 722.

8 Maintenance: New design requirements and recommendations proposed by JAC Paper No.1093 introduced - see Part 8.

9 Flight Tests - Installation and Structures: New requirements and recommendations proposed by JAC Paper No.1091 introduced in skeleton form - see Part 10.

FOR INFORMATION ONLY

LIST OF IMPORTANT CHANGES INTRODUCED BY AMENDMENT 1

Attention is drawn to the following important changes introduced by Amendment 1.

1 Accident Data and Cockpit Voice Recorders: New requirements for Accident Data and Cockpit Voice Recorders proposed by JAC Paper No.1039 introduced - see Chapter 100 paras 20, 21 and 22.

2 Introduction: Tables 1A and 1B withdrawn and reference made in para 6 to DEF STAN 00-00 (PART 3) SECTION 5 - 'MOD Departmental Standards and Specifications'.

PART 1

GENERAL AND OPERATIONAL REQUIREMENTS

CONTENTS

CHAPTER 100

GENERAL REQUIREMENTS

1 MOCK-UPS

1.1 At an early stage in the development cycle, a full scale Rotorcraft mock-up shall be fabricated by the Rotorcraft manufacturer to function as a design tool in determining the optimum Rotorcraft configuration. (For further details see DEF STAN 05-123 Chapter 230)[1].

2 STANDARD ITEMS (See also Chapter 400)

2.1 The requirements of the appropriate Defence Standards listed in Chapters 1 and 2 of Defence Standard 00-00 (Part 3) Section 1 shall be met. Where these requirements standardize a given item (eg a piece of equipment or a type of fluid or gas), no other item shall be used to perform functions for which this standard item is suitable, unless the use of an alternative item is:

(i) authorised by the Rotorcraft Specification,

(ii) permitted by the requirements of this publication, or

(iii) approved by the appropriate Rotorcraft Project Director (See also the introduction to DEF STAN 00-00 (Part 3) Section 1).

3 INSTALLATION INFORMATION FOR ITEMS OF EQUIPMENT

3.1 All items of equipment shall be installed in accordance with the information given in the relevant Aircraft Equipment Installation Information (AEII), Radio Installation Memorandum (RIM) or other equivalent document (See DEF STAN 05-123 Chapter 205).

4 STRENGTH

4.1 Whenever they are applicable, strength clauses are introduced into the systems and installations chapters. These have, of necessity, to vary in form, but in all cases the applicability of the general strength requirements of the Rotorcraft to the installation shall be considered.

5 VIBRATION (See Chapter 501)

6 TESTS

6.1 PROTOTYPE TESTS

6.1.1 Unless otherwise stated, tests shall be considered as applying to prototype installations, and shall be conducted on at least one prototype.

6.2 COMPONENT TESTS

6.2.1 When a component which is governed by a separate specification is employed, acceptance tests in accordance with the requirements of the appropriate specification shall have been carried out on that component prior to its installation. Where necessary however, such components shall be included in the subsequent functioning tests of the complete installation or system.

7 PREVENTION OF INCORRECT ASSEMBLY OF SYSTEMS

7.1 The requirements of this Para govern the design of systems or parts in systems to prevent incorrect assembly.

7.2 For parts in systems which are likely to cause accidents or major damage if incorrectly assembled, the design shall be such that their incorrect assembly is mechanically impossible. This requirement shall be applied without qualification to the following systems and to such other systems as may be agreed:

(i) flying control systems,

(ii) fuel systems,

(iii) escape systems,

(iv) pitot-static systems,

(v) undercarriage retraction and lowering systems, and

(vi) electrical and hydraulic breakdown joints at transport joints.

Note: If compliance with this requirement involves severe penalties, the Contractor should inform the Rotorcraft Project Director at an early stage in the design.

7.3 For parts in other systems, every effort shall be made to ensure that their incorrect assembly to other parts of the system or to parts of other systems is as difficult as possible. As a minimum requirement, such parts shall be provided with clearly discernible permanent markings which correspond to similar markings on fixed parts of the system or of the Rotorcraft. Colour markings by themselves are not acceptable.

7.4 In fluid systems, the direction of flow shall be shown on the pipes and components as near as possible to the union.

8 CONDITIONS OF OPERATION (See also Chapter 101, Paras 1 and 2)

8.1 All installations and systems shall function correctly under all conditions, on the ground, in flight and at altitude, for which they are required to operate.

9 POWER-OPERATED SYSTEMS

9.1 INDEPENDENCE OF SERVICES
9.1.1 No single failure of any power-operated service or system shall prevent adequate functioning of any other power-operated service which is vital to any of the following:

(i) safety of the Rotorcraft in flight or in landing,

(ii) escape of the crew from the Rotorcraft, and

(iii) ability of the Rotorcraft to perform its operational mission or, if the failure would mean cancelling the mission, return safely.

Note: Failure here means technical defect or malfunctioning from any cause whatsoever. In addition, the spirit of the requirement should be met, so far as is practicable, when failure results from enemy action instead of technical defect.

9.2 PROVISION OF POWER FOR USE IN AN EMERGENCY

9.2.1 Failure of any or all the engines in flight shall not result in the pilot being unable to operate those powered services which are essential to the making of a descent and an emergency landing.

9.2.2 The Contractor's method of meeting the requirement shall be discussed with the Rotorcraft Project Director at an early stage in the design.

10 ALIGNMENT OF DIRECTIONALLY SENSITIVE EQUIPMENT AND WEAPONS (See also Leaflet 100/3)

10.1 The design of the Rotorcraft shall include provision for checking and adjusting the alignment of all directionally sensitive equipment and weapons with respect to a reference datum in the Rotorcraft and to a tolerance acceptable to the whole system. Where practicable, ship-borne Rotorcraft shall have means for checking and adjusting alignment at sea.

Note: Rotorcraft fitted with inertial navigation devices may need to be provided with special facilities for their alignment on the ground and, for ship-borne Rotorcraft, at sea.

10.2 The reference datum should, whenever practicable, be common to all directionally sensitive equipment and weapons. When this is not possible, means of checking alignment between reference and common datums shall be provided.

10.3 Whenever possible, a method of alignment common to all equipment and weapons in the Rotorcraft shall be used.

10.4 The amount of ground equipment required for checking alignment shall be kept to a minimum.

10.5 The Rotorcraft designer shall discuss with the equipment designers concerned and with the Rotorcraft and Equipment Project Directors at an early stage in the design of the Rotorcraft, the alignment procedure to be adopted in respect of all directionally sensitive equipment and weapons. The alignment procedure shall include:

(i) the detailed method of alignment,

(ii) the reference datum of the Rotorcraft, and

(iii) a statement of the alignment accuracy (See Leaflet 100/3, Para 3.1 (iii)).

11 EXTERNAL VISION

11.1 All types of equipment provided for external vision and/or its recording (eg camera windows, lenses, night vision equipment), shall have a means of preventing obscuration (by oil, fuel, dirt, ice, condensed water etc.) both during flight and take-off.

12 EXTERNAL LIGHTS

12.1 LANDING LAMPS

12.1.1 All Rotorcraft shall be provided with landing illumination aids to permit a safe landing on those surfaces for which night landings are envisaged.

12.2 EXTERNAL LIGHTING CIRCUITS

12.2.1 On all Rotorcraft required to operate at night, with the exception of Naval types, the external lighting circuits shall be controlled by a single master switch.

12.2.2 All external lighting shall be dimmable. An ON/OFF/DIM switch located on the throttle is a desirable feature.

12.2.3 There shall be no possibility of downward recognition lights becoming obscured with mud during take-off.

13 DESTRUCTION OF ROTORCRAFT

13.1 The provision of stowage for Rotorcraft destructors shall be discussed and agreed with the Rotorcraft Project Director.

14 LOOSE ARTICLE HAZARDS - CONTROL SYSTEMS

14.1 The aim shall be to design flight and propulsion control systems so that loose articles, from whatever source, cannot enter or jam the system.

14.2 Consideration shall be given during detail design of the Rotorcraft to the prevention of loose articles being generated by parts becoming detached from components and structure and creating a flight safety hazard in the flight or propulsion control systems.

14.3 Where practical adequate access shall be built in to facilitate visual detection of loose articles, and areas where loose articles can lodge shall be reduced to a minimum.

14.4 The Rotorcraft designer shall undertake an assessment of all flight and propulsion control systems in accordance with the principles of JAR25.1309 (JAC Paper No.1169) to show by analysis, and where necessary by ground, flight or simulator tests that the occurrence of a mechanical jamming by loose articles degrading the handling qualities of the Rotorcraft below Level 3 (Chapter 600, Para 7), is extremely improbable.

14.5 Where guards are used they shall comply with the requirements of Paras 15.2 and 15.3.

15 PREVENTION OF ACCIDENTAL DAMAGE

15.1 Certain parts of the Rotorcraft structure and control systems can be vulnerable to accidental damage, for example, by personnel gaining access to the Rotorcraft or its components. The designer shall minimise this risk and shall provide detachable guards for any of these parts. Particular attention shall be paid to flight and propulsion control systems.

15.2 Guards shall be so arranged that loose articles such as screws, nuts, rivets and other material cannot enter and jam the system.

15.3 Guards shall be readily detachable for inspection purposes, using a minimum number of simple and reliable fasteners which are permanently attached to the guard to prevent them creating an additional hazard.

16 JETTISONING OF STORES (See also Chapter 107, Table 9)

16.1 It shall be possible to jettison safely within an appropriate envelope, all external stores that could be critical for operational or flight safety reasons. The details shall be discussed and agreed with the Rotorcraft Project Director.

16.2 When the Rotorcraft is on the ground, it shall be possible to release mechanically any store and/or its jettisonable carrier without entering the cockpit.

17 FOLDING INSTALLATIONS (NAVAL ROTORCRAFT)

17.1 Folding installations for rotors, fuselages, and tail units shall be provided where necessary to meet specified limiting dimensions. It shall be possible to operate such installations in wind speeds up to 45 knots from any quarter and 55 knots from ahead to 30° on each side.

18 EXTERNAL COMMUNICATIONS

18.1 On twin-engined Naval Rotorcraft, the failure of any one engine shall not reduce the efficiency of those electrical services essential for continuous external communications.

19 NBC EQUIPMENT

19.1 Provision shall be made for the incorporation of NBC equipment for all crew members in all operational Rotorcraft.

20 CARRIAGE OF UNDERSLUNG LOADS

20.1 When the Rotorcraft is provided with an installation for the carriage of underslung loads (See Chapter 205), consideration shall be given to the effect of the underslung load on the performance of installed avionic equipment and on other Rotorcraft systems. In particular the effect of the underslung load on the radio altimeter or other Doppler equipment and on the polar diagrams of aerials shall be considered. In particular also, the effect of prolonged flight in a nose-down attitude on fuel flow and lubrication systems shall be considered.

21 ACCIDENT DATA RECORDERS

21.1 All Rotorcraft, except primary training types, shall be fitted with an Accident Data Recorder (ADR). The ADR shall, unless otherwise stated in the Rotorcraft specification, comply with the European Organisation for Civil Aviation Equipment (EUROCAE) specification ED-55, Minimum Operational Performance Specification for Flight Data Recorder Systems.

22 COCKPIT VOICE RECORDERS

22.1 All passenger carrying and multi-crew Rotorcraft shall be fitted with a Cockpit Voice Recorder (CVR). Where no ADR is fitted, there shall be at least 2 cockpit voice channels plus an area microphone channel, each of 30 minutes minimum duration, but preferably lasting for the whole flight period. The durations shall be agreed with the Rotorcraft Project Director. The CVR shall be crash protected as required by the Rotorcraft specification or the Rotorcraft Project Director.

23 SONAR LOCATOR BEACONS

23.1 At least two Sonar Locator Beacons (SLB) shall be installed in a Service aeroplane as follows:

23.1.1 For Rotorcraft fitted with an Aircraft Integrated Monitoring System (AIMS) which have parameters relevant to post-crash analysis recorded on a Crash Survivable Memory Unit (CSMU):

(i) An SLB shall be fitted to the CSMU and shall operate at a frequency of 37.5 kHz or as stated in the Staff Requirement (Air) SR(A).

(ii) A second SLB shall be secured to the airframe and shall operate at a frequency of 9.5 kHz or as stated in the SR(A).

23.1.2 For Rotorcraft fitted with an Accident Data Recorder (ADR) contained in a single removable crash-protected unit:

(i) An SLB shall be fitted to the ADR and shall operate at a frequency of 37.5 kHz or as stated in the SR(A).

(ii) A second SLB shall be secured to the airframe and shall operate at a frequency of 9.5 kHz or as stated in the SR(A).

23.1.3 For Rotorcraft not fitted with an AIMS or an ADR:

(i) An SLB shall be secured to the airframe in the area of the cockpit and shall operate at a frequency of 37.5 kHz or as stated in the SR(A).

(ii) A second SLB shall be secured to the airframe, at a location different to that specified for the requirement at Para 23.1.3 (i), and shall operate at a frequency of 9.5 kHz or as stated in the SR(A).

23.1.4 In addition to the SLBs required above, a further SLB operating at a frequency of 37.5 kHz shall be fitted to each crash-protected CVR (See Para 22.1).

23.1.5 The location of the SLBs on the airframe shall be as agreed with the Rotorcraft Project Director.

23.2 SLBs shall be installed in such a way that they are afforded free contact with water on immersion, and that they do not separate from the recording device or airframe when subjected to impact shock loadings agreed with the Rotorcraft Project Director. Each SLB shall be installed in such a position that the sonic energy is best transferred to the surroundings.

23.3 Consideration shall be given to integrating the SLB function within either the CMSU or ADR/CVR as appropriate.

23.4 The minimum operating duration and the maximum operating depth will be stated in the SR(A) or by the Rotorcraft Project Director.

23.5 Any installation difficulties shall be resolved with the Rotorcraft Project Director.

24 DESIGN REQUIREMENTS FOR AIRCRAFT EQUIPMENT

24.1 Equipment for Service aircraft shall comply, where appropriate, with the requirements of DEF STAN 00-970, BS3G1O0 and other relevant British Standards, and the Preferred Defence Standards listed in DEF STAN 00-00 (Part 3) Section 1.

25 DAMAGE CONTAINMENT - COMPONENTS (EXCEPT ENGINES) INCORPORATING HIGH ENERGY ROTORS

25.1 Unless there is a reasonable assurance that such rotors will not fail, the component shall either be demonstrated as being capable of containing a failed rotor, or be so located that failure will not hazard the Rotorcraft or its occupants.

26 RELIABILITY

26.1 Reliability requirements will be stated in the Rotorcraft specification (DEF STAN 00-40 and 00-41 also refer).

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/64	-

LEAFLET 100/3

GENERAL REQUIREMENTS

ALIGNMENT OF DIRECTIONALLY SENSITIVE EQUIPMENT AND WEAPONS

1 INTRODUCTION

1.1 This Leaflet explains the need for aligning all directionally sensitive equipment and weapons to a common datum and describes the procedure for determining the alignment tolerances required. Further information is given in Report No. AAEE/Tech/236/Nav (Ref.1).

1.2 Advice on alignment matters generally may be obtained from A & A E E Boscombe Down, Salisbury, Wilts.

2 THE NEED FOR ACCURATE ALIGNMENT

2.1 It is important that accurate alignment should be provided on all directionally sensitive equipment and weapons, for example:

(i) compasses,

(ii) sextants,

(iii) astro-trackers,

(iv) drift sights,

(v) accelerometers,

(vi) gyroscope platforms,

(vii) inertial navigation platforms,

(viii) radar (including Doppler radar),

(ix) radio compasses,

(x) bomb and gun aiming equipments,

(xi) guns, rockets and guided weapons, and

(xii) survey cameras.

2.2 Ideally, the datum chosen for alignment purposes should relate the equipment axes to the rotorcraft velocity vector axes. The datum commonly used is the nominal rotorcraft centre line, although this is not necessarily the best; the use of the inertial platform or Doppler main axes are examples of other methods which may be considered.

3 METHOD OF DETERMINING ALIGNMENT TOLERANCES

3.1 The alignment procedure requires that a statement of alignment accuracy be prepared (see Chapter 100, para 10.5). For this purpose, an error budget should be compiled for the overall system, taking into account:

(i) the accuracy required from the overall system,

(ii) the accuracy desired from each element of the system (the rotorcraft designer is responsible for stating this accuracy),

(iii) the accuracy attainable from each equipment or weapon within the system, allowing for errors in the test gear and degradation in alignment between servicings (this should be obtained from the equipment designer or the Rotorcraft Project Director, as appropriate),

(iv) the loss of accuracy in transmitting the required data, e.g., errors in synchros, gearing, etc., (links in the transmission chain which could introduce disproportionately large errors should be examined and the equipment designer should be consulted as to possible methods of improvement), and

(v) changes in alignment due to flexure of the rotorcraft structure arising during normal ground rigging and flight operations.

3.2 The above factors should then be examined as described in Ref.1 to determine the alignment accuracy required from each element of the system.

4 DESIGN RECOMMENDATIONS

4.1 Important factors to be considered in reaching an acceptable alignment policy (in addition to the requirement of Chapter 100, para 10) are:

(i) alignment techniques and equipment should be standardized whenever possible,

(ii) equipment and components should preferably be capable of being removed and replaced without subsequent system re-alignment, and

(iii) where it is not possible to pre-align the sensitive axes of the equipment to the mounting or case with sufficient accuracy, the overall alignment technique should be designed to align the equipment in relation to the sensitive axes rather than to the mounting or case.

4.2 The siting of all equipment and weapons which require alignment should be carefully considered during the initial design of the rotorcraft and they should:

(i) be kept as close together as practicable, commensurate with compatibility requirements, unless it can be shown that the data transmission and flexural effects are not significant to the weapon systems accuracy (when extremely accurate alignment is required, it may be desirable to use a common mounting frame),

(ii) preferably be mounted at or near points of minimum flexure; if this is not feasible, siting on the nominal centre line is preferable to locations outboard of the centre line,

(iii) be easily accessible during the alignment process (where optical methods are used, this feature is of special importance), and

(iv) be placed so that their removal is not necessary to provide access to other equipment.

REFERENCES

Reference	Author	Title
1	Sqn. Ldr. P.H.R. Clifford and Flt. Lt. I.K. Bartley	The alignment of equipment and projectiles on aircraft. Report No. AAEE/Tech/236/Nav, October 1963.

CHAPTER 101

OPERATION IN VARIOUS CLIMATIC CONDITIONS

1 TEMPERATURE LIMITS

1.1 BASIC OPERATIONAL REQUIREMENTS

1.1.1 Unless otherwise specified, all rotorcraft shall be suitable for operation at all heights which the rotorcraft can achieve in all parts of the world.

1.1.2 The rotorcraft shall also be capable of withstanding the temperatures which occur when the rotorcraft is parked in the open without cover.

1.1.3 It shall not be necessary to remove or attach any item, neither shall it be necessary for Service Units to alter the rotorcraft in any way, to cater for the effects of the appropriate maximum and minimum temperature conditions (see, however, Note to para 1.1.4).

1.1.4 Design features incorporated to ensure compliance with the requirements in any one temperature condition shall not appreciably impair the performance or characteristics of the rotorcraft when it is operating under any other condition within the specified temperature envelope.

Note: The requirements of paras 1.1.3 and 1.1.4 may be waived if it can be shown that the fitting of easily removable special equipment to cater for extreme temperature conditions would improve performance appreciably.

1.2 REQUIREMENTS FOR DESIGN (see also Leaflet 101/2)

1.2.1 The rotorcraft and all its instruments and equipment shall function satisfactorily at the temperatures they can acquire in flight in the world-wide outside air temperature limits of Fig.1. Appropriate allowances shall be made for shielding of instruments and equipment, and for the effects of thermal lag.

1.2.2 To comply with the requirements of para 1.1.2 (see also Leaflet 101/2, para 3) and in order to be capable of immediate operation, the rotorcraft and all its instruments and equipment required for operation on the ground or during take-off shall be capable of functioning at all temperatures within the range:

+ 70°C to - 30°C for world-wide use.

1.2.3 In addition to para 1.2.2, to cater for excessive absorption of solar radiation and the occurrence of more severe ground conditions than those given in Fig.1, the rotorcraft and all its instruments and equipment shall not be damaged by the acquisition of temperatures within the range:

+ 90°C to - 40°C for world-wide use.

1.2.4 All parts of the rotorcraft and its instruments and equipment which will be required to function satisfactorily in the landing condition (e.g., undercarriage lowering mechanism) shall do so at the limiting temperatures retained in the landing condition after flight under the most adverse temperature conditions.

1.2.5 When an operating temperature range is stated in the specification, Aircraft Equipment Installation Information (AEII) or equivalent document for an item of equipment, adequate heating or cooling shall be provided to enable the equipment to perform its operational function satisfactorily.

1.2.6 The limiting temperatures for which individual items of equipment are to be designed and tested will be stated in the specification for the equipment. Unless otherwise laid down in this specification, all other airborne and ground instruments and equipment are to satisfy a test schedule on the lines of that given in BS3G100 or DEF STAN 07-55 as appropriate.

1.2.7 All parts of the rotorcraft and those instruments and equipment in the cabin which are essential for use after a failure of the pressurisation or air conditioning system shall function satisfactorily after such a failure.

1.3 TEST REQUIREMENTS

1.3.1 When a new type of rotorcraft is adopted for Service use, ground and flight tests will normally be made at the Experimental Unit operating under Arctic conditions. During these tests, the rotorcraft and its equipment will be required to operate at ground temperatures down to -35°C.

2 HUMIDITY LIMITS (see also Leaflet 101/3)

2.1 The rotorcraft and all its instruments and equipment shall function satisfactorily within the design humidity temperature envelope appropriate to the ambient air formed by a minimum relative humidity of 6% together with:

(i) at altitudes of 10,000 ft. and below, the design maximum humidity - temperature limits shown in Fig.2, and

(ii) at altitudes above 10,000 ft., a relative humidity of 100% associated with the appropriate maximum temperature specified in Fig. 1.

2.2 In certain specific cases (see Leaflet 101/3, para 5) subject to the approval of the Rotorcraft Project Director, the design mean humidity - temperature limits of Fig.3 may be used to define the maximum humidity boundary at altitudes below 10,000 ft.

2.3 The design and test requirements for items of equipment are given in BS3G100.

Note: BS3G100 does not contain tests appropriate for testing equipment which is designed for use following ditching.

3 CLEAR VISION (see also Volume 1 Leaflet 104/3)

3.1 A clear view for piloting, search, navigation, bomb and weapon aiming shall be available under all weather conditions. Sufficient area of transparent panels shall remain clear for these purposes, at all speeds and altitudes in external conditions of snow, rain and icing.

4 FLIGHT IN ICING CONDITIONS

4.1 The Rotorcraft Specification will state when flight in icing conditions has to be possible. The detail requirements for such cases are given in Chapter 711.

5 WEATHERPROOFING (see also Chapter 1013)

5.1 The entry of rain, snow, etc., into the rotorcraft shall be prevented both in flight and on the ground. Attention is also drawn to Chapter 407.

6 DUST AND SAND PROOFING

6.1 There shall be no risk of sand or dust entering any working parts unless the parts are designed to operate satisfactorily under these conditions.

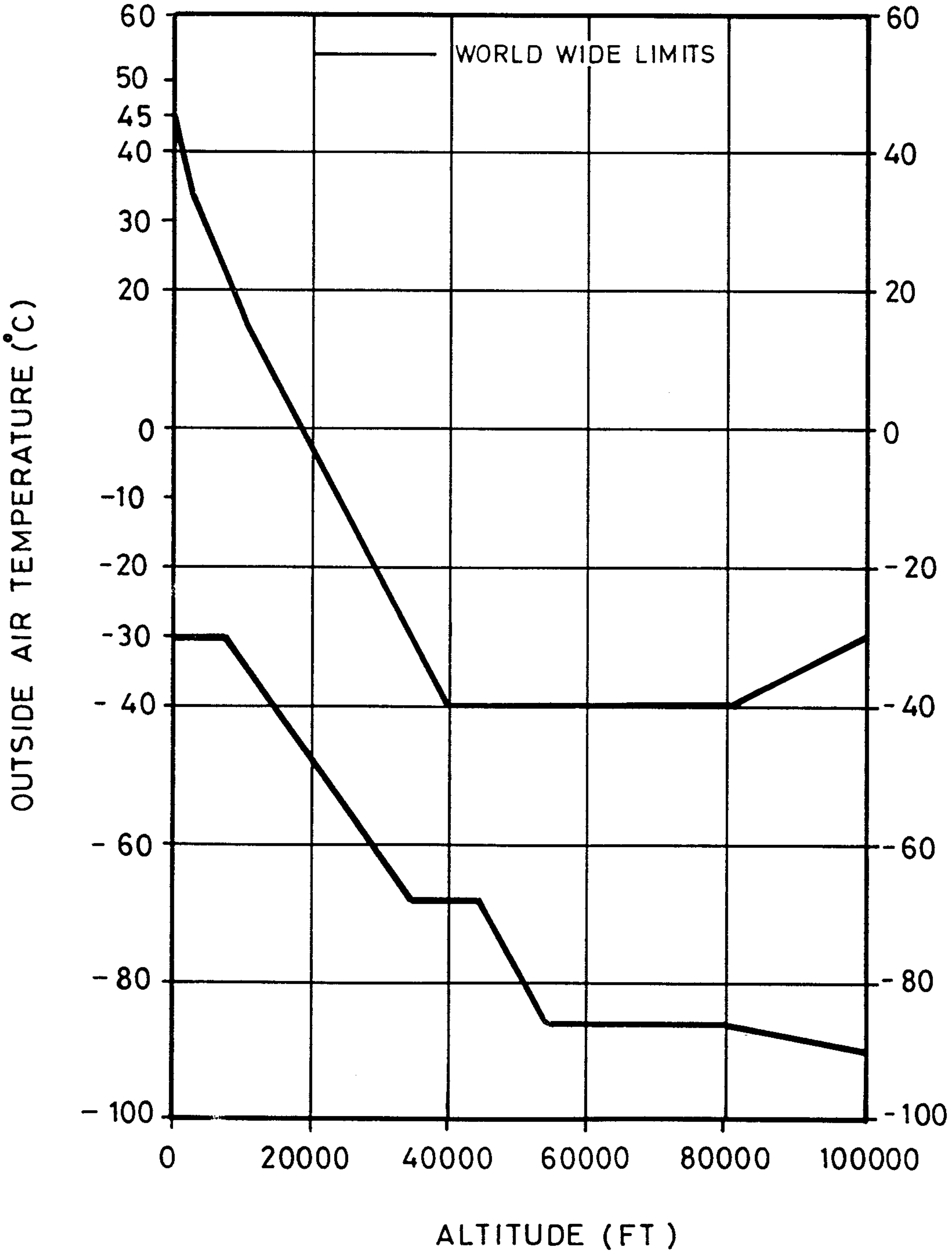

FIG.1 MAXIMUM AND MINIMUM ATMOSPHERIC TEMPERATURES FOR DESIGN PURPOSES

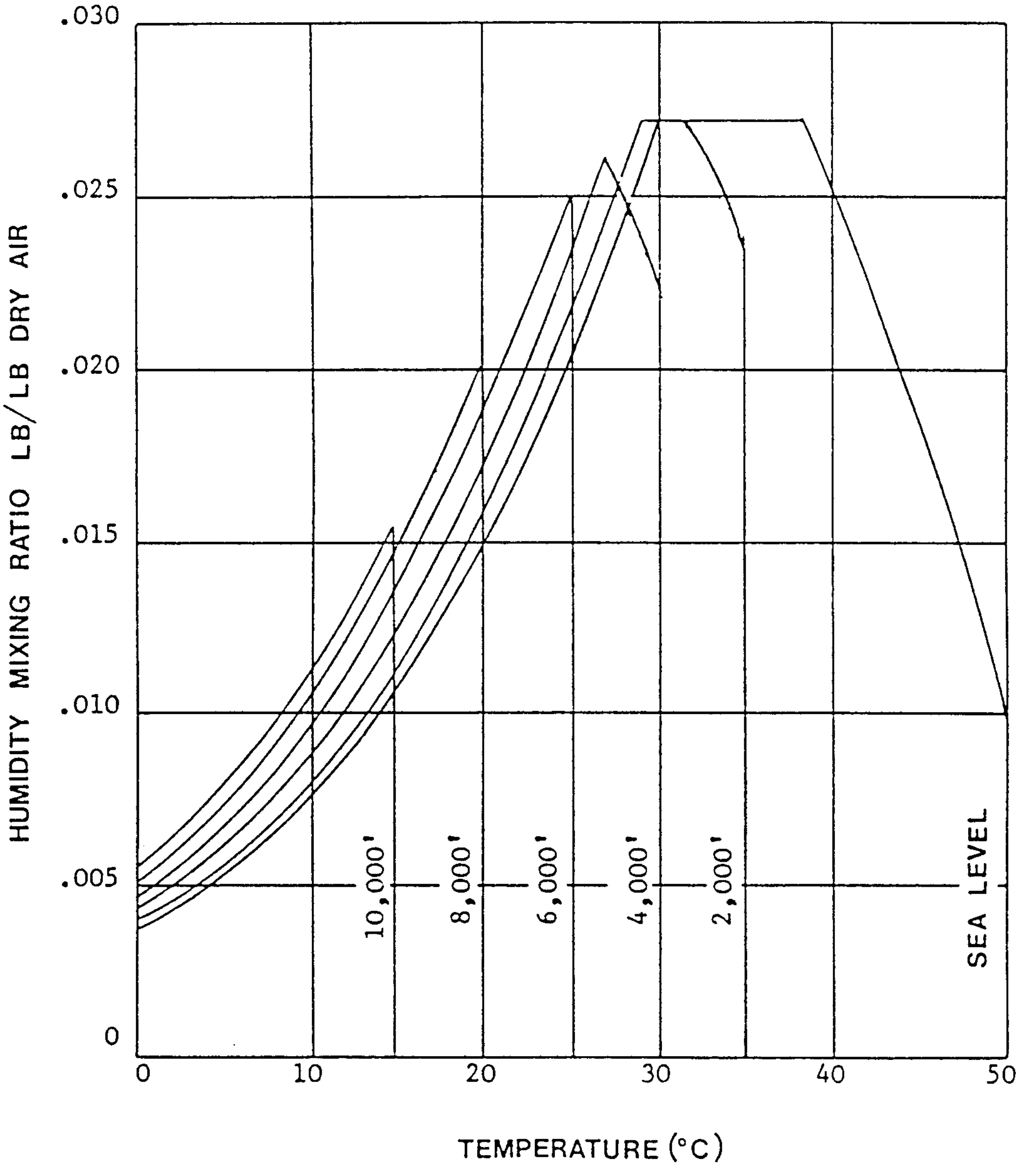

FIG.2 DESIGN MAXIMUM HUMIDITY LIMITS RELATED TO TEMPERATURES AT 0-10,000 FT. ALTITUDE

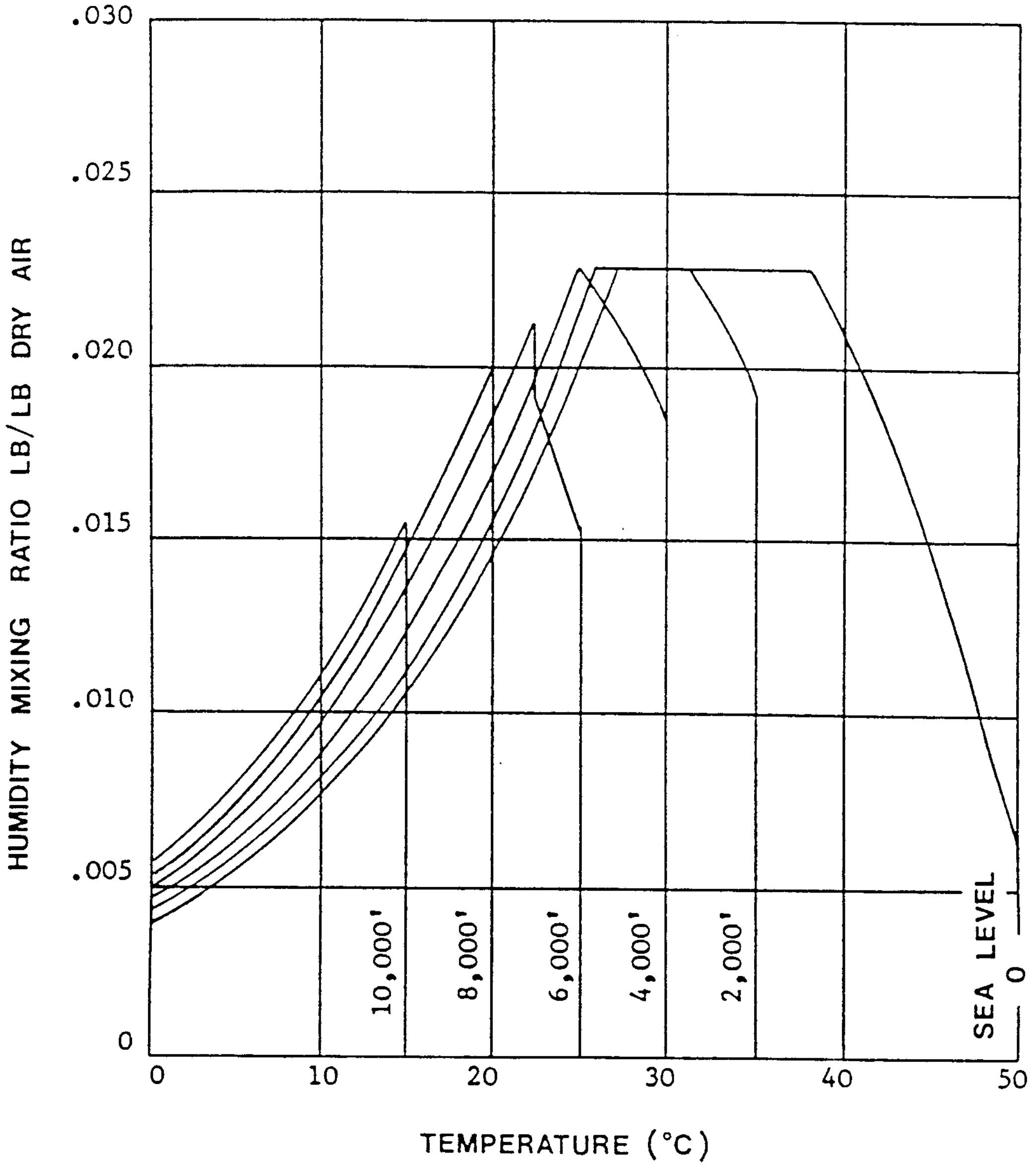

FIG.3 DESIGN MEAN HUMIDITY LIMITS RELATED TO TEMPERATURE AT 0-10,000 FT. ALTITUDE (see para 2.2)

LEAFLET 101/0

OPERATION IN VARIOUS CLIMATIC CONDITIONS

REFERENCE PAGE

Note: See also list of references given in Leaflet 101/1.

MOD(PE)SPECIFICATIONS/PUBLICATIONS.

AvP35	Environmental Handbook for guided weapons.
DTD/RDI381Q	Specification and schedules of tests for airborne heat exchangers.
AD/S & G/1185/P	Aircraft covers.
AD/S & G/1686/E & P	Aircraft lightweight covers.

RAE Reports

Mech.Eng.20	Report of the MOA Working Party on the internal cooling of high speed military aircraft.

RAE Technical Reports

65137	Estimation of heat transfer to flat plates, cones and blunt bodies.
68304	Cabin air requirements for crew comfort in military aircraft.
69188	A method of simulating solar radiation through transparencies in cabin conditioning tests.
81015	Tests of direct infringement air distribution systems for aircrew cooling with plain or low entrainment nozzles.

RAE Technical Notes

Aero 2196	Cabin air conditioning in high-speed flight.
Aero 2346	Introductory notes on the problems of aerodynamic heating.
Mech.Eng.269 ..	Maximum and minimum atmospheric temperatures for aircraft design purposes.
Mech.Eng.338 ..	Thermal environment test of the Bristol 188 research aircraft cockpit and cooling systems.

Mech.Eng.349	..	An experimental evaluation of heat transfer to fuel in integral wing tanks of high speed aircraft.
Mech.Eng.351	..	A study of thermal properties of aircraft cabin insulation using electrical analogue technique.
Mech.Eng.372	..	The development of a cooling air distribution system for the TSR2 pilot's cockpit.
Mech.Eng.405	..	Thermal conduction tests on cabin wall insulation assemblies for a supersonic transport aircraft.

Scientific and Technical Memoranda

4/48	..	Special design features for aircraft operating under very low-temperature conditions.

Flying Personnel Research Committee Papers

FPRC Report 1026.	..	The protection of aircrew against cold following failure of the pressure cabin at altitude.
FPRC Memo 133	..	Specification for thermal comfort in aircraft cabins.
FPRC Memo 134		An analysis of environments compatible with thermal comfort in man.

Defence Standards

07-55 ..	..	Environmental testing of Service Material.
00-1 ..	..	Climatic environmental conditions affecting the design of material for use by NATO Forces in a Ground Role.

British Standards

BS3G100 ..	..	General requirements for equipment use in aircraft.
BS2011 ..	..	Environmental testing.

LEAFLET 101/1

OPERATION IN VARIOUS CLIMATIC CONDITIONS

STANDARD ATMOSPHERIC CONDITIONS

1 INTRODUCTION

1.1 The purpose of this Leaflet is to define standard atmospheres for use in performance estimations.

1.2 It should be noted that the temperature standards laid down in Chapter 101 are the maximum and minimum temperature envelopes for design purposes only and do not define realistic atmospheres which could occur at any time (see Leaflet 101/2).

2 RELATIVE PRESSURES AND DENSITIES

2.1 Table 1 gives relative pressures and densities at altitudes up to 15,000 metres in SI units. Table 2 is an equivalent table in non SI units.

TABLE 1

RELATIVE PRESSURES AND DENSITIES - S.I. UNITS

Air density at sea-level (barometer 1.013250 x 10^5 N/m²
temp 15°C) is 1.2250 kg/m^3

Altitude (Pressure Basis) m	Relative Pressures (I.C.A.O.)	Relative Densities Associated with Conditions Stated				
		International Standard (I.C.A.O.)	Tropical Maximum	Temperate and Arctic Maximum	Tropical and Temperate Minimum	Arctic Minimum
0	1.000	1.000	0.906	0.951	1.138	1.291
500	0.942	0.953	0.862	0.905	1.072	1.190
1000	0.887	0.907	0.820	0.862	1.010	1.097
1500	0.835	0.864	0.780	0.820	0.955	1.011
2000	0.785	0.822	0.741	0.779	0.908	0.949
2500	0.737	0.781	0.703	0.740	0.862	0.892
3000	0.692	0.742	0.668	0.703	0.818	0.837
3500	0.649	0.705	0.633	0.667	0.776	0.792
4000	0.608	0.669	0.600	0.632	0.735	0.750
4500	0.570	0.634	0.568	0.599	0.696	0.709
5000	0.533	0.601	0.538	0.568	0.659	0.670
5500	0.498	0.569	0.509	0.537	0.623	0.633
6000	0.466	0.539	0.481	0.508	0.589	0.597
6500	0.435	0.509	0.454	0.480	0.556	0.563
7000	0.405	0.481	0.428	0.453	0.525	0.531
7500	0.378	0.454	0.404	0.428	0.495	0.500
8000	0.351	0.429	0.380	0.403	0.466	0.470
8500	0.327	0.404	0.358	0.380	0.439	0.442
9000	0.303	0.381	0.337	0.357	0.412	0.415
9500	0.282	0.358	0.316	0.336	0.388	0.389
10000	0.261	0.337	0.297	0.316	0.364	0.365
10500	0.242	0.317	0.279	0.296	0.341	0.341
11000	0.223	0.297	0.261	0.276	0.317	
11500	0.206	0.275	0.244	0.255	0.293	
12000	0.191	0.254	0.229	0.236	0.271	
12500	0.176	0.235	0.214	0.218	0.250	
13000	0.163	0.217	0.201	0.201	0.231	
13500	0.151	0.200	0.186		0.214	
14000	0.139	0.185	0.172		0.197	
14500	0.129	0.171	0.159		0.182	
15000	0.119	0.158	0.147		0.169	

TABLE 2

RELATIVE PRESSURES AND DENSITIES - NON-S.I. UNITS

Air density at sea-level (barometer 29.92 in (1013.2 mbar) temp 15°C) is 0.002378 slugs/ft^3

Altitude (Pressure Basis) ft	**Relative Pressures (I.C.A.O.)**	**Relative Densities Associated with Conditions Stated**				
		International Standard (I.C.A.O.)	**Tropical Maximum**	**Temperate and Arctic Maximum**	**Tropical and Temperate Minimum**	**Arctic Minimum**
0	1.000	1.000	0.906	0.951	1.138	1.291
1,000	0.964	0.971	0.879	0.923	1.098	1.229
2,000	0.930	0.943	0.853	0.896	1.058	1.169
3,000	0.896	0.915	0.827	0.869	1.020	1.112
4,000	0.864	0.888	0.802	0.843	0.983	1.058
5,000	0.832	0.862	0.778	0.818	0.953	1.007
6,000	0.801	0.836	0.754	0.793	0.923	0.970
7,000	0.772	0.811	0.731	0.769	0.895	0.934
8,000	0.743	0.786	0.708	0.745	0.868	0.899
10,000	0.688	0.738	0.664	0.699	0.814	0.832
12,000	0.636	0.693	0.623	0.656	0.763	0.779
14,000	0.587	0.650	0.583	0.615	0.714	0.728
16,000	0.542	0.609	0.545	0.575	0.668	0.680
18,000	0.499	0.570	0.509	0.538	0.624	0.634
20,000	0.460	0.533	0.475	0.502	0.583	0.590
22,000	0.422	0.498	0.443	0.469	0.543	0.550
24.000	0.388	0.464	0.413	0.437	0.504	0.511
26,000	0.355	0.432	0.384	0.407	0.470	0.474
28,000	0.325	0.403	0.357	0.378	0.437	0.440
30,000	0.297	0.374	0.331	0.351	0.405	0.407
32,000	0.271	0.347	0.306	0.326	0.375	0.377
33,000	0.259	0.334	0.295	0.313	0.361	0.362
34,000	0.247	0.322	0.283	0.302	0.347	0.348
35,000	0.235	0.310	0.273	0.290	0.334	
36,000	0.224	0.298	0.262	0.277	0.318	
37,000	0.214	0.284	0.252	0.264	0.303	
38,000	0.204	0.271	0.242	0.252	0.289	
39,000	0.194	0.258	0.232	0.240	0.275	
40,000	0.185	0.246	0.223	0.229	0.263	
41,000	0.176	0.235	0.214	0.218	0.250	
42,000	0.168	0.224	0.206	0.208	0.238	
44,000	0.153	0.203	0.189		0.217	
46,000	0.139	0.185	0.171		0.197	
48,000	0.126	0.168	0.156		0.179	
50,000	0.114	0.152	0.141		0.162	

LEAFLET 101/2

OPERATION IN VARIOUS CLIMATIC CONDITIONS

TEMPERATURE LIMITS FOR DESIGN PURPOSES

1 INTRODUCTION

1.1 This Leaflet gives some information on the derivation of the temperature requirements of Chapter 101.

2 OUTSIDE AIR TEMPERATURES (see Ref.1 for further information and data)

2.1 MAXIMUM AND MINIMUM AIR TEMPERATURES AT ALTITUDE

2.1.1 The maximum and minimum outside air temperature envelopes of Chapter 101, Fig.1 for world wide operation at altitudes have been derived from the analysis of air temperatures measured regularly at various places throughout the world over a number of years and correspond to a frequency of occurrence of 10 days in any one year.

2.1.2 It should be noted that the envelopes do not define realistic atmospheres which could occur at any one time and place. (For standard atmospheres for rotorcraft performance, see Leaflet 101/1).

2.2 MAXIMUM AIR TEMPERATURES AT SEA LEVEL

2.2.1 The maximum air temperature at sea level of +45°C (see Chapter 101, Fig.1), although not the highest air temperature recorded, has been taken as the maximum at which it is likely that rotorcraft required for world wide use will be required to operate.

2.2.2 It should be noted that for full operation in the Mediterranean compliance with the world wide maximum temperature of +45°C is necessary, as air temperatures of + 60°C and over have been recorded in the Sahara not far from the Mediterranean coast.

2.3 MINIMUM AIR TEMPERATURES AT SEA LEVEL

2.3.1 The analysis of the minimum air temperatures at sea level measured in Northern Canada and Alaska indicated that for world wide operation a temperature of -40°C should be specified. The Air and Naval Staffs have however granted a concession and a temperature of -30°C has been specified in Chapter 101. However when subjected to a temperature of -40°C the rotorcraft and its equipment should not be damaged and should be capable of immediate operation without adjustment when the temperature rises to -30°C.

2.3.2 It should be noted that for full operation in North West Europe, compliance with the world wide minimum temperatures is necessary.

2.3.3 When a rotorcraft is adapted for Service use, ground and flight tests will normally be made at the Experimental Establishment operating under Arctic conditions. During these tests the rotorcraft will be operated as nearly as possible under the conditions in which it would be used by Service Units. To allow for the difference between the sample rotorcraft and other production rotorcraft and also to allow for the variation in behaviour of the same rotorcraft under different standards of servicing, the tests will be made at temperatures down to -35°C instead of the design temperature of -30°C.

3 TEMPERATURES ACQUIRED BY ROTORCRAFT AND THEIR EQUIPMENT

3.1 GROUND CONDITIONS

3.1.1 The rotorcraft and its equipment when standing in the sun will acquire much higher temperatures than the maximum outside air temperatures specified in Chapter 101, Fig.1. During tropical trials, temperatures of +65°C to +70°C have been recorded in the cockpits standing in the sun. In addition, tests have shown that the upper metal surfaces under Australian dry tropic conditions reach 87°C about 5-10 times per year and that in extreme cases temperatures of over 100°C may be recorded.

3.1.2 Hence to take account of the solar radiation which can be expected in a hot soak condition at an outside air temperature of +45°C, the requirements of Chapter 101, para 1.2 call for the rotorcraft and its equipment to be able to operate when they have acquired a temperature of +70°C. In addition, to allow for the possibility of higher outside air temperatures and for excessive absorption of heat due to solar radiation, the rotorcraft and its equipment when subjected to a temperature of +90°C should not be damaged and should be capable of immediate operation without adjustment when the temperature drops to +70°C. If however it can be shown that the piece of equipment is unlikely to acquire such high temperatures, e.g. because of its position in the rotorcraft, a case should be submitted to the Rotorcraft Project Director responsible for the rotorcraft so that a concession may be granted.

REFERENCES

No.	Author	Title, etc
1	I.I. McNaughton	Maximum and minimum atmospheric temperatures for aircraft design purposes. R.A.E. Tech. Note Mech. Eng. 269, August, 1958

LEAFLET 101/3

OPERATION IN VARIOUS CLIMATIC CONDITIONS

HUMIDITY CONDITIONS

1 DEFINITIONS

1.1 Humidity is the general term for atmospheric water vapour content and, since this content can be defined in a variety of ways, a variety of humidity terms have been introduced. A clear understanding of certain of these terms is necessary to avoid confusion.

Absolute Humidity	is the mass of water vapour per unit volume of air.
Specific Humidity	is the weight of water vapour per unit weight of air. In meteorology it is applied strictly to the weight per unit weight of moist air but in physics and engineering it is used on both a moist air and dry air basis.
Humidity Mixing Ratio	is the weight of water vapour per unit weight of dry air.
Saturation Humidity	defines the humidity when the air is saturated with water vapour. Saturation humidity increases with rise in temperature. At temperatures below 0°C the air can be saturated with respect to either ice or water but the values are not identical. It is recommended that saturation with respect to water should be used in the temperature range 0°C to -15°C and with respect to ice at temperatures below -15°C. Air saturated with respect to water is super-saturated with respect to ice.
Percentage Saturation	is the ratio, expressed as a percentage, of the actual humidity mixing ratio to the saturation humidity mixing ratio at the same temperature. It is not always the same as relative humidity.
Relative Humidity	is the ratio, expressed as a percentage, of the partial pressure of the water vapour to the saturation vapour pressure at the same temperature.

2 HUMIDITY MIXING RATIO

2.1 If the laws of ideal gases are assumed to apply to water vapour air mixtures the humidity mixing ratio, W, can be calculated to be:-

$$W = 0.622\, p/(P-p) \text{ lb./lb. dry air} \qquad (1)$$

Where p = partial pressure of water vapour, and

P = total pressure of the mixture.

Although water vapour deviates more than air from ideal gas behaviour the accuracy obtained by using equation (1) is acceptable for engineering purposes.

2.2 Since Relative Humidity, RH, can be expressed as:-

$$RH = 100 \ p/p_s \qquad (2)$$

Where p_s = saturation vapour pressure,

equation (1) can be reduced to:-

$$W = 0.622 \, p_s \frac{RH}{100} \Big/ \left(P - p_s \frac{RH}{100}\right) \quad \text{lb./lb. dry air} \quad (3)$$

2.3 Equation (3) can be used to determine the humidity mixing ratio at any required value of RH by substituting appropriate values of P and P_s obtainable from standard reference tables. For convenience, values of saturation vapour pressure in mm. Hg. over the range -65°C to +50°C are given in Table 1.

3 PERCENTAGE SATURATION

3.1 Percentage saturation, %S can be expressed as:

$$\%S = 100 \quad W/W_s$$

Where W_s = Saturation Humidity Mixing Ratio.

Using equations (1) and (2) it can be shown that:

$$\%S = RH \times \frac{P - p_s}{P - p}$$

Thus, except at saturation where $p = p_s$, Percentage saturation is not equal to Relative Humidity. However, at normal sea level temperature and pressure P is generally large compared to p and ps so that the ratio $(P-p_s)/(P-p)$ tends to unity and %S tends to equal RH. As P decreases or temperature rises (P_s increases) the ratio departs from unity and the two can no longer be assumed equal.

4 HUMIDITY LIMITS

4.1 The maximum humidity limits of Figs. 2 and 3 of Chapter 101 were derived from sea level vapour pressure values by assuming vertical air mass movement. In general these upper boundaries are formed by two curves the one rising from the left being fully saturated air and the one falling to the right being unsaturated air.

4.2 In Fig. 2 the sea level maximum limit was derived mathematically from that of Fig. 3 to give a boundary which would be exceeded on average only once per month using the standard variation in average monthly mean vapour pressure for two stations, Aden and Habbaniya, assumed to be typical of warm humid zones and hot dry zones. The limits of Fig. 2 are therefore maximum limits which would not generally be exceeded more than once per month.

4.3 The sea level limit of Fig. 3 was derived from an envelope of average monthly mean vapour pressure and average monthly mean maximum temperature values from recording stations in warm humid and hot dry climates. This sea level limit is therefore a mean limit for the stations with the most humid conditions. The altitude limits of Fig. 3, being derived from sea level limit, are therefore also mean limits.

5 MAXIMUM HUMIDITY LIMITS FOR DESIGN PURPOSES

5.1 As stated in Chapter 101, the maximum humidity limits of Fig. 2 of the Chapter shall be used for design purposes at altitudes below 10,000 ft., but in certain specific cases (see para 5.2), with the prior approval of the Rotorcraft Project Director, the mean humidity limits of Fig. 3 may be used.

TABLE 1

SATURATION VAPOUR PRESSURE

Temp °C	Saturated Vapour Pressure (p_S) mm.Hg.	
	Over Ice	Over Water
-65	0.0041	
-60	0.0081	
-55	0.0157	
-50	0.0295	
-45	0.0540	
-40	0.0962	
-35	0.1675	
-30	0.2855	
-25	0.4790	
-20	0.7740	
-15	1.239	1.434
-10	1.945	2.148
-5	3.010	3.160
0	4.580	4.580
5		6.535
10		9.200
15		12.776
20		17.525
25		23.75
30		31.82
35		42.20
40		55.30
45		71.90
50		92.50

CHAPTER 102

EMERGENCY ESCAPE

1 INTRODUCTION

1.1 The requirements of this Chapter are, unless otherwise specified, applicable to all types of Rotorcraft and aim to ensure that the occupants will be able to leave the Rotorcraft quickly and safely in an emergency.

2 GENERAL PRINCIPLES

2.1 The conditions covered relate to emergency escape:

(i) after crash landing or ditching (Chapter 307),

(ii) from Rotorcraft on the ground

2.2 Every occupant, when wearing the clothing and personal equipment specified in the Aircrew Equipment Assembly Schedule for the Rotorcraft, shall be able to leave the Rotorcraft safely, irrespective of its attitude, by a suitable exit in the shortest possible time under the conditions of Para 2.1.

2.3 To facilitate passage to the exits, footholds, handholds and ladders shall be provided where necessary. They shall be of rigid construction, and shall be permanently fixed in position, except that they may fold if they can be brought into use immediately, and are unlikely to jam as a result of structural distortions during a crash landing.

2.4 Exits and their approaches shall be free from obstructions and projections where clothing or personal equipment might be caught.

3 EXITS

3.1 Emergency exits shall be provided to facilitate the rapid evacuation of all occupants in the event of a crash landing. It is acceptable and often desirable that doors, hatches etc, primarily provided for other reasons (e.g. normal entrance door) should serve also as emergency exits. Any exit may serve for more than one emergency condition provided that the requirements stated for each such condition are met.

3.2 CREW EXITS

3.2.1 Emergency exits shall be provided in the light-crew compartment and shall be such as to afford the flight crew means of rapidly leaving the Rotorcraft. Two such exits shall be provided and shall be either:

(i) located one on each side of the Rotorcraft, or

(ii) one on the side of the Rotorcraft and the other in the roof or floor.

3.2.2 Emergency exits shall be provided for all aircrew without easy access to flight crew or passenger compartment emergency exits. Two such exits shall be provided as in Para 3.2.1 and shall be at least the size of Type IV in Para 3.3.1.

3.3 PASSENGER EXITS

3.3.1 Types of Exits. Passenger exits are classified by type as follows:

(i) Type I. A rectangular opening of note less than 610mm wide by 1200mm high, with the sill at floor level. Corner radii shall be not greater than 1.3 of the width of the exit.

(ii) Type II. The same as Type 1, except that the opening is not less than 510mm wide and 1120mm high and, if located over a wing or sponson, a step up inside the Rotorcraft of not more than 250mm and a step down outside the Rotorcraft of not more than 430mm is permitted.

(iii) Type III. A rectangular opening of not less than 510mm wide by 910 mm high with a step up inside the Rotorcraft of not more than 510mm and, if located over a wing or sponson, a step down outside the Rotorcraft of not more than 690mm. Corner radii shall not be greater than 1.3 of the width of the exit.

(iv) Type IV. A rectangular opening of not less than 480mm wide by 660mm high, with a step up inside the Rotorcraft of not more than 740mm and, if located over a wing or sponson, a step down outside the Rotorcraft of not more than 910mm. Corner radii shall not be greater than 1.3 of the width of the opening.

NOTE: Larger openings than those specified will be acceptable, whether or not of rectangular shape, provided the prescribed openings can be inscribed therein, and further provided that the base of the openings affords a flat surface not less than the width specified.

(v) for Rotorcraft having a maximum weight not exceeding 2730 kg (6,000 lb) the emergency exit shall consist of an opening that will admit an ellipse not less than 483mm x 660mm (19in x 26in).

3.3.2 Location of Passenger Exits:

(i) the optimum fore and aft location of emergency exits shall be decided on Rotorcraft bearing in mind the relevant considerations which will include but not necessarily be confined to:

(a) the disposition of passengers in the fuselage and the ease with which they can reach the exits;

(b) the probability of occurrence of damage to different parts of the fuselage in Emergency Landing conditions;

(c) the need to avoid passengers leaving the Rotorcraft in areas where dangerous conditions (e.g. spilt liquids, hot engine parts, rotors, propellers) can be encountered;

(d) the need to avoid areas that might become potential fire hazards in an Emergency Landing or Crash Landing.

(ii) sufficient additional exits shall be provided to allow evacuation should the Rotorcraft come to rest on its side after an emergency landing, unless the probability of this situation can be shown to be extremely remote.

3.3.3 Number of Passenger Exits. Emergency exits shall be provided on each side of the fuselage in accordance with the following:

(i) the number and size of emergency exits shall be related to the seating capacity as shown in Table 1.

(ii) for Rotorcraft having a Maximum Weight not exceeding 2730kg (6,000lb) there shall be one emergency exit as defined in Para. 3.3.1 (v) on each side of the fuselage.

3.3.4 Where the passenger accommodation is divided into two or more compartments, each compartment shall be provided with exits, unless the passage ways connecting compartments are such that they would not become blocked or retard passenger movement in the event of a Crash Landing.

3.3.5 Access to Passenger Exits. Easy means of access to the exits shall be provided to facilitate use even in darkness (see also Para 4); exceptional agility shall not be required of passengers using the exits. The following shall be complied with:

(i) passage ways between individual passenger compartments and passage ways leading to normal exits and Type I and Type II exits and any exits where an escape chute is required shall be unobstructed and not less than 510mm wide.

(ii) the main aisle at any point between the seats shall be not less than:

(a) for Rotorcraft having a maximum passenger seating capacity of more than 19 persons, 380mm wide up to a height above the floor of 630mm and 510mm wide above that height.

(b) for Rotorcraft having a maximum passenger seating capacity of 19 or less persons, 310mm wide up to a height above the floor of 630mm and 510mm wide above that height.

(c) for Rotorcraft having a maximum passenger seating capacity of 10 or less persons, 310mm wide up to a height above the floor of 630mm and 460mm wide above that height.

(iii) adjacent to each exit where an escape chute or similar means of escape needs to be used there shall be sufficient space to allow a crew member to assist in the evacuation of passengers without reduction in the unobstructed width of the passage way to such an exit.

Access shall be provided from the main aisle to all Type III and IV exits and such exits shall not be obstructed by seats or other protrusions to an extent which would significantly interfere with their use.

3.4 DITCHING EMERGENCY EXITS

3.4.1 With the Rotorcraft in the configuration appropriate to a planned ditching, the most adverse static water level shall be established. At least one exit per side, of Type III or larger shall be located above the water level so established. For passenger seating capacities greater than 59, additional exits shall be provided such that there is at least one Type III emergency exit located above the water level for each additional 35 passenger seats or part of 35.

3.5 MARKING OF EXITS

3.5.1 All emergency exits shall be clearly marked so that their intended use and their means of operation are obvious to the occupants and, as appropriate, to rescue personnel approaching the Rotorcraft outside. Details of the required markings are given in Chapter 103.

3.6 CONTROLS

3.6.1 Each emergency exit shall be openable and jettisonable, when applicable, by one hand by a single positive movement of a single control, operate by a pull between 111N and 178N. When the exit is jettisonable directly outwards, the control shall be such that there is no risk of the operator's hand being pulled outward by the cover.

3.6.2 Consideration should be given to the operation of external controls by members of crash/rescue crews who may be wearing bulky protective clothing. Crash/rescue crew members should be able to operate the external controls whilst wearing the full protective clothing outfit. For this purpose the Design Authority should use Anthropometric data for Metacarpal Breadth relative to 95th percentile man[1], making due allowance for the wearing of protective gloves/gauntlets. The Aircraft Project Director shall be approached to confirm details of the protective clothing outfit to be entered for unless this information is provided in the Aircraft Specification.

3.7 EXIT OPERATION

3.7.1 The doors and hatches provided for emergency escape shall:

(i) jettison outwards,

or

(ii) pull away completely from their frames, and be capable of being jettisoned through the exit after an emergency landing,

or

(iii) be hinged to open outwards.

3.7.2 When the door or hatch is hinged, means shall be provided to open it against all aerodynamic loads which may occur, and to lock it in an open position in a manner which does not reduce the size of the exit.

3.8 PRECAUTIONS AGAINST INADVERTENT OPERATION

3.8.1 Design precautions shall be taken to reduce to a minimum the risk of inadvertent or accidental release or jettison of exits by crew or passengers. Inadvertent release or jettison of exits shall not be possible from other causes including:

(i) vibration or buffeting,

(ii) deformation or flexing due to loads within the fully factored flight envelope,

(iii) temperature variations on the airframe etc.

3.8.2 It shall be impossible for the locks on hatches or doors to be left partially engaged. The setting of locks when closed shall be such that inadvertent disengagement or disturbance is impossible. There shall be clear indications to the appropriate crew member to show when all the locks are properly engaged. Electrical indicators are acceptable only where functional checks of the indicating system and direct visual inspection of the locks can be made during the pre Take-Off Drill. After the release or jettison mechanism has been operated, it shall be impossible to restore the control to its normal position until the locking mechanism has been reset.

TABLE 1

	Emergency Exits each side of the fuselage			
Passenger Seating Capacity (inclusive)	Type I	Type II	Type III	Type IV
1 - 19	-	-	1	-
20 - 39	-	1	-	1
40 - 59	1	-	-	1
60 - 79	1	-	1	-
80 - 109	1	-	1	1

NOTES:

(1) It is not the intention to require that exits necessarily be at locations diametrically opposed to each other.

(2) It is acceptable to provide two Type IV exits instead of each Type II exit required.

(3) Where compensating factors exist which justify an increase in seating capacity beyond those specified in Table 1 such an increase up to a maximum of ten extra passengers is permissible with the agreement of the Rotorcraft Project Director.

4 EMERGENCY ESCAPE/EVACUATION ILLUMINATION[2]

4.1 The following requirements shall apply in the design and location of power supplies, controls, lighting fixtures and associated equipment used to provide emergency escape illumination in Rotorcraft..

(i) emergency escape illumination shall be designed so that no beam of light is directed into the occupant's eyes in such a way as to compromise their ability to escape.

(ii) emergency escape illumination shall be provided independent of normal

(iii) the emergency escape illumination shall be so designed, installed and located in such a manner that will minimise damage to or loss of any portion of the emergency escape illumination as a result of ditching or an emergency landing.

(iv) break-up of the fuselage shall not render any portion of the emergency

(v) emergency escape illumination shall be continuously lighted or automatically energised when an emergency occurs.

(vi) if any automatically energised system of emergency escape illumination is used, provision shall be made for alternate manual operation from a single location easily accessible to a flight crew member (see also Chapter 103).

(vii) the emergency escape lighting system shall provide not less than 500 cd/m^2 ambient illumination at all exits and in the centre of aisleways leading to exits measured at seat arm rest height and in all aircrew stations and passenger compartments.

(viii) all exit signs, arrows and placards shall be electrically lighted or self-luminous and shall be no less than 500 cd/m^2.

5 ABANDON AIRCRAFT COMMAND SIGNAL

(See Chapter 105 and Chapter 107)

6 EMERGENCY ALIGHTING

6.1 GENERAL

6.1.1 The design of the Rotorcraft shall be such as to afford the occupant as much protection as is reasonably practical in the event of an emergency alighting on land or water. (See Chapter 307).

6.1.2 There shall be no intrusions or obstructions to the crew members movements during deceleration. when correctly strapped into the seat.

6.2 JETTISONABLE OR FRANGIBLE HOODS OR HATCHES

6.2.1 No action shall be required of the appropriate crew members other than that necessary to operate the jettison control or fragmenting device.

7 TESTS

7.1 JETTISONABLE DOORS AND HATCHES

7.1.1 Prototype installations or new designs of jettisonable doors and hatches and their associated locking and jettisonable mechanisms, shall be subjected to a comprehensive series of continual tests on the general lines given in Leaflet 102/1 to ensure that:

(i) the strength of the component and its adjacent structure and the design of the locking mechanisms are adequate under the most adverse combinations of Para 3.8.1, and

(ii) the component can be jettisoned with certainty and safety when the release mechanism is operated.

The tests should be completed before the first flight.

7.1.2 During the development flight programme, close and critical inspection of the door and hatch locking mechanisms shall be made after every flight to establish that the locking mechanism is satisfactory.

7.2 For additional strength, stiffness, and energy absorption requirements see Chapter 307.

7.3 EMERGENCY EXITS

7.3.1 The proper functioning of all emergency exits shall be demonstrated.

7.4 ESCAPE FROM ROTORCRAFT ON THE GROUND

7.4.1 When the Specification for a Large Rotorcraft indicates that it should be possible for the Rotorcraft to be configured to have a seating capacity of more than 44 passengers, it shall be demonstrated in accordance with JAR 29-803 that a full compliment of crew and passengers can be evacuated from the Rotorcraft to the ground within 90 seconds.

7.4.2 For other types of Rotorcraft it shall be demonstrated that escape of the maximum number of occupants is achievable within the time defined in the Rotorcraft Specification.

REFERENCES

Reference	DEF STAN	ASCC Air Standard	STANAG
1	00-25	-	-
2	-	10/66	-

LEAFLET 102/1

EMERGENCY ESCAPE

GROUND TEST SCHEDULE FOR JETTISONABLE DOORS AND HATCHES

1 INTRODUCTION

1.1 This Leaflet describes the ground tests required by Chapter 102, para 7.1 which are necessary to ensure that doors and hatches cannot be jettisoned inadvertently in the air or on the ground and that they release when required and, when released, both separate from the rotorcraft and pursue paths endangering neither the rotorcraft nor its occupants.

2 SCOPE OF GROUND TESTS

2.1 Because of the widely variable conditions under which rotorcraft operate, the tests should be as comprehensive as possible covering every foreseeable condition to which the door or hatch might be exposed in service either on the ground or in the air. It is recognized that such tests can in general, be made to simulate actual conditions only very approximately, but if the tests are made sufficiently searching and faults uncovered by them are rectified, then flights can be undertaken with some confidence.

3 TESTS FOR POSITIVE ATTACHMENT OF DOORS AND HATCHES

3.1 A series of tests should be made to check that:

(i) the strength and stiffness of the locking mechanism are adequate to prevent inadvertent loss of the door or hatch under the most adverse combination of loads that can be sustained in flights due to:

(a) aerodynamic effects,

(b) vibration effects, and

(c) inertia forces due to flight manoeuvres.

In the above test, account should be taken of structural distortion due to the most adverse temperature conditions to be expected, and the most adverse combination of component tolerances on assembly of the complete mechanical system in the rotorcraft.

(ii) the door or hatch cannot be engaged without fully locking the mechanism and without adequate warning to the crew, or that during periods of vibration or buffeting, the moving parts of the toggle or other locks will not:

(a) creep into an unsafe position, or

(b) disengage altogether.

3.2 If possible the test of para 3.1(i) should be carried out in a climatic chamber. In making vibrations and buffeting tests of para 3.1(ii), it is preferable to use the cockpit and cabin section of an actual rotorcraft, but a representative rig may be used provided it is possible to produce it in a reasonably close approximation of the vibration that would have been set up in the rotorcraft.

3.3 During the tests, close examination of the components of the locking mechanism should be made at intervals when the structure is unloaded, to ascertain:

(i) whether any uneven distribution of loads on the locking members has occurred,

(ii) whether all locking members have remained adequately and correctly engaged throughout the test, (recording measuring devices should be fitted for this purpose),

(iii) whether deformation or, maladjustment of the locking members has occurred, and

(iv) whether any of the moving parts show signs of wear that might lead to eventual failure.

4 DOOR OR HATCH JETTISONING TESTS

4.1 Ground tests should be made to check that the door or hatch will jettison with certainty and safety when required to do so. These should include full scale tests under load on the ground with the door or hatch locked to the rotorcraft or representative structure to demonstrate that:

(i) the release functions correctly from all stations both free and under load,

(ii) the release functions correctly from all stations both free and under load after a cycle of vibration and buffeting,

(iii) the release functions correctly at high and low temperature, taking account of temperature lag,

(iv) the power supplied for forced jettisoning under all those conditions is adequate, and

(v) the release control loads are within the limits specified.

4.2 After each release test called for under para 4.1, the whole of the mechanism and the door or hatch structure should be examined for deformation or other damage and for signs that inadvertent release when under load in flight might be possible.

LEAFLET 102/2

EMERGENCY ESCAPE

DOOR AND HATCH LOCKING MECHANISMS

1 INTRODUCTION

1.1 This Leaflet points out some of the disadvantages inherent in door and hatch locks which employ the principle of geometric locking. It is recommended that in future designs of this sort, a means of preventing the movement of the linkage under any conditions that are likely to be encountered should be incorporated.

2 THE GEOMETRIC OR TOGGLE LOCK

2.1 A geometric lock is obtained when the pieces of a four bar linkage are moved into such a position that the force producing motion passes through at least two moving and one fixed centre. In this position the system is said to be at dead centre and its configuration may be either a triangle or a straight line. Both figures are stable structures and there is no tendency to move the linkage further.

2.2 It is common practice however, to allow the crank (i.e., a link with one fixed and one moving centre), to go over dead centre so that there is a resultant force opposing any movement of the crank in the opposite direction. In some designs a spring is added to assist in keeping the crank on the safe side of dead centre.

3 THE ADVANTAGES OF GEOMETRIC OR TOGGLE LOCKING SYSTEMS

3.1 The most important advantages of the geometric type of lock are the lack of complication and the fact that locking and unlocking forces can be kept low and can easily be made to conform with the specified requirements for the magnitude of the operating force in a given system.

3.2 Under static conditions, a system of locking based on this principle is extremely stable and only the actual shearing or the bending of links can cause the device to unlock inadvertently when correctly adjusted. The design of the mechanism should, however, take into account the effect of incorrect adjustment and wear, wherever possible.

4 PRECAUTIONS FOR GEOMETRIC LOCKS

4.1 The main disadvantage of such a device is the fact that the lock is not positive. In certain applications a device of this kind may be subject to high accelerations, vibration, fluctuating loads or relative movement between different parts of the locking system and if this happens, slight movements of the crank may result either from inertia forces or from on and off loading. If the movement of the crank is over dead centre to the unsafe side, the resultant force from the loading on the lock will eventually cause it to open.

4.2 The adjustment of a geometric lock is very critical, since the actual position of the dead centre is precise and strictly speaking there can be no tolerance on the crank position either way. However, it is common practice to adjust the linkage so that the crank is stopped slightly over dead centre and a small tolerance of -0 to Δ° can be allowed. The angle of the crank over dead centre cannot be permitted to be too great, otherwise the resultant force due to the loading on the hook or latch will result in unduly high operating forces for unlocking. Also it will be necessary to employ stronger pivots and stops.

4.3 The large number of parts usually involved in the system, connecting the locks or latches with their control points, means that the build up of tolerances on assembly must be carefully controlled. Otherwise it may not be possible to adjust the crank in the locking mechanism within the correct over dead centre tolerance.

4.4 In some installations a number of locking mechanisms are connected together in series with their control rods so that the tolerance build up one way or the other is critical. This arrangement does not make for easy adjustment of the over dead centre positions of the crank, since any adjustment carried out on one lock is passed through the series connecting rods to the others. An easier arrangement to manage is one in which the control rods are run in parallel with the locks, for then adjustments can be made to each lock separately without disturbing the others.

5 MALFUNCTIONING OF LOCKING SYSTEMS

5.1 The arc through which cranks in a system employing geometric locks must rotate to provide catastrophic failure is obviously small. Its magnitude will depend on two things:

(i) the over dead centre adjustment, and

(ii) the friction in the whole system.

The friction must not be too great, otherwise the operating forces will be unduly high. On the other hand, if it is very low only a small movement of the crank to the unsafe side of dead centre will be significant.

5.2 Conditions most likely to produce malfunctioning of geometric locks are:

(i) relative movement between different parts of the system produced by flexing of the airframe,

(ii) expansion and contraction of the airframe, linkage system etc, due to wide temperature changes, and

(iii) vibration.

Any or all of these conditions are likely to produce failure, since the locking system is not positive and gradual creep of the cranks to the unsafe side of dead centre can take place.

6 PREVENTION OF FAILURE THROUGH CREEP

6.1 In principle the solution to the problem of creep from any cause is simply to fit a retractable stop in the path of the crank which cannot then move into an unsafe position until the stop is removed.

6.2 Practically, this could only be done at the expense of some extra complication, since the withdrawal of the stop would have to form part of the sequence of operations involved in unlocking the system. In addition the stop itself would have to be insensitive to all those conditions which could cause disturbance in a geometric lock. On the other hand the physical dimensions of such a stop could be relatively small since the shearing force due to the crank would be low.

7 OTHER CAUSES OF FAILURE

7.1 Other failures of door and hatch locking mechanisms are mainly brought about by:

(i) incomplete engagement,

(ii) inaccurate adjustment,

(iii) wear at pivots, etc, and

(iv) poor maintenance.

In general such failures could be made less likely by making the whole locking circuit as accessible as possible and by fitting visual indicators at each locking point in addition to a central warning system.

8 INSPECTION OF LOCKING MECHANISMS (see also Leaflet 102/1, para 4.2)

8.1 As part of the development flight test programme, close and critical inspection of the door and hatch locking mechanisms should be made after every flight, to find out whether:

(i) there has been any tendency for the moving parts of toggle or other locking devices to creep into unsafe positions,

(ii) there has been any distortion or malfunction of the connecting links under load, or

(iii) there are any signs of wear that could cause failure if allowed to progress.

8.2 Wherever possible, recording measuring devices should be fitted to the locking mechanism since visual checks may not always be reliable due to the inaccessibility of parts of the mechanical system.

CHAPTER 103

OPERATIONAL COLOURING AND MARKINGS

1 INTRODUCTION

1.1 External colouring and markings shall be in accordance with AP119A-0601-0 Chapter 3.[4]

1.2 This Chapter specifies the operational requirements for the colouring and markings to be used in the interior of all rotorcraft. Colours shall be in accordance with BS381C (see Table 1).

1.3 Chapter 407 gives precautions against corrosion and details of materials and paint schemes to be used. Wherever possible, adhesive markings shall be used.

2 EMERGENCY CONTROLS

2.1 These requirements apply to the emergency controls defined in Chapter 105, para 11.3.

2.2 These emergency controls shall be coloured matt black with diagonal golden yellow stripes[1] 4.0mm to 6.5mm wide, separated by black bands 8.5mm to 12.5mm wide. Where possible, the wider stripes should be used, but at least two complete yellow stripes should appear on any emergency control. When electrical switches are emergency controls, an area of at least 25mm square around each dolly shall bear the appropriate emergency colour markings.

2.3 The emergency lighting switch shall be indicated by a self-luminous marker. The marker shall have the phosphor and exciter sealed in such a way that the external radiation is negligible. Unprotected radio-active paints shall not be used.[2]

2.4 FLYING CONTROL LOCKING LEVERS

2.4.1 All flying control locking levers shall be painted red.

2.5 ESSENTIAL LETTERING

2.5.1 Essential lettering to denote the purpose or mode of operation of controls shall be in small white letters on a black background, and clearly visible to the aircrew. When a number of controls are grouped together, the lettering for each control shall be located in a position acceptable to the pilot which, where practical, shall be below the control when the operation is vertical and to the rear when operation is lateral or fore-and-aft. For emergency controls, the lettering shall be on the control only when this can be done without impairing the clarity of the black and yellow stripes; in other cases, the lettering shall be adjacent to the control and there shall be no possibility of its being taken to refer to the wrong control.

3 CANOPY OR HATCH CONTROLS[2]

3.1 NORMAL

3.1.1 The opening controls inside will be marked as required.

3.2 EMERGENCY

3.2.1 The controls inside shall be marked as follows:

(i) Slanting stripes alternately in black and yellow, painted on the control or outlining it.

(ii) An orange/yellow arrow will indicate the operating direction.

(iii) The words 'CANOPY JETTISON' or other appropriate wording may be added inside.

4 EMERGENCY EXITS AND THEIR OPERATING CONTROLS

4.1 The words "EMERGENCY EXIT", plus any supplementary information (e.g. "DITCHING EXIT") as may be necessary, together with instructions in simple terms for operating (e.g. "PULL" or "TURN") with an arrow indicating the direction of operation shall be placed as near as possible to the control. The arrow shall be golden yellow and the letters shall be black on a golden yellow background. The letters of the title shall be not less than 25.4mm high and the letters of the instructions not less than 12.7mm high. On transport and maritime rotorcraft the contour of the emergency exits shall be indicated by slanting alternatively black and golden yellow stripes.[2]

4.2 Emergency exit controls shall be coloured in accordance with para 2.2. Each control shall incorporate, if possible, a self-luminous marker (see para 2.3) but if this is not possible, the marker shall be installed adjacent to the release lever. On transport rotorcraft, a self-luminous marker (see para 2.3) incorporating the words "EMERGENCY EXIT" shall be installed above the main entry door(s).[2]

4.3 The controls operating normal entrances/exits shall be clearly marked and operating instructions applied in the immediate vicinity. The markings are to be in golden yellow letters 25.4mm high.

5 LIFERAFT RELEASE AND FLOTATION CONTROLS

5.1 The words:

(i) "LIFERAFT RELEASE" for the control for liferaft inflation and release, or

(ii) "AIRCRAFT-FLOTATION" for the control for aircraft flotation gear,

together with instructions in simple terms for operating, (e.g. "PULL TO RELEASE LIFERAFT"), shall be placed as near as possible to the control. The letters shall be golden yellow on a black background. The letters of the title shall be not less than 25.4mm high and the letters of the operating instructions not less than 12.7mm high. If the control is in the form of a switch, the size of the letters may be reduced to a size appropriate to the location[2].

5.2 The operating handles or levers shall be coloured in accordance with para 2.2.[2]

6 AIRFRAME NOTICES

6.1 Airframe notices shall be restricted to:

(i) instructions to which crew or passengers may have a definite need to refer in the air,

(ii) the identification of removable panels or cover plates which would simplify or assist the maintenance procedure, where practicable,

(iii) emergency or warning notices, neglect of which is likely to cause damage to the rotorcraft or injury to personnel.[2]

Non-emergency airframe notices shall be positioned in such a manner that they do not detract from the emergency notice.

6.2 Emergency notices should be confined to operations relating to a real emergency. Where the instruction is adequately covered by periodical drill, the notice should be considered redundant. Emergency notices should be direct and concise and all preamble should be avoided.

7 COLOUR STANDARDS AT CREW STATIONS[3] (see also Chapter 115 - Night Vision Goggles*)

7.1 On all rotorcraft, a lustreless matt grey finish (except for the operational markings required by paras 2, 3, 4, 5, and 7) shall be provided for the whole of the structural interior at all crew stations (see Note 1). The structural interior is to include:

(i) main instrument panel,

(ii) consoles and pedestals,

(iii) other knobs and handles not adjacent to transparencies (canopies, windows, windscreens etc.),

(iv) instrument and switch mounting panels,

(v) floors, walls and ceilings (see Note 2),

(vi) control columns, rudder bar mechanism, (directional control mechanism and pedals)

(vii) desks, plotting table tops, writing surfaces, non upholstered seats and their associated structure.

Note 1: May be a lustreless matt black where low light levels and NVG compatibility are required.

2: A different colour scheme or finish may be an advantage at crew stations with restricted or no external transparencies. Where this is the case, Air Staff agreement should be sought.

*JAC Paper No 1131

3: Control and display surfaces may be grey or black.

7.2 A lustreless matt black finish shall be provided for the following:

(i) visible parts of instrument and equipment cases, instrument bezels, mounting flanges and instrument controls,

(ii) control panel surfaces and their mounted controls,

(iii) canopy and windscreen framing,

(iv) glare shields and horizontal surfaces above the top of the instrument panel. Where glare shields are not used, the black colour shall extend 13mm below the top of the instrument panel (see Note 1),

(v) surfaces which are at or above the lowest extent of the crew station transparency,

(vi) foot pedals,

(vii) control column grip and control wheels,

(viii) switch base plates,

(ix) communication and navigation controllers,

(x) autopilot controllers.

Note 1: Other surfaces reflecting directly into the forward windscreen and impairing the pilot's view shall be finished in matt black.

2: Every effort shall be made to minimise the reflectivity of any unavoidably unpainted metallic surfaces of switches or knobs.

7.3 The interior colour schemes in other aircrew stations shall be determined according to the characteristics of the various station areas/equipments with consideration being given to compartment size and lighting, climatic control and the physiological and psychological well-being of the aircrew.

7.4 The selection and application of materials for interior rotorcraft colours shall result in a smooth and uniform appearance with minimum specular reflectance. Colours shall be chosen to avoid harsh contrast between adjacent areas.

8 MAINTENANCE AREAS

8.1 A white gloss finish (except for the markings required by paras 3, 4, 5 and 6 above) shall be provided for the areas where maintenance takes place. A detailed schedule of the areas to be so painted shall be agreed with the Rotorcraft Project Director and should include the following areas:

(i) undercarriage bays, main and auxiliary,

(ii) equipment bays,

(iii) under the floors of cockpits and cabins,

(iv) baggage holds,

(v) bomb bays.

8.2 The finish required by para 8.1 shall not prejudice any camouflage or other protection applied to the rotorcraft nor shall it be used in areas where unwanted reflections could occur during flight operations.

8.3 Sealed areas, such as integral tanks, may be excluded from the requirement.

8.4 No more than two coats of white gloss finish shall be applied.

TABLE 1

COLOUR CODE

Colour	BS 381C Reference	See Para
Golden yellow	356	2.2,3.1 4.1
Red	538	2.4
Grey	632	6.1

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/23	3341
2	51/2	3230
3	10/58	3701
4	-	3687

CHAPTER 104

VIEW AND CLEAR VISION

1 INTRODUCTION

1.1 This Chapter states the basic operational requirements for the view to be provided for the pilot and other crew members and for the maintenance of clear vision.

2 GENERAL REQUIREMENTS FOR AIRCREW STATIONS[1]

2.1 External vision appropriate to rotorcraft class, type and operational mode, shall be provided at all crew stations.

2.2 The optical quality, curvature, and visual angles of incidence of the rotorcraft transparencies shall produce the least possible optical distortion and deviation and shall minimize reflections. Transparencies shall be compatible with the use of Passive Night Goggles at wavelengths ranging from 400 nm to 950 nm. (See Volume 1 Chapter 721 for minimum light transmission requirements).

2.3 Viewing areas defined as essential shall be kept clear of obstructions to vision, such as misting, rain, icing and insects (see para 5 below).

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3 BASIC REQUIREMENTS FOR PILOTS' STATIONS[1]

3.1 The windscreen directly in front of the pilots shall be as free of structure as possible. No windscreen structure in this area shall obstruct binocular vision.

3.2 For approach, including hovering and landing conditions, downward and forward vision shall be provided for the pilots to assure effective vision when using all landing aids.

3.3 In tandem rotorcraft, the front seat headrest and other cockpit structure shall be designed to permit vision appropriate to the rear crew member's task.

3.4 In side-by-side rotorcraft, the cockpit area inboard of the pilot and co-pilot shall not incorporate obstructions which interfere with the external vision appropriate to each crew member's task.

3.5 Controls, consoles, instrument panels, weapon aiming ancillary equipment and other structures shall be located so as not to critically restrict the vision of the pilot(s), neither directly nor as a consequence of reflections from the surfaces of the transparencies.

3.6 In rotorcraft utilizing in-flight refuelling, sufficient vision shall be provided for the pilot to see the tanker, refuelling signal lights, boom and probe (or drogue) when approaching and when in position for refuelling.

4 VISION REQUIREMENTS FOR PILOTS[1]

4.1 It is recommended that requirements and minimum angles of unimpaired external vision available to the pilot(s) when measured from the aircraft design eye position, as defined in Chapter 106, conform generally to those described in the vision requirements below. Increases in the pilot's field of view required for various rotorcraft types and/or mission modes and to satisfy specific visual requirements may extend the minimum angular values shown below. The reference plane from which the vision angles are specified shall be the pilot's horizontal vision plane. The zero reference in azimuth shall be that plane of pilot's vision parallel to the horizontal in cruising flight.

Note: It should be noted that a pilot does not view from the aircraft design eye position when wearing, Passive Night Goggles (PNG); typically, his head is forward from the aircraft design eye position, and the PNG may extend the viewing point further forward.

4.2 SIDE-BY-SIDE PILOT AND CO-PILOT (IF APPLICABLE) (FIG.1)

4.2.1 The following angles are designated with respect to the right side pilot. When a co-pilot is designated on the left side, then equivalent vision angles shall be provided for that side:

(i) At 0 degrees azimuth, 25 degrees down and 20 degrees up.

(ii) At 10 degrees left azimuth, 20 degrees down increasing linearly to 30 degrees down at 10 degrees right azimuth.

(iii) Between 10 degrees and 135 degrees right azimuth, 50 degrees down shall be provided.

(iv) From 0 degrees to 80 degrees right azimuth, upward vision shall increase linearly from 20 degrees to 40 degrees.

(v) From 80 degrees to 100 degrees right azimuth, 40 degrees up.

(vi) From 100 degrees to 135 degrees right azimuth, upward vision may decrease linearly from 40 degrees to 20 degrees.

(vii) From 10 degrees to 100 degrees left azimuth, every effort should be made to provide 20 degrees up and 20 degrees down.

(viii) There shall be no vertical obstructions to vision (pillars or posts) between 20 degrees right azimuth and 20 degrees left azimuth.

Note: The vision plot on Fig.1 generally represents the vertical vision angles between the azimuth location specified herein. (For diagrams in the alternative form of 'Aitoff's Equal Area Projection of a Sphere' see Mil Std 850).

5 BASIC OPERATIONAL REQUIREMENTS FOR CLEAR VISION

5.1 GENERAL

5.1.1 A clear vision for piloting, search, navigation and weapon aiming shall be available under all weather conditions. Sufficient area on transparent panels shall remain clear for these purposes, in external conditions of snow, rain and ice, and in internal conditions of mist and ice.

5.2 RAIN REMOVAL FROM WINDSCREENS

5.2.1 Adequate vision through the windscreen(s) shall be provided in all rain conditions up to heavy rain (see Volume 1 Leaflets 104/1 and 104/3).

5.2.2 On multi-engined rotorcraft, failure of one engine shall not impair the performance of the system.

5.2.3 The Contractor shall demonstrate, by ground tests before the rotorcraft flies and by flight tests at an early stage in the development of the rotorcraft, the effectiveness of the system installed.

5.3 PROTECTION OF WINDSCREENS FROM ICE (See Chapter 711)

5.4 PROTECTION OF WINDSCREENS FROM INSECTS, DIRT, ETC.

5.4.1 The pilot's view shall not be obstructed by insect debris, dust, dirt or salt from sea spray.

5.5 REQUIREMENTS FOR DESIGN

5.5.1 Requirements for the design of transparent panels are given in Chapter 715.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/53	3622

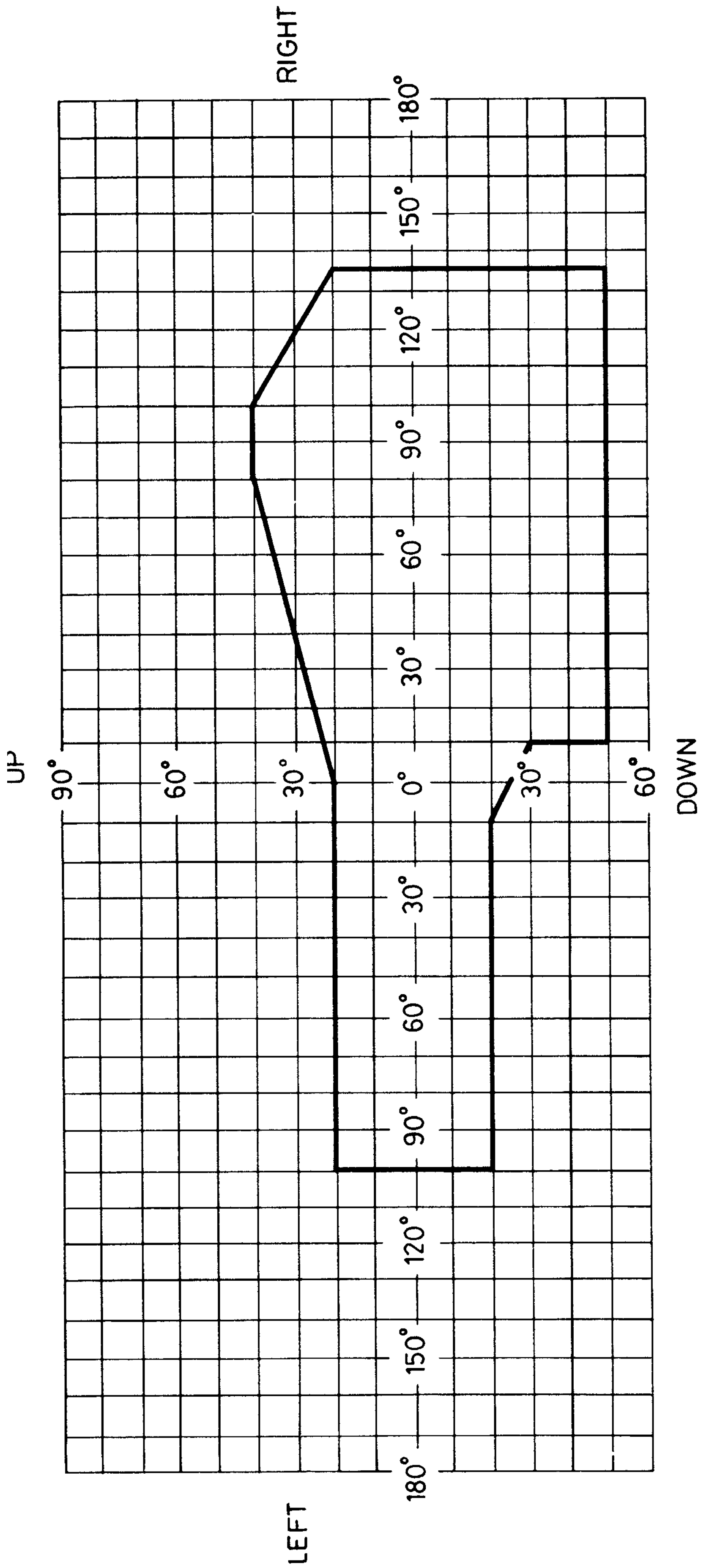


FIG.1 ROTORCRAFT VISION PLOT

Note: Fig.1 is based on the minimum visibility for any rotorcraft, therefore it may be necessary to expand the limits stated in para 4.2.1.

CHAPTER 105

CREW STATIONS - GENERAL REQUIREMENTS

1 INTRODUCTION

1.1 The requirements of this Chapter apply, unless otherwise specified, to all types of rotorcraft.

1.2 Each new station, while in the mock-up stage, shall be examined by the Rotorcraft Project Director and compared with the criteria contained in this chapter. Trials to determine the suitability of the crew stations layout shall be held with personnel harnessed in the seats and wearing the clothing and equipment detailed in the Aircrew Equipment Assembly Schedule (AEA) appropriate to the particular rotorcraft.

2 NOMENCLATURE[7]

2.1 The Nomenclature to be used for labels and legends on controls, panels and displays in aircrew stations shall be in accordance with Leaflet 105/1 and/or the following rules:

(i) The construction of nomenclature shall be based on common usage, syllable construction and phonetics.

(ii) Where possible, the entire wording shall be used.

(iii) Where the entire wording cannot be used, contraction which retains the phonetic sound structure or the original word shall be used.

(iv) Where space for a contraction is not available, approved abbreviations shall be used.

(v) Acronyms and symbols, other than those in common usage shall be avoided.

(vi) Words of four letters or less shall not be abbreviated unless common usage has rendered the word and its abbreviations completely synonymous in recognition and intelligibility.

3 LAYOUT

3.1 The layout of the crew stations shall be such that all members of the crew, when wearing the clothing and equipment detailed in the Aircrew Equipment Assembly Schedule appropriate to the particular rotorcraft, are capable of:

(i) carrying out their normal duties without discomfort, hindrance or delay under all normal flight conditions, and

(ii) carrying out without hindrance or delay all emergency procedures, including the operational tasks specified in Chapter 605, para 3.

3.2 The accommodation shall be such that fatigue is minimised, particularly on rotorcraft with a long endurance. Whenever possible, crew stations should be placed reasonably close together to assist mutual co-operation and morale.

4 FREEDOM OF MOVEMENT

4.1 Free, easy and safe movement between crew stations shall be possible on multi-seat rotorcraft. Foot and/or hand holds or equivalent aids shall be provided. Padding or guards shall be provided on any prominent structure or equipment which may cause injury or hamper the movement of the crew whilst at their stations or moving about the rotorcraft.

5 NOISE (see Chapter 108)

5.1 The noise level at crew stations shall be as low as possible, and in any case not exceed the level at which intercommunication will be satisfactory.

6 INTERCOMMUNICATION

6.1 Intercommunication shall be provided between all operational stations inside the rotorcraft, and also to a point(s) on the outside of the rotorcraft for use by ground personnel.

7 DESIGN OF SEATS AND HARNESSES

7.1 Crew seats, harnesses, and their attachments shall meet the crash landing and ditching requirements of Chapter 307 unless fully factored flight loads are greater.

7.2 The strength requirements for seats and harnesses of other occupants are given in Chapter 714.

7.3 Seats with provision for fitting the equipment detailed in the Aircrew Equipment Assembly Schedule (AEA) for the rotorcraft shall be provided at all crew stations.

7.4 Any seat which might not be occupied during flight shall be provided with a means of preventing the movement of articles attached to or associated with the seat and which normally remain in the rotorcraft. The stowage arrangements shall ensure the security of the articles under all conditions and manoeuvres throughout the flight envelope.

8 RESTRAINT AND PARACHUTE HARNESS FOR AIRCREW

8.1 A harness shall be fitted at each crew station to restrain the crew member during flight manoeuvres and during the crash landing and ditching conditions of Chapter 307. See Chapter 111 for the design requirements for restraint and parachute harness for aircrew.

8.2 For crew members whose duties require them to stand near an open door in flight, a body harness providing vertical head-up overhead suspension and adequate restraint for both pelvis and thorax, shall be provided. See Chapter 714 para 3.8 for strength requirements.

9 ARMOUR PROTECTION

AL/2 (See Chapter 113).

10 COCKPIT STRUCTURE

10.1 The construction and layout of the cockpit structure and equipment shall be such as to prevent serious injury to as far as is reasonably practicable to the crew whilst harnessed in their seats, or damage to survival equipment, during flight manoeuvres and under the crash landing conditions of Chapter 307.

10.2 In meeting the requirements of para 10.1, consideration shall be given to the following:

(i) The crew dimensions for 'small' and 'large' airmen given in Volume 1 Leaflet 105/3.

(ii) The seat and harness deflection under 90% of the ultimate loads derived from the crash cases.

(iii) Possible structural deformation at this load.

10.3 Within the envelope defined in para 10.2, there shall be no hazardous structure or equipment which might reasonably be foreseen as being a potential cause of injury to a crew member or damage to his survival equipment in the event of a crash or heavy landing.

11 CONTROLS

11.1 IDENTIFICATION OF CONTROLS

11.1.1 All controls and their function, where not immediately obvious, shall be clearly identified in unambiguous terms (see para 2) on or adjacent to each control. The method of marking shall be in accordance with Chapter 103.

11.2 ACTUATION

11.2.1 Controls and switches at all crew stations shall be so positioned that actuation upward, forward or clockwise will increase performance of the rotorcraft or the associated component.[1]

11.2.2 Actuation downward, aft or anti-clockwise shall decrease performance.[1]

11.2.3 The direction of actuation is related to the operator when at his operational station, and not to the rotorcraft.[1]

11.2.4 Three-position toggle switches shall be "OFF" when the dolly is at the central position, unless operationally or technically undesirable.

11.3 EMERGENCY CONTROLS

11.3.1 An emergency control is one which must be used instantly to avoid or correct a dangerous or disastrous situation, and as such it shall be easily accessible, shall not be complicated to operate and shall be easy to recognise.

11.3.2 It is important that the number of emergency controls shall be kept to the minimum. Unless otherwise agreed by the Rotorcraft Project Director, only those controls listed below shall be classified as emergency controls and be marked as such:[2]

(i) Emergency canopy or hatch jettison control.

(ii) Emergency external stores jettison switch.

(iii) Internally-carried armament stores jettison.

(iv) Landing gear down-lock override switch.

(v) Emergency fuel switches.

(vi) Engine emergency shutdown and fire extinguishing agent discharge switches.

(vii) Emergency override trim switch.

(viii) Emergency automatic flight control system disconnect switch.

(ix) Abandon aircraft switch.

(x) Hoist cable cutter control.

(xi) Sonar cable cutter.

(xii) Emergency external load release.

(xiii) Emergency/jettison door control.

(xiv) Emergency flotation gear control.

(xv) Liferaft release control.

(xvi) Emergency electrical supply.

(xvii) Emergency flare firing.

11.3.3 Controls for the operation of standby services (e.g., undercarriage - standby lowering) shall not be classified as emergency controls.

11.3.4 Emergency controls shall be marked in accordance with the requirements of Chapter 103.

11.4 PREVENTION OF INADVERTENT OPERATION

11.4.1 Locking wire or pins shall not be used to prevent inadvertent operation of controls or switches for any normal, standby or emergency services which may be required at any time during flight.

11.4.2 All emergency and standby controls and switches which are liable to accidental operation shall be protected by guards. For the emergency controls and switches defined in para 11.3, the guards shall be so designed that operation of the controls by a single movement of a gloved hand is possible; such guards should, where practicable, be of the "blinker" type. For standby controls and switches, a gate device is an acceptable form of guard.

11.5 CONTROL WHEELS, KNOBS AND SWITCHES

11.5.1 The design and installation of all hand operated controls and switches shall be such that there is no difficulty in obtaining an effective grip or in operating the control with cold hands.

11.5.2 Control knobs shall be of distinctive shape to assist both visual identification and actual identification with the gloved hand, and where applicable, shall be in accordance with DEF STAN 16-24.

11.5.3 Control wheels and knobs should be made of material of poor thermal conductivity, (e.g. plastic) where they will be touched or gripped. They shall be as large as possible consistent with their function and if necessary the periphery shall be corrugated in preference to being milled or knurled.

11.5.4 All hand operated controls and switches shall be so positioned that they may be readily operated with gloved hands. Press switches shall not be used unless they are essential to the operation of the service controlled (eg press-to-transmit).

11.6 TELL TALE DEVICES FOR EMERGENCY AND STANDBY SYSTEMS

11.6.1 A 'tell tail' device to indicate that an emergency or standby system has been operated shall only be fitted where the system is irreversible from the cockpit control or where the fact that the system has been operated is not immediately obvious.

11.6.2 Copper wire may be used as a tell-tale in accordance with AP101A-0001-1 Chapter 2, but only where no other method is possible because there is a chance that an incorrect quality of wire (for example steel locking wire) might be used as a replacement. To prevent pieces of broken wire becoming a loose article hazard the method of attaching the tell-tale shall follow the principles contained in AP101A-0001-1 Chapter 2.

12 WARNING CAUTIONARY AND ADVISORY SIGNALS[3]

12.1 GENERAL

12.1.1 All Crew Stations - Three distinct categories of signal (see para 12.1.3) shall be used to inform crew members of the conditions which exist relating to the operation of the rotorcraft and/or its equipment. Both audio and visual means may be used, as specified by the Rotorcraft Project Director for transmitting these signals.

12.1.2 Pilot's Station - The pilot shall be provided with a warning system in accordance with the requirements of Chapter 107 para 14.1.

12.1.3 Definitions - The three categories of signal are defined as follows:

(i) Warning Signal

(a) A signal indicating the existence of an imminent catastrophic condition requiring immediate action or a limitation to the flight envelope of the rotorcraft.

(b) A master warning signal may be used to indicate operation of any one of a number of warning signals.

(ii) Caution Signal

(a) A signal indicating the existence of a hazardous or impending hazardous condition requiring attention but not necessarily immediate action.

(b) A master caution signal may be used to indicate operation of any one of a number of caution signals.

(iii) Advisory Signal

A signal used to indicate aircraft configuration, a condition of performance, the operation of essential equipment, or to attract attention for routine purposes.

12.2 VISUAL SIGNALS

12.2.1 Warning and cautionary lights shall be installed within the pilot's 30° cone of vision (ie 15° around the visual axis). The axis of the cone of vision for the pilot is measured from the design eye position (as defined in Chapter 106, para 2.1.1), directly forward to the top of the flight instrument panel (or coaming). When space is limited or the required number of warning or cautionary lights is excessive, warning or cautionary lights may be grouped outside the pilot's 30° cone of vision. In these cases a master warning - or master caution light shall be installed in the pilot's 30° cone of vision. For other stations the axis of the cone of vision is measured from the design eye position to the centre of the instrument or control panel. The apex of the cone in every case is at the design eye position of the crew member concerned.

12.2.2 When a master warning or caution signal is used, the master caution must be capable of being cancelled, but without cancelling the appropriate signal(s) on the central warning panel. The grouping of indicators on a warning panel should follow ergonomic principles.

12.3 AUDITORY SIGNALS

12.3.1 Auditory signals can be of a verbal or non-verbal form (see Volume 1 Leaflet 105/2), the preference being dependent on the type of warning required. The number of non-verbal signals should be minimised. When auditory signals are used as warning signals they must operate in conjunction with a visual signalling device. Auditory signals should be clearly audible under all flight conditions, including where necessary, when helmets or ear defenders are not worn (see para 12.3.2 (i)). There shall be provision for overriding and recalling the signals. The signals, when activated, shall be presented until either:

(i) the causative condition is corrected, or

(ii) a signal of higher priority is present, or

(iii) the signal is silenced by the override switch (under this condition the system is still armed for all other functions).

12.3.2 Auditory signals used for warning purposes shall conform with the following requirements:

(i) Standard Warning System (see Chapter 107, para 13.2). The warning signal shall be in accordance with the requirements of Specification EL 1960.

12.4 ABANDON AIRCRAFT COMMAND SIGNAL

12.4.1 In multi-seat rotorcraft, a signal assembly, operable from the first pilot's station (see Chapter 107, Table 3, Item 1), shall be installed to provide each other crew member with a visual signal commanding him to abandon the rotorcraft. The signal at each crew station shall be an illuminated red warning sign, labelled ABANDON AIRCRAFT. Each warning light shall be installed in a prominent position and shall have no dimming facilities.

12.4.2 If a crew member has to move from one station to another in the course of his duties, a warning sign shall be provided at each of his stations.

12.4.3 The Rotorcraft Project Director will decide on the need for a warning sign at the co-pilot's station when his seat is side-by-side with the first pilot.

12.5 TEST FACILITY

12.5.1 A test facility shall be provided for testing the light sources in all visual signals; where appropriate this facility will also test sub-system circuit integrity, and auditory signals.

13 INDICATORS[3]

13.1 The requirements of this para apply only to indicators which denote a condition (e.g. magnetic indicators and lamps) and not a varying quantity.

13.2 IDENTIFICATION OF INDICATORS

13.2.1 The design of the legends used for indicators shall be in accordance with the numeral and letter form requirements of DEF STAN 66-26 (Pt 6). The legends used shall be a minimum of 3.17 mm in height. When not energized, the legends on visual signals shall not be readable. The functions of each indicator shall be clearly identified in unambiguous terms adjacent to, or on, each indicator. A list of legends for this purpose is given in Leaflet 105/1.

13.3 ILLUMINATED INDICATORS

13.3.1 All illuminated indicators, unless otherwise specified shall have dimming, mechanical or electrical under the control of the appropriate crew member so that at night they do not impair his vision. If automatic dimming is considered, a photosensitive device should be incorporated.

13.3.2 Indicators should be so positioned or shielded that reflections from external light sources (including sunlight) do not give the impression that they are energized. When energized, display contrast shall be sufficient to ensure readability under all conditions except for operation at low light levels, when dimming, partial masking, or extinguishing (after warning has been noted) may be permitted. Warning signals shall not be totally extinguished (but see master caution signal - para 12.2.2).

(i) Warning Lights. The legend shall be opaque on a red background.

(ii) Caution Lights. The legend may be opaque on a yellow/amber background or translucent yellow/amber on an opaque background.

(iii) Advisory Lights. Advisory lights may be either of the incorporated legend type or non-legend type. When a non-legend light is used, a readily readable label under day and night conditions shall be provided adjacent to the light. Legend advisory lights installed in flight compartments shall employ translucent legends on an opaque background.

Notes:

(1) Provision shall be made to ensure the illumination of warning and cautionary legend light signals in the case of any single light source failure.

(2) The indicators should not be dimmed to an extent that their attention getting properties are degraded. However, signals capable of dimming to extinction may be required for special roles (e.g. passive night vision goggles).

(3) Care should be taken to ensure that brightness balance is maintained between different signals.

13.4 COLOUR CODING OF ILLUMINATED INDICATORS

13.4.1 The colour of visual signals shall be in accordance with Publication CIE No 2.2 (TC-1.6) 1975. Chromaticity co-ordinates shall be within the limits as defined in Tables 1 and 2 at 60% and 100% of rated voltage for dimmable lights, otherwise the rated voltage shall be applied to the terminals.

(i) Warning Signal - Red

(ii) Cautionary Signal - Yellow or Amber

(iii) Advisory Signal - Green, White or Blue, these colours shall have the following meaning:

(a) Green: That a unit or component is in tolerance, or a condition is satisfactory, or that it is appropriate to proceed with a sequence of events already set in motion.

(b) White or Blue: To indicate a status or position or action, without implying safe or unsafe condition, for the location of components, action or test in progress (warming up etc). The condition shall be identified with a displayed legend or label.

Notes: (1) White, Blue and Green colours should be capable of adequate dimming in situations where there is a requirement for full night adaptation for aircrew.

(2) Non-legend lights of this type shall not be installed in flight compartments.

Recommended boundaries for light signals are given in Figs. 1 to 5.

13.4.2 RELEASE OF NUCLEAR WEAPONS
If indicator lamps are required on the control panel for the purpose of controlling the release of nuclear weapons, the lamps shall be coloured as follows:

(i) green to denote safe,

(ii) amber to denote ready for release (all control unit lights which are on immediately prior to live release will be coloured amber. For safe release or safe jettison, all release circuit lights will be amber, but certain weapon arming lights may be green), and

(iii) red to denote conditions not covered by amber or green, e.g., unsafe, dangerous, transition or emergency.

As a general principle, as few lights as possible shall be used.

13.5 DIMMER SCREENS
13.5.1 If a dimmer screen is required over an individual warning light, the iris type shall be used.

13.6 MAGNETIC INDICATORS
13.6.1 When used in a cautionary category, magnetic indicators shall show a change from black to black and white diagonal stripes to indicate the non-availability or failure of a service. In the black and white diagonal stripe presentation, there shall be at least two white stripes of not less than 0.76mm width.

14 INSTRUMENTS

14.1 When a colour coding system is used to denote the dangerous, cautionary and normal operating ranges of instruments, the markings shall be in the following colours:

Red	for dangerous,
Yellow	for cautionary,
Green	for normal.[4]

The colour shall be displayed at the periphery of the instrument with the needle pointing to, but not covering, the colour segments.

15 LIGHTING

15.1 GENERAL

15.1.1 All interior portions, of the rotorcraft to which the crew have normal access during flight shall be adequately lighted by windows by day and by electric light by night. Sunblinds shall be provided unless otherwise specified by the Rotorcraft Project Director.

15.1.2 The instrument and panel lighting shall be arranged to avoid reflections from the windscreen, other transparent panels or objects in the crew stations.

15.2 INSTRUMENT LIGHTING

15.2.1 All instruments shall be illuminated by integral lighting, tungsten-white in colour unless red is specified by the Rotorcraft Project Director.[5] Wherever possible, split circuitry shall be used for the instrument lighting power supplies to safeguard against complete loss of lighting through a single failure.

15.3 PANEL LIGHTING

15.3.1 With the exception of the main instrument panel and specified panels at other crew stations, all console panels and, as far as practicable, other equipment requiring illumination shall be lit by the plastic plate system (see Specification EL 1818) or electroluminescent units (see Specification EL 2033). With the plastic plate system the loss of any one light source shall not result in the marks or letters on any instrument or legend becoming invisible at full light intensity. Unless otherwise specified by the Rotorcraft Project Director, tungsten white lighting shall be used.[5]

15.4 FLOODLIGHTING

15.4.1 Floodlighting shall be installed at all crew stations to provide general and standby illumination of the instruments and console panels. The colour of this light shall be white, unless red is specified by the Rotorcraft Project Director.[5]

15.5 EMERGENCY LIGHTING

15.5.1 In those rotorcraft not fully safeguarded against electrical failure, an independent system of floodlighting shall be installed to illuminate the essential instruments at the pilot's station. It shall operate on a single self-contained circuit and derive its power from an independent emergency battery. The colour of this light shall be white, unless red is specified by the Rotorcraft Project Director.[5]

15.5.2 The circuit shall be controlled by a master switch at the pilot's station (see Chapter 103, para 2.3 and Chapter 107, Table 7, Item 4 for the marking and positioning of the switch).

15.6 ISOLATED INSTRUMENTS

15.6.1 Where it is not practicable for isolated instruments to be illuminated by integral lighting, they shall be illuminated by pillar lamps. Tungsten white lighting shall be used, unless red lighting is specified by the Rotorcraft Project Director.

15.7 WANDER LAMP

15.7.1 A wander lamp consisting of a tungsten-white floodlight, fitted with an alternative red filter, shall normally be fitted at each pilot's station. In the case of side-by-side seating, one lamp only need be fitted, provided that it is mounted between the pilots and is readily accessible to both. The lamp shall derive its power from the main electrical supply, and shall retract into a fixed stowage when not required for use.

15.8 LIGHTING CONTROLS

15.8.1 With the exception of the emergency floodlighting and anti- dazzle lights (see para 15.9), it shall be possible to control the intensity of illumination of all instruments and panels from full intensity to zero. Individual dimmer switches shall be provided within easy reach of each crew member to control the lighting at his station. Where more than one dimmer switch is required at a crew station, the dimmer switches shall be grouped together. Consideration shall be given at the design stage to separate dimming of instruments such as the main attitude indicator, which have their own special lighting requirements.

15.9 ANTI DAZZLE LIGHTS

15.9.1 If protection from the effects of nuclear explosions is called for in the Rotorcraft Specification, a high intensity white light shall be provided at the pilot's station[5] to give an illumination of:

(i) not less than 100 lumens/sq.ft. on the artificial horizon,

(ii) not less than 50 lumens/sq.ft. on the ASI and heading indicator, and

(iii) about 5 lumens/sq.ft. over the rest of the instrument panel.

15.9.2 Where there are two pilot's stations, the requirement of para 15.9.1 shall apply to both sets of instruments.

15.9.3 If more than one light has to be fitted to meet the requirements of para 15.9.1, the lights shall be wired so that failure of one light does not cause failure of other lights in the circuit.

15.9.4 The light(s) shall be controlled by a switch at the pilot's station[5] (see Chapter 107, Table 7, Item 2). With the switch in the "DIM" position, the illumination shall be about 1 per cent of the value given in para 15.9.1.

15.9.5 In cases where a separate light is required for another crew member, an on/off switch shall be used and so wired in circuit that it cannot operate the lights with the pilot's switch at "OFF". The crew member's switch, shall be protected against inadvertent operation.[5]

15.9.6 As the alignment of the light(s) is critical, the attachments to the rotorcraft shall be as rigid as possible to prevent accidental movement.

16 BLACKOUT CURTAIN

16.1 Blackout curtains at windows and screening of the lights shall be provided in the cabin, to ensure that on a clear dark night no light shall be visible beyond a distance of 100 metres.

16.2 Where it is necessary to separate the pilot's station from the rest of the crew compartment for night flying, etc. a curtain shall be used.

17 FIRST-AID KITS

17.1 Stowages shall be provided on all rotorcraft for first-aid kits to the scale specified for the particular type of rotorcraft.

17.2 Each stowage shall be designed to ensure that the kit is carried securely and without chafing, and shall be such that the kit may be inspected without removal from the stowage.

17.3 All kits shall have a quick release type of attachment.

17.4 When intended for use during flight, the kit shall be easily accessible to the crew and its location shall be marked with a red "Greek" cross on a white background.[6]

17.5 At least one kit shall be accessible from outside the rotorcraft in the event of a crash landing, and its location shall be marked in accordance with the requirements of AP119A-0601-0 Chapter 3.

17.6 In transport rotorcraft, a notice shall be displayed in the passenger cabin stating which member of the crew is in control of the kit for use in flight.

18 PERSONAL SURVIVAL PACKS

18.1 Stowage space shall be provided for the carriage of Personal Survival Packs of the type stated in the Rotorcraft Specification.

18.2 Personal Survival Packs are intended for use after a forced landing and their installation shall not prejudice the fitting of liferafts or life saving equipment intended for use after a ditching.

18.3 The packs shall be capable of being easily installed and shall be readily removable after a forced landing.

19 AIRCREW OVERNIGHT KIT STOWAGES

19.1 A dedicated stowage space for aircrew overnight kit shall be provided for each flight crew member. The stowage may be an area of the freight or luggage compartment, an internally or externally accessed airframe compartment, or externally fitted baggage pod.

20 HAND FIRE EXTINGUISHER

20.1 In all types of rotorcraft, hand operated fire extinguishers complying with BS 5423 shall be provided so that each member of the crew will have access to a hand extinguisher without leaving his station. One hand extinguisher shall be easily accessible from outside as well as inside the rotorcraft and its location shall be marked in accordance with AP119A-0601-0 Chapter 3. It is desirable that this hand extinguisher be stowed together with, or adjacent to, the fireman's axe and heat resisting gloves called for in para 21.3.

21 AXES AND HEAT RESISTING GLOVES

21.1 In multi-seat rotorcraft, fireman's axes and stowages shall be provided according to the number and position of crew stations. The number of stowages will be decided not later than at the Mock-up Conference.

21.2 Stowage shall be provided on all multi-seat rotorcraft for a pair of heat resisting gloves.

21.3 The stowage for one fireman's axe and the heat resisting gloves shall be accessible and readily identifiable from both inside and outside the rotorcraft and its location shall be marked in accordance with AP119A-0601-0 Chapter 3. It is preferable that these items shall be together in one stowage (see also para 20.1).

22 SANITATION

22.1 For passenger carrying rotorcraft, see Chapter 714.

22.2 In multi-seat rotorcraft where the endurance is greater than five hours and the crew can leave stations during flight, a chemical closet shall be installed. Handholds shall be provided near the closet.

22.3 In all other rotorcraft, where the endurance is greater than two-and-a-half hours, urine containers and funnels shall be installed at each crew station in easily accessible positions.

23 CUP HOLDERS

23.1 On rotorcraft with an endurance of four hours or more, cup holders shall be provided at each crew station. The cup holders shall be so designed/positioned that accidental spillage will not contaminate the rotorcraft services.

24 SIGNAL PISTOLS

24.1 A signal pistol, which can be loaded in and fired from a substantially horizontal stowage shall be fitted in all RAF rotorcraft and in naval rotorcraft employed in anti-submarine, search and rescue and short range transport duties.

24.2 The pistol shall be:

(i) accessible to the pilot in light rotorcraft, and

(ii) in any position convenient to the crew in other rotorcraft.

25 CROSS REFERENCES TO OTHER CHAPTERS

25.1 A number of requirements directly related to crew stations appear elsewhere in this publication and the most important of these are listed below (see also the Alphabetical Index).

Chapter	Para	Subject
101	3	Clear vision
	5	Air conditioning
102		Emergency escape
103	-	Operational colouring and markings
104	-	View and clear vision
106	-	Pilot's station - Layout
107	-	Pilot's cockpit - Controls and instruments
111	-	Restraint and parachute harness for aircrew

REFERENCES

Reference	ASCC Air Std.	STANAG
1	10/12	3217
2	10/23	3341
3	10/30	3370
4	-	3436
5	10/19	3224
6	51/2	-
7	10/47	3647

TABLE 1 xy CO-ORDINATES OF THE INTERSECTION POINTS OF THE BOUNDARY LINES

COLOUR		CO-ORDINATE POINTS				COLOUR		CO-ORDINATE POINTS			
		Q	R	S	T			K	L	M	N
Red	x	0.665	0.645	0.721	0.735	Yellow	x	0.560	0.546	0.612	0.618
	y	0.335	0.335	0.259	0.265		y	0.440	0.426	0.382	0.382
								A	B	C	D
						White	x	0.285	0.440	0.453	0.500
							y	0.332	0.432	0.440	0.440
		a	b	c	d			E	F	G	H
Green	x	0.305	0.321	0.228	0.028		x	0.525	0.565	0.542	0.500
	y	0.689	0.493	0.351	0.385		y	0.440	0.413	0.382	0.382
								I	J		
							x	0.440	0.285		
							y	0.382	0.264		
		j	k	l	m						
Blue	x	0.090	0.186	0.233	0.148						
	y	0.137	0.214	0.167	0.025						

TABLE 2 RECAPITULATION OF RECOMMENDED GENERAL BOUNDARIES FOR COLOURS OF LIGHT SIGNALS

Colour of Signal	Boundary	Para ref. In Publication CIE No 2.2(TC-1.6)1975	Equation	Figure	Line
Red	Purple	4.2.4.1.1	y = 0.980 - x	2	RS
	Yellow	4.2.4.1.2	y = 0.335	2	RQ
Green	Yellow	4.2.4.2.1	x = 0.360 - 0.080 y	3	ab
	White	4.2.4.2.2	x = 0.650y	3	bc
	Blue	4.2.4.2.3	y = 0.390 - 0.171x	3	cd
White	(To be used if the intermediate colour is to be perceived as white)				
	Red	4.2.4.3.1	y = 0.790 - 0.667x	4	EF
			x = 0.255 + 0.750y	4	FG
	Purple	4.2.4.3.2	y = 0.382	4	GI
	Blue	4.2.4.3.3	y = 0.047 + 0.762x	4	IJ
		4.2.4.3.4	x = 0.285	4	JA
	Green	4.2.4.3.5	y = 0.150 + 0.640x	4	AC
			y = 0.440	4	CE
Yellow	(To be used if the intermediate colour is to be perceived as yellow)				
	Red	4.2.4.4.1	y = 0.382	2	MN
	White	4.2.4.4.2	y = 0.790 - 0.667x	2	LM
	Green	4.2.4.4.3	y = x - 0.120	2	KL
Blue	(To be used if the fifth colour is to be perceived as blue)				
	Green	4.2.6.1.1	y = 0.065 + 0.805x	5	jk
	White	4.2.6.1.2	x = 0.400 - y	5	kl
	Violet	4.2.6.1.3	x = 0.133 + 0.600y	5	lm

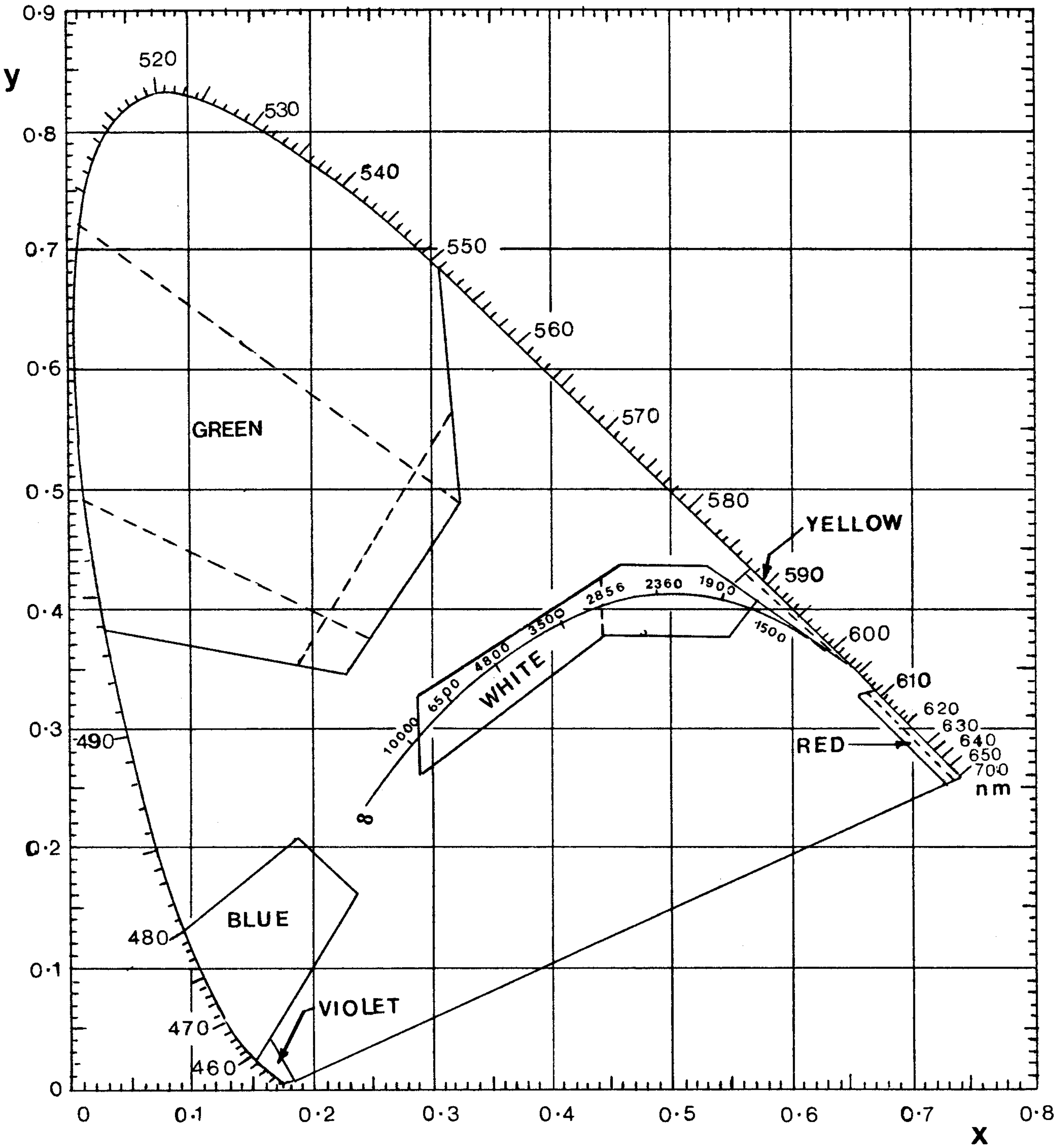


FIG.1 - RECOMMENDED BOUNDARIES FOR LIGHT SIGNALS

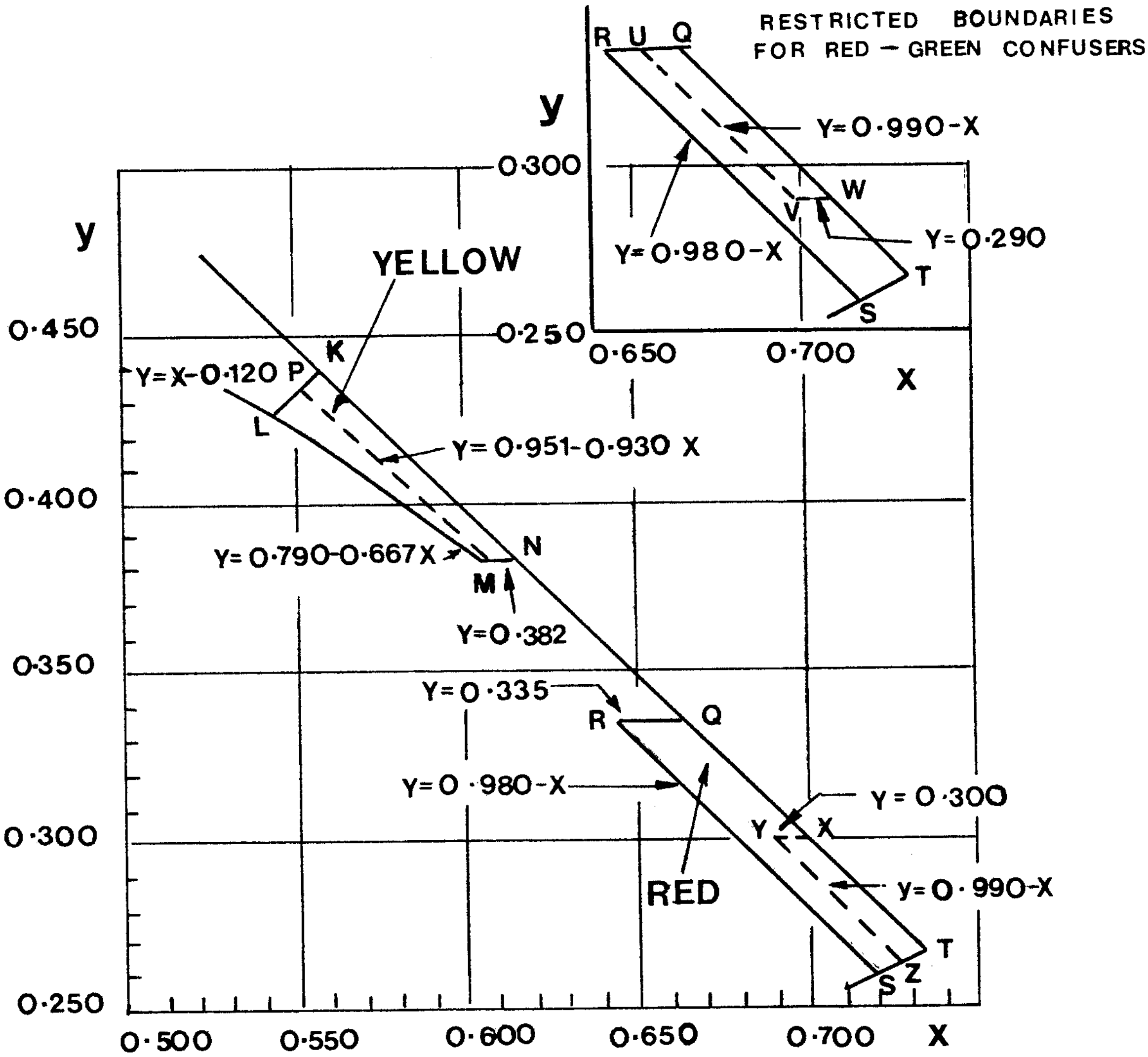


FIG.2 - RECOMMENDED BOUNDARIES FOR YELLOW AND RED LIGHT SIGNALS

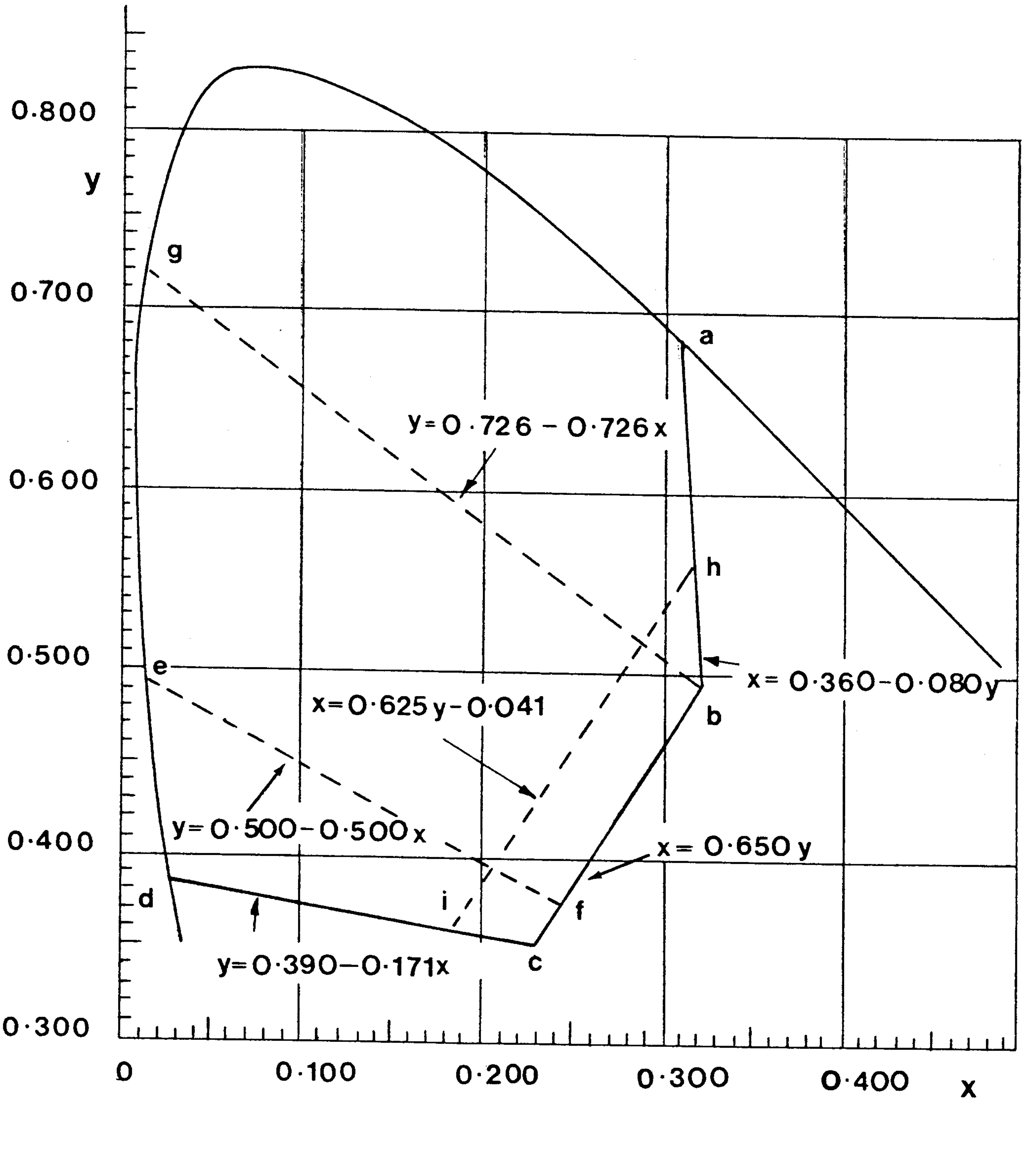


FIG.3 - RECOMMENDED BOUNDARIES FOR GREEN LIGHT SIGNALS

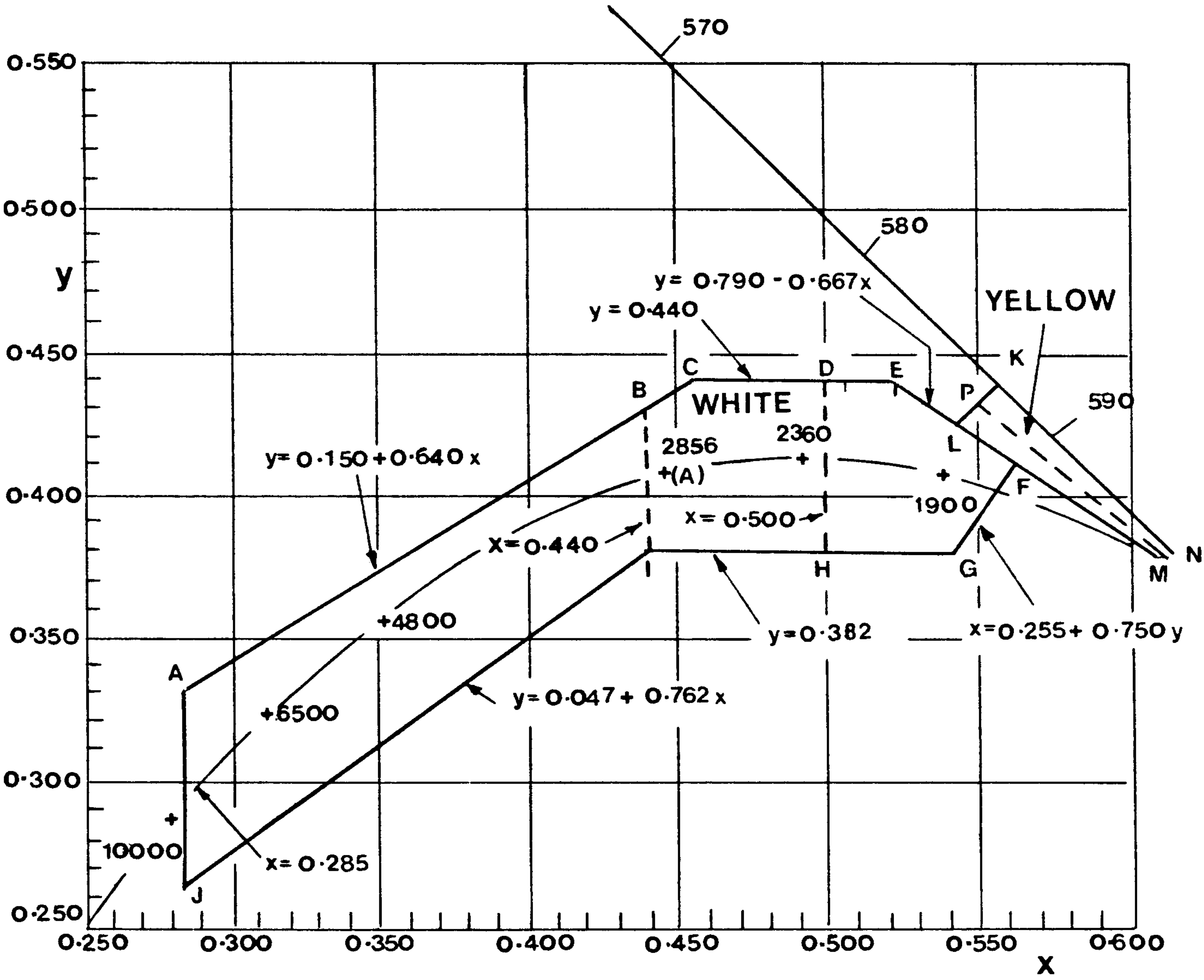


FIG.4 - RECOMMENDED BOUNDARIES FOR WHITE AND YELLOW LIGHT SIGNALS

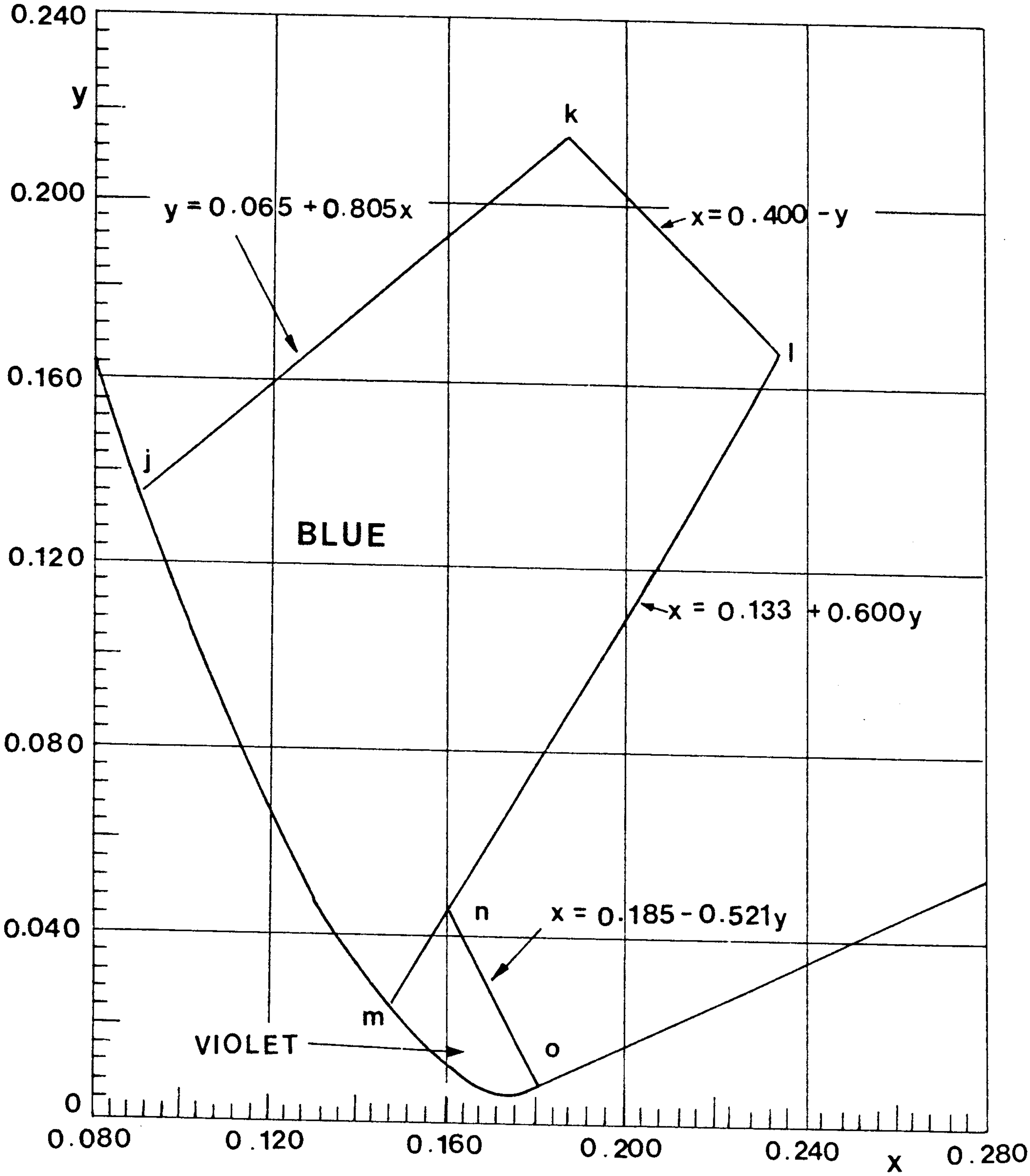


FIG.5 - RECOMMENDED BOUNDARIES FOR BLUE AND VIOLET LIGHT SIGNALS

LEAFLET 105/0

CREW STATIONS - GENERAL REQUIREMENTS

REFERENCE PAGE

<u>MOD(PE) Specifications/Publications</u>

EL1818	..	..	Transilluminated console panels
EL1960	..	..	Standard centralised warning system Mk.2
EL2033	..	..	Electroluminescent panels for aircraft consoles

<u>RAE Technical Notes</u>

Mech.Eng.69	..	..	Noise reduction in aircraft

<u>Defence Standards</u>

25-7	..	..	Horns electric
42-13	..	..	Fire extinguishers (automatic and hand types) and associated equipment for aerospace vehicles.
16-24	..	..	Shapes of airframe and engine control handles and knobs in fixed wing aircraft.

LEAFLET 105/1

CREW STATIONS - GENERAL REQUIREMENTS

NOMENCLATURE

Nomenclature and legends for use on controls, panels, and displays in aircrew stations:

AGREED NATO NOMENCLATURE[1]

NOMENCLATURE		LEGEND
Absolute	:	ABS
Accumulator	:	ACC
Acquire	:	ACQ
Aft	:	AFT
Afterburner	:	A/B
Agility	:	AGTY
Aileron	:	AIL
Air	:	AIR
Air conditioner	:	AIR COND
Aircraft	:	ACFT (A/C)
Air Data	:	AIRD
Airspeed Indicator	:	ASI
Air to Air Missile	:	AAM
Air to Air) Refuelling)	:	AAR A/R
Air to Ground Missile	:	AGM
Air to Surface Missile	:	ASM
Align,-ment	:	ALGN
Alternating current	:	AC
Alternator	:	ALTNR
Altitude	:	ALT
Amperes	:	AMP
Angle of attack	:	AOA

NOMENCLATURE		LEGEND
Antenna	:	ANT
Anti	:	ANTI
Approach	:	APP
Arm	:	ARM
Armament	:	ARMT
Arrester Hook	:	HOOK
Attack	:	ATTK
Attitude	:	ATT
Attitude and Heading) Reference System)	:	AHRS
Augment(ation)	:	AUG
Automatic	:	AUTO
Automatic Direction Finding	:	ADF
Automatic flight control system	:	AFCS
Automatic Pilot	:	AUTO-PLT
Auxiliary	:	AUX
Auxiliary power unit	:	APU
Azimuth	:	AZ
Back Beam Mode	:	BBM
Back Up	:	B/U
Bail out	:	BAIL OUT
Battery	:	BATT
Beacon	:	BCN
Bearing	:	BRG
Bomb Release Safety Lock	:	BRSL
Boost	:	BOOST
Brake	:	BRAKE
Breaker (circuit-)	:	BRK (BRKR)

NOMENCLATURE		LEGEND
Bright	:	BRT
Built in Test	:	BIT
Built in Test Equipment	:	BITE
Bus (electrical)	:	BUS
Cabin/cockpit Pressure	:	CPR
Calibration	:	CAL
Canopy	:	CANOPY
Caution	:	CAUTION
Centre	:	CTR
Centre Line	:	CL (C/L)
Centre of Gravity	:	CG
Central Air Data Computer	:	CADC
Central Warning Panel	:	CWP
Central Warning System	:	CWS
Chaff	:	CHAFF
Channel	:	CHAN
Climb	:	CLIMB
Cockpit Voice Recorder	:	CVR
Command	:	COMD
Command and Stability Augmentation System	:	CSAS
Communications	:	COM/COM
Communications Control Systems	:	CCS
Communication, Navigation, Identification	:	CNI
Compass	:	COMP
Compressor	:	COMPR
Computer	:	CMPTR

NOMENCLATURE		LEGEND
Connect	:	CON
Consoles	:	CONS
Constant Speed Drive	:	CSD
Continuous Wave	:	CONTW
Control	:	CONTR
Control Display Unit	:	CDU
Collision	:	COLL
Cooling	:	COOL
Course	:	CRS
Cylinder	:	CYL
Damper	:	DAMP
Decrease	:	DECR
Defroster	:	DEFROST
De-ice	:	DE-ICE
Detector	:	DETR
Depression	:	DEPR
Destination	:	DEST
Direct current	:	DC
Disconnect	:	DCON
Disengage	:	DISENG
Display	:	DISP
Distance	:	DIST
Distance Measuring Equipment	:	DME
Door	:	DOOR
Doppler	:	DPLR
Down	:	DN
Drift	:	DRFT

NOMENCLATURE		LEGEND
Eject	:	EJECT
Elevate (or) (ation)	:	EL (ELEV)
Emergency	:	EMERG
Emergency Power Unit	:	EPU
Engage	:	ENGAGE
Engine	:	ENG
Enter	:	ENT (ENTR)
Exhaust	:	EXH
Exhaust Gas Temperature	:	EGT
External	:	EXT
Failure	:	FAIL
Feet	:	FEET
Feet per Minute	:	FPM
Feather	:	FEATHER
Fire	:	FIRE
Flaps	:	FLAPS
Flight	:	FLT or FLGT (FLT)
Flight Director	:	FD
Flow	:	FLOW
Formation	:	FORM
Forward	:	FWD
Forward Looking Infra Red	:	FLIR
Forward Looking Radar	:	FLR
Frequency	:	FREQ
Frequency Modulation	:	FM
Fuel	:	FUEL
Generator	:	GEN
Glide	:	GLIDE

NOMENCLATURE		LEGEND
Glide Slope	:	G/S
Governor	:	GOV
Ground Mapping Radar	:	GMR
Ground Speed	:	GS
Gyroscope	:	GYRO
Heading	:	HDG
Head Up Display	:	HUD
Heater	:	HTR
Heating	:	HTG
Height	:	HT
Helium	:	HELIUM
High	:	HI
High Frequency	:	HF
Hoist	:	HOIST
Hold	:	HOLD
Home-on-Jam	:	HOJ
Homing/Home	:	HOME
Hydraulic	:	HYD
Icing	:	ICE
Identification of Friend or Foe	:	IFF
Inboard	:	INBD
Increase	:	INC
Indicated airspeed	:	IAS
Indicator	:	IND
Inertial Navigation System	:	INS
Infra Red	:	IR
Inlet guide vane	:	IGV

NOMENCLATURE		LEGEND
Instruments	:	INST
Instrument Landing System	:	ILS
Internal (Interior)	:	INTR
Inverter	:	INV
Jettison	:	JETT
Knots	:	KTS or KNOTS
Knots Calibrated Airspeed	:	KCAS
Landing Gear	:	LDG GEAR
Latitude	:	LAT
Left	:	L
Level	:	LEVEL
Light,s,ing	:	LTG
Limit	:	LIMIT
Localiser	:	LOC
Lock	:	LOCK
Longitude	:	LONG
Loran	:	LORAN
Low	:	LOW
Low altitude bombing system	:	LABS
Lower	:	LWR
Lower Side Band	:	LSB
Low Frequency	:	LF
Low Light Television	:	LLTV
Mach Number	:	MACH
Magnetic	:	MAG
Manual	:	MAN
Marker	:	MKR

NOMENCLATURE		LEGEND
Master	:	MASTER
Master Armament Safety Switch	:	MASS
Maximum	:	MAX
Medium	:	MED
Memory	:	MEM
Microphone	:	MIC
Military	:	MIL
Millibar	:	MB
Miniature Detonation Cord	:	MDC
Minimum	:	MIN
Minute	:	MIN
Miscellaneous	:	MISC
Missile	:	MSL
Mode	:	MODE
Monitor	:	MON
Moving	:	MVG
Navigation/Navigate/Navigator	:	NAV
Navigators Hand Controller	:	NHC
Negative	:	NEG
Normal	:	NORM
Nose	:	NOSE
Nozzle	:	NOZZLE
Nuclear, Bacteriological, Chemical	:	NBC
Off	:	OFF
Offset	:	OFS
Oil	:	OIL
On	:	ON

NOMENCLATURE		LEGEND
On-Top Fixing	:	OTF
Open	:	OPEN
Operate	:	OPR
Out	:	OUT
Outboard	:	OUTBD
Overheated	:	HOT
Override	:	ORIDE
Oxygen	:	OXY
Pedestal	:	PED
Personal Equipment Connector	:	PEC
Pilot	:	PLT
Pilot's Display Recorder	:	PDR
Pilot's Hand Controller	:	PHC
Pitch	:	PITCH
Positive	:	POS
Pounds per hour	:	PPH
Power	:	PWR
Present Position	:	PP
Pre	:	PRE
Position and Homing Indicator	:	PHI
Pressure	:	PRESS
Primary	:	PRI
Propeller	:	PROP
Pump	:	PUMP
Quantity	:	QTY
Radar	:	RADAR
Radar Warning Receiver	:	RWR

NOMENCLATURE		LEGEND
Radar or Radio Altitude	:	RAD ALT
Radome	:	RADM
Range	:	RNG
Rapid	:	RPD
Rapid Data Entry	:	RDE
Reconnaissance	:	RECCE
Receiver	:	RCVR
Rectifier	:	RECT
Reference	:	REF
Regulator	:	REG
Reheat	:	RHT
Release	:	REL
Retract	:	RETR
Repeat	:	RPT
Repeater	:	RPTR
Reverse	:	REVR
Revisionary	:	REVY
Revolutions per minute	:	RPM
Right	:	R
Rockets	:	RKTS
Roll	:	ROLL
Rotor	:	ROTOR
Route	:	RTE
Secondary	:	SEC
Secondary altitude and heading reference system	:	SAHR
Secondary Surveillance Radar	:	SSR
Section	:	SECT

NOMENCLATURE		LEGEND
Select	:	SEL
Sensitivity Time Control	:	STC
Shut	:	SHUT
Sideways Looking Airborne Radar	:	SLAR
Single Sideband	:	SSB
Skid	:	SKID
Special Weapon	:	SPLWPN
Speed	:	SPD
Squelch	:	SQ
Stability	:	STAB
Stability Augmentation System	:	SAS
Standby	:	STBY
Static	:	STAT
Store	:	STR
Surface	:	SURF
Synchronize	:	SYNC
System	:	SYS
Tacan	:	TACAN
Tail	:	TAIL
Take-off	:	T/O
Tank	:	TANK
Target	:	TGT
Temperature	:	TEMP
Terrain avoidance	:	TER AVD
Terrain clearance	:	TER CLR
Terrain Following	:	TER FLW
Terrain following Radar	:	TFR

NOMENCLATURE		LEGEND
Test	:	TEST
Time over target	:	TOT
Time to Go	:	TTG (T/G)
Thrust	:	THRUST
Track	:	TRK
Transfer	:	XFER
Transmission	:	XMSN
Transmit	:	XMIT
Transmit/Receive	:	T/R
Transmitter	:	XMTR
Transponder	:	XPDR
Trim	:	TRIM
True air speed	:	TAS
Turbine Blade Temperature	:	TBT
Turbine Bearing Temperature	:	TBT
Turbine inlet temperature	:	TIT
Ultra High Frequency	:	UHF
Unit	:	UNIT
Undercarriage	:	U/C
Universal Transverse Mercator	:	UTM
Unlock	:	UNLOCK
Upper	:	UPR
Upper Side Band	:	USB
Vacuum	:	VAC
Variation	:	VAR
Velocity	:	VEL
Ventilation	:	VENT

NOMENCLATURE		LEGEND
Vertical/Short Take Off Landing	:	VSTOL
Vertical Velocity	:	VVEL (VV)
Very High Frequency	:	VHF
Vibration	:	VIB
Volume	:	VOLUM (VOL)
Volts	:	V
Warning	:	WARN
Water	:	WATER
Waypoint	:	WPT
Weapon	:	WPN
Wheels	:	WHEELS
Winch	:	WINCH
Windshield/ Windscreen	:	W/S
Wingspan	:	WGSP
Yaw	:	YAW

NOTE: Legend in brackets is the preferred UK abbreviation and is not Agreed NATO Nomenclature.

ADDITIONAL UK NOMENCLATURE

NOMENCLATURE		LEGEND
Aircraft	:	A/C
Accident Data Recorder	:	ADR
Breaker (circuit)	:	BRKR
Centre Line	:	C/L
Computed	:	COMP
Directional Gyro	:	DG
Discrete	:	DIS/DSCRT
Elevat(e) (or) (ion)	:	ELEV

NOMENCLATURE		LEGEND
Enter	:	ENTR
End of Tape	:	EOT
Fuel Flow	:	FF
Free Fall	:	FREE
Fuel Remaining on Tank	:	FROT
Acceleration	:	G
Gas Turbine Starter	:	GTS
Head Down Display	:	HDD
High Pressure	:	HP
Invalid	:	INV
Input	:	I/P
Jet Pipe Temperature	:	JPT
Low Pressure	:	LP
Multifunction Display	:	MFD
Multipurpose Display	:	MPD
Outside Air Temperature	:	OAT
Output	:	O/P
Pilots Display Unit	:	PDU
Processor	:	PROC
Press to Transmit	:	PTT
Recording	:	REC
Retarded	:	RTD
Slats	:	SLATS
Starboard	:	STBD
Symbology	:	SYMB
Turbine Gas Temperature	:	TGT
Up Front Control Panel	:	UFCP
Video Cassette Recorder	:	VCR

NOMENCLATURE		LEGEND
Video Display Unit	:	VDU
Volume	:	VOL
Vertical Velocity	:	VV

REFERENCES

Reference	ASCC Air Std	STANAG
1	10/47	3647

LEAFLET 105/3

CREW STATIONS - GENERAL REQUIREMENTS

AIRCREW ANTHROPOMETRY

1 INTRODUCTION

1.1 This leaflet contains dimensions and weights to be used when designing aeroplanes and cockpit installations. The information is taken from Ref 1 which contains a statistical summary of 62 direct and 26 derived body measurements of 2,000 RAF aircrew (see Chapter 105 para 1.2).

2 DIMENSIONS

2.1 A pictorial index of the measurements in Ref 1 is reproduced in Figs. 1 to 4 together with the 3rd and 99th percentile values. Information on the inter relationship between specific measurements is given in Ref 2.

2.2 The measurements are based upon nude body dimensions so allowances must be made for clothing, harness restraint systems and seat geometry.

2.3 The four most critical body dimensions for cockpit workspace govern aircrew selection. These limitations (1982) and their relation to the 3rd and 99th percentile measurements are:

	Minimum	Maximum
Sitting Height	865mm (11 mm < 3%ile)	1010mm (3 mm > 99%ile)
Buttock-Knee Length	560mm (2 mm > 3%ile)	660mm (12 mm < 99%ile)
Buttock-Heel Length	1000mm (2 mm > 3%ile)	1200mm (1 mm < 99%ile)
Functional Reach	740mm (4 mm > 3%ile)	900mm (11 mm < 99%ile)

Note: The small differences between the 3rd and 99th percentile and the critical measurements quoted give some flexibility in selection and growth of aircrew after selection.

2.4 Limited information exists on specific anthropometric dimensions for female aircrew although general anthropometric information on female military personnel is contained in Ref. 3. Specialist advice must be sought for the interpretation of Ref. 3 to relate to specific aircrew issues.

3 WEIGHTS

3.1 Weights for the 3rd, 50th and 99th percentile range of aircrew are given below:

Percentile Values

%	Kg
3	59.46
50	74.46
99	96.50

3.2 These values are for the nude body so allowances must be made for clothing and aircrew equipment. Both the minimum and maximum aircrew equipment assembly weights must be considered. Typically these are given by the summer/land clothing assembly and the winter/sea assembly respectively. Specialist advice should be sought in the definition of appropriate clothing assemblies and additional aircrew equipment such as maps, NBC protection, personal weapons, body armour, survival equipment and Night Vision Goggles (NVGs).

4. EFFECTS OF CLOTHING AND SEAT GEOMETRY ON NUDE DIMENSIONS

4.1 EYE POSITION

4.1.1 The relationship between the seat reference point and the eyeball position of subjects strapped into aeroplane seats is complex. Detailed information has been published (Ref 4).

4.2 SHOULDER BREADTH

4.2.1 The clothing assemblies will add 10-20 mm to the nude Bideltoid breadth (min and max respectively).

4.3 BUTTOCK-KNEE LENGTH

4.3.1 The clothing assemblies will add 10-20 mm to the nude Buttock-knee length (min and max respectively).

5 POSITION OF CENTRE OF GRAVITY (CG)

5.1 Where, for design purposes, the position of the CG of the complete body is required, it may be assumed to be at elbow height sitting (Fig 1 dimension “0”) and 100 mm (small) to 150 mm (large) forward of the buttock/shoulder tangent line.

REFERENCES

No.	Title, etc.
1	RAF/IAM Report 531/PAE Technical Report 73083/FPRC Report 1327/HMSO R&M 3372 - An Anthropometric Survey of 2,000 Royal Air Force Aircrew 1970/71.
2	RAE TR 81017 Scatter Diagrams based on the Anthropometric Survey of 2,000 Royal Air Force Aircrew. (1970/71) - Simpson R E and Hartley E V.
3	DRA/AS/MMI/CR95076/1 - The 1995 Anthropometric Survey of UK Military Females - Judy E. Aplin, Huda Nammari, April 1995.
4	The Relationship between the Seat Reference Point and the Eyeball Position of Subjects Strapped into Aircraft Type Seats. D. G. Beeton; September 1975. Published as University of London MSc (Ergonomics) thesis - to be published during 1983 as an RAF IAM Report.

	PERCENTILE 3rd	PERCENTILE 99th
A Bideltoid breadth	427	514
B Biacromial breadth	370	452
C Hip breadth, sitting	332	415
D Stool height	376	471
E Thigh clearance height	137	191
F Acromial height, sitting	558	681

	PERCENTILE 3rd	PERCENTILE 99th
G Shoulder height, sitting	614	727
H Sitting eye height	765	896
J Sitting height	876	1007
K Vertical functional reach, sitting	1281	1515
L Knee Height, sitting	514	623

	PERCENTILE 3rd	PERCENTILE 99th
M Functional reach	736	889
N Cervicale height, sitting	627	745
O Elbow rest height, sitting	200	306
P Stomach depth	203	306
Q Buttock - knee length	558	672
R Buttock - heel length	998	1211

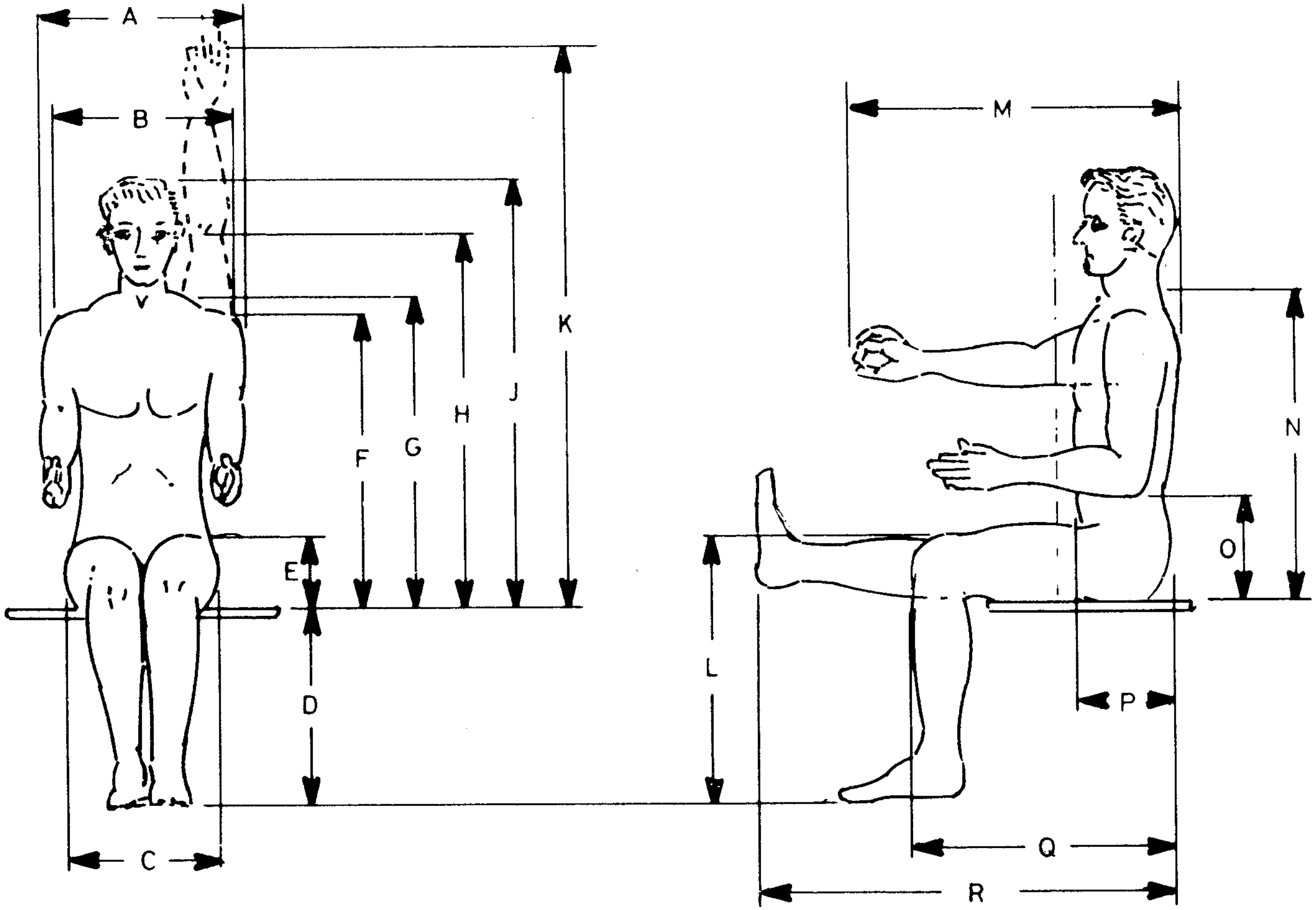


FIG.1 RAF AIRCREW 3rd AND 99th PERCENTILE VALUES (mm)

	PERCENTILE 3rd	PERCENTILE 99th
A Neck circumference	351	421
B Vertical trunk circumference (mean)	1504	1774
C Chest circumference	870	1111
D Waist circumference	720	1011
E Buttock circumference	895	1108
F Wrist circumference	157	197
G Thigh circumference	496	659
H Calf circumference	327	416

	PERCENTILE 3rd	PERCENTILE 99th
I Ankle circumference	203	255
J Eye height, standing	1549	1808
K Axilla height	1238	1478
L Axilla - wrist length	424	544
M Crotch height	775	960
N Waist to waist over shoulder	817	982
O Crotch length	654	858
P Axilla - cervicale length	146	216
Q Axilla - fingertip length	607	749

	PERCENTILE 3rd	PERCENTILE 99th
R Fingertip height	606	749
S Cervicale - crotch length	612	727
T Shoulder height, standing	1396	1646
U Cervicale - vertex length	231	293
V Cervicale - waist length	355	443
W Waist height	1023	1227
X Cervicale height	1410	1661
Y Stature	1661	1924

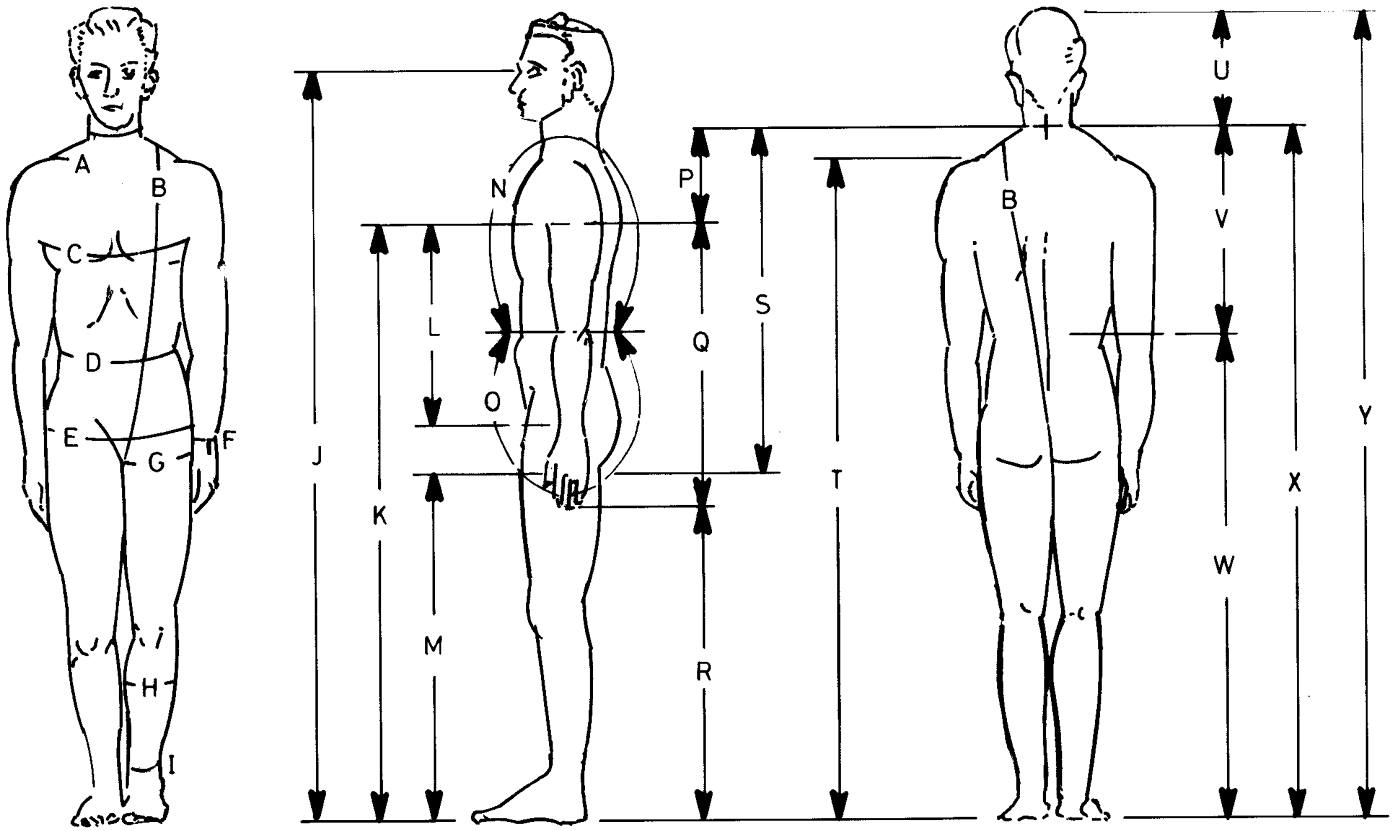


FIG.2 RAF AIRCREW 3rd AND 99th PERCENTILE VALUES (mm)

		Percentile 3rd	Percentile 99th
A	Span	1692	2014
B	Inter-elbow span	909	1088
C	Elbow-fingertip length	442	531
D	Elbow-wrist length	261	323
E	Hand length	173	216
F	Elbow-functional reach	384	464

		Percentile 3rd	Percentile 99th
G	Elbow, fully bent, circumference	310	384
H	Knee, fully bent, circumference	406	495
J	Foot breadth	87	106
K	Ball of foot circumference	228	278
L	Instep-sole circumference	228	273
M	Foot length	244	297
N	Heel-instep circumference	300	359

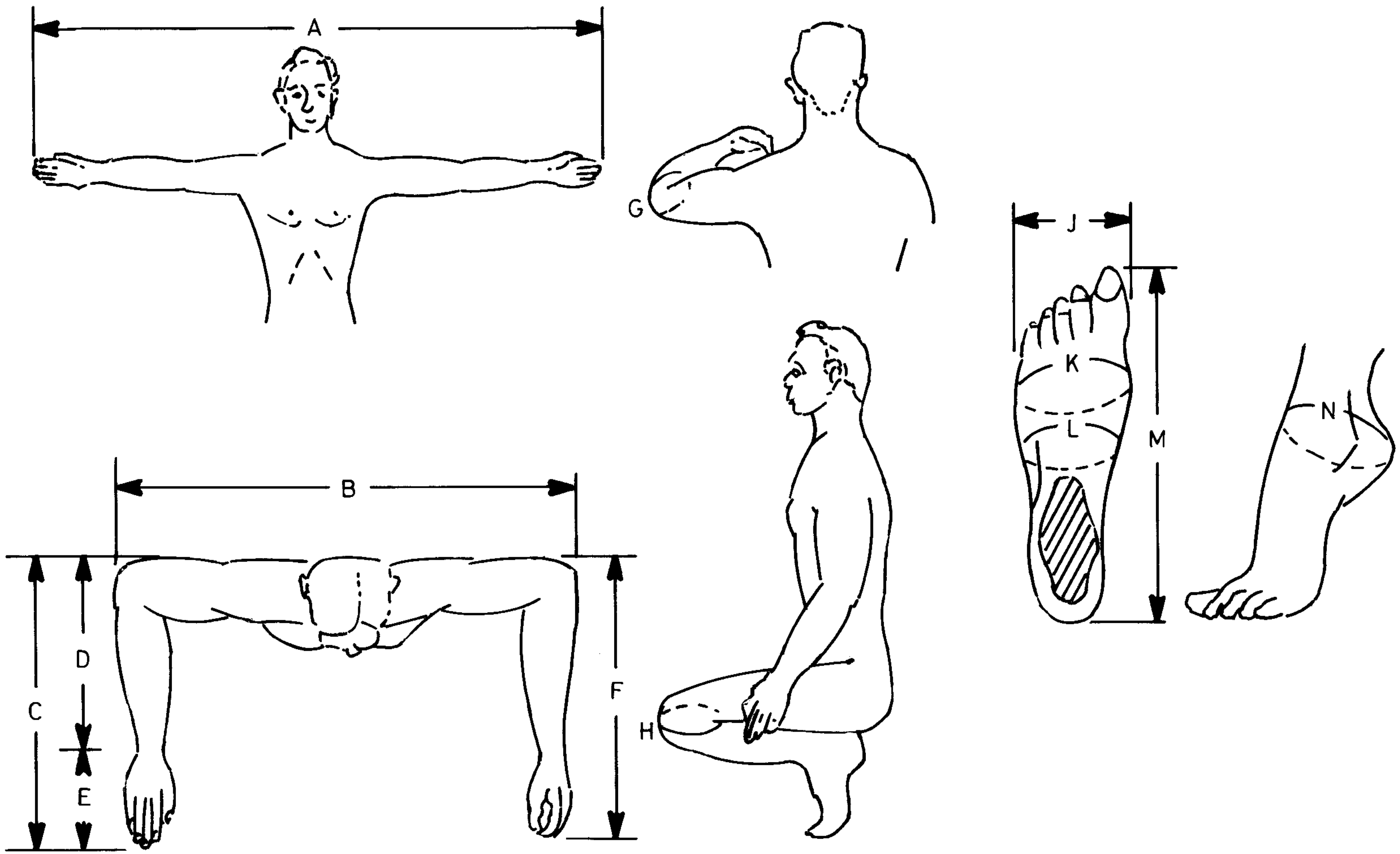


FIG.3 RAF AIRCREW 3rd AND 99th PERCENTILE VALUES (mm)

		PERCENTILE	
		3rd	99th
A	Head breadth	147	171
B	Bitragion - coronal arc	330	385
C	Head circumference	551	608
D	Bitragion diameter	129	152
E	Menton to vertex	210	252
F	Tragion to vertex	118	145
G	Tragion to menton, vertical	81	120
H	Nasion to vertex	88	128
J	Tragion to Nasion, vertical	6	45

		PERCENTILE	
		3rd	99th
K	Pupil to vertex	97	130
L	Tragion to pupil, vertical	3	36
M	Nasion to menton, vertical	110	140
N	Head length	186	213
O	Tragion to brow ridge, horizontal	85	112
P	Tragion to back of head	88	118
Q	Tragion to menton, horizontal	81	117
R	Menton to back of head	179	224
S	Maximum head diagonal from menton	247	280

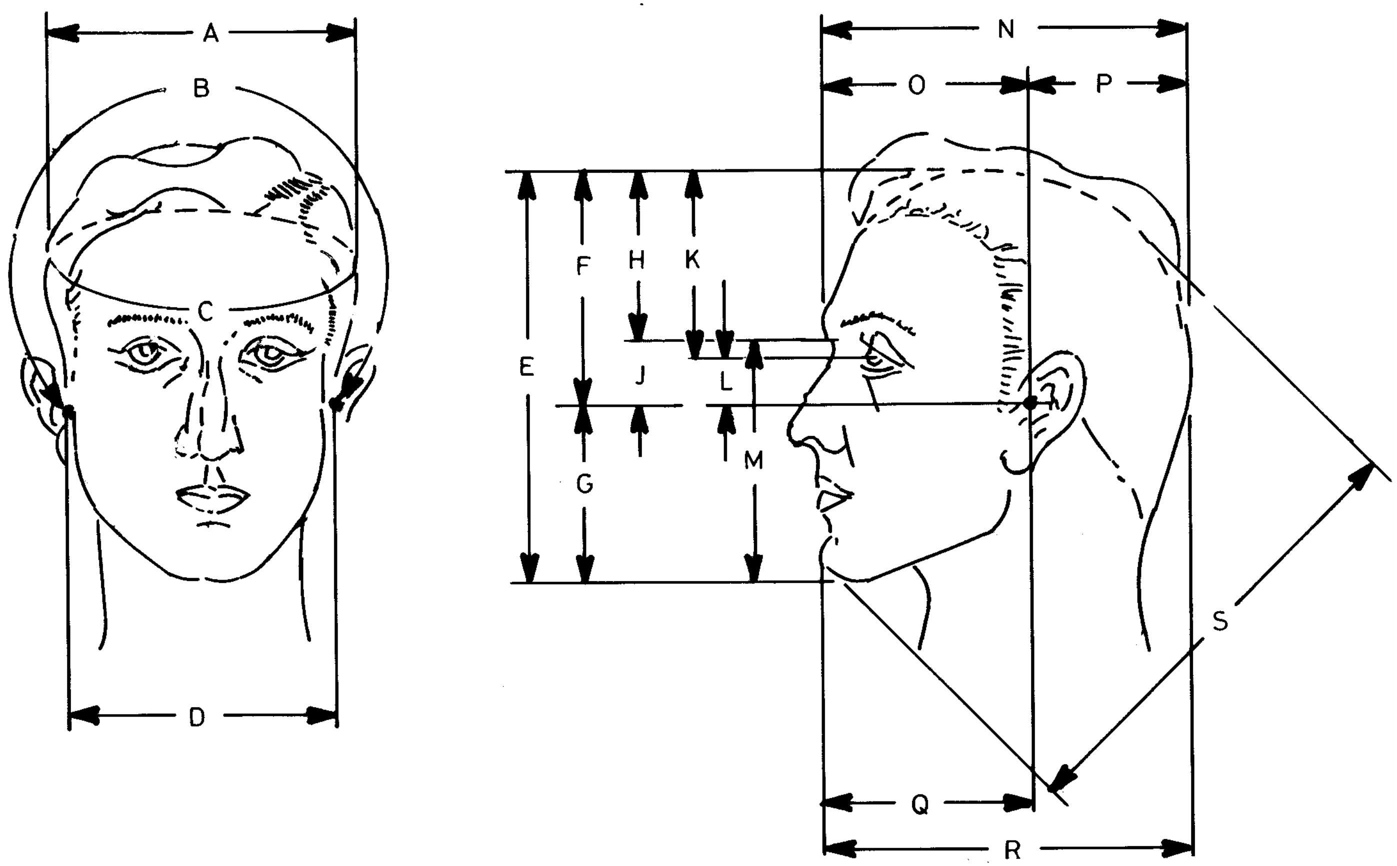


FIG.4 RAF AIRCREW 3rd AND 99th PERCENTILE VALUES (mm)

LEAFLET 105/4

CREW STATIONS - GENERAL REQUIREMENTS

GUIDELINES FOR THE DESIGN OF CREWSTATION LIGHTING AND DISPLAYS

1 INTRODUCTION

1.1 Crewstation lighting and displays (CL&D) should be designed such that they:

(i) Permit satisfactory transfer of visual data from the crewstation instruments and displays under all relevant lighting conditions.

(ii) Permit satisfactory view of crewstation controls.

(iii) Permit satisfactory transfer of visual data from the outside world (where the outside world constitutes all vision external of the rotorcraft crewstations).

(iv) Achieve the above without increasing the probability of detection by the enemy.

1.2 This leaflet gives general guidelines for the legibility of displays and controls under a wide range of ambient illumination levels and partially implements STANAG 3224 Annex B[1].

1.3 Three methods for specifying display legibility are discussed. These are the Mathematical Modelling technique, that specified by ARINC 725 and the 'conventional' method. The Mathematical Modelling technique is the preferred method whilst the ARINC 725 method may only be used where the Mathematical Modelling technique cannot be used (para. 2.2). The 'conventional' method, although detailed in paras 3 to 5, is not recommended and should only be used where expressly permitted by the Rotorcraft Project Director.

2 GENERAL

2.1 DISPLAY LEGIBILITY

2.1.1 The suitability of crew station lighting and displays is dependent on several factors including;

(i) Crewstation configuration	From narrow cockpits of high altitude fighters to the cabins of transport aircraft.
(ii) Mode of operation	Day (with unaided vision)or Night (with or without NVIS)
(iii) Ambient lighting conditions	Location and characteristics of ambient light.
(iv) Characteristics of the display	Emission from active and inactive parts of crew station displays and diffuse and specular reflectivity of crewstation display surfaces.

(v) Crewmember Performance — Performance of Crewmember with respect to visual tasks.

2.1.2 The conventional method of specifying the adequacy of display lighting (which specifies colour, luminance and luminance contrast) has failed to fully represent the effect that the above factors have on display legibility. Paras 3 to 5 defines the conventional method of defining display performance, para 2.1.3 describes the Mathematical Modelling Technique and para 2.2 details the preferred method for specifying display performance.

2.1.3 Predictive Sunlight Legibility Mathematical Modelling Techniques[3]

(i) Mathematical modelling techniques have been developed which utilise the above factors (para 2.1.1) to quantify, and thereby facilitate the prediction of, display legibility (as perceived by the crewmember) under extreme lighting conditions. This allows the acceptability of a display, in terms of sunlight legibility, to be precisely specified.

(ii) The level of display legibility (as expressed quantitatively using the mathematical modelling technique described in para 2.1.3(i)) that is required by a particular display is dependent on the ambient lighting condition to which that display will be exposed. Displays which are positioned in the crewstation such that they are exposed to a severe ambient lighting environment will require a higher level of display legibility than those displays exposed to a less severe ambient lighting environment. The mathematical modelling technique may also be utilised to identify, unambiguously, the most suitable display technology where the available display technologies have not progressed sufficiently to achieve the required level of display legibility.

(iii) The basis of these techniques is illustrated in Fig 1. In essence the spectral emission of the display foreground (active part of display) and the spectral emission of the display background (inactive part of display) are mathematically combined with the specular and diffuse ambient illumination which enters the eye directly and which enters the eye through diffuse and specular reflection from the display (see Fig 2).

(iv) The resulting power spectra represent the total light energy entering the pilots eye from the display foreground and background. These spectra are then analysed further to simulate the eye response. The result of this analysis is expressed in terms of "chrominance just noticeable difference" (CJND) and the "luminance just noticeable difference" (LJND).

(v) The chrominance just noticeable difference (CJND) is the smallest chrominance difference that would normally just be noticed when there is no illuminance difference (ie iso-luminance). Similarly the luminance just noticeable difference (LJND) is the smallest luminance difference that would normally just be noticed when there is no chrominance difference (ie iso-chromatic). Both CJND and LJND relate to a detection probability of 50% as determined through subjective evaluation by Standard observers. The CJND and LJND are combined to give the Perceived Just Noticeable Difference (PJND). The PJND difference represents the smallest combination of luminance and chrominance difference (between active and inactive parts of the display) which would just be noticeable. (see reference No. 4 for detailed description.)

(vi) For a display to be adequately sunlight legible it must have CJND, LJND and PJND values above a given value. The given value, i.e. the design criteria, should be defined by the Aeroplane Project Director. Consequently the complete visual aspects of a display may be fully specified by specifying PJND, CJND and colour (where colour is as defined in para 3.5). (It is not necessary to specify LJND. This is because LJND can be derived from PJND and CJND.)

(vii) This technique is applicable to, and allows direct comparison between, emissive, transmissive, transflective and reflective displays.

2.2 PREFERRED METHOD OF SPECIFYING DISPLAY PERFORMANCE

2.2.1 A predictive Mathematical Modelling technique should be implemented where possible. Where implementation of predictive Mathematical Modelling techniques is not possible due to the complexities of these techniques (ie specification of acceptable LJND, CJND and PJND and specification of the spectrum of the prevailing ambient lighting conditions) then the Discrimination Index, as specified in ARINC 725 (Electronic Flight Instruments (EFI)), or Mil-S-22885/108, should be used to define display legibility. The 'conventional' (luminance contrast) method of defining display legibility is inferior to both the Discrimination Theory and Mathematical Modelling technique and should only be used where expressly specified by the Rotorcraft Project Director. Paras 3 to 5 describe the 'conventional' method in more detail.

2.3 NIGHT VISION IMAGING SYSTEM (NVIS)

2.3.1 The design criteria for NVIS compatible crewstation lighting and displays is beyond the scope of this document. It is fully discussed in Leaflet 116/1 (JAC Paper No.1131).

3 DESCRIPTION OF TECHNICAL TERMS

3.1 AMBIENT ILLUMINATION LEVEL RANGE

3.1.1 The range of naturally occurring levels of illumination typically encountered in a crewstation (and to be considered by this leaflet) is between 10^{-4} lux (equivalent to obscured starlight) and 10^{5} lux (measured at altitudes above cloud). The illumination should be considered to be diffused (ie equal energy from all directions in the superior hemisphere) up to at least 10^4 lux; between 10^4 lux and 10^5 lux the distribution becomes predominately directional from the sun. "Day" is considered to be between 10^2 and 10^5 lux; "night" is between 10^{-4} and 10^2 lux. Levels of artificial illumination will fall well within these values and will be predominantly directional

3.2 CONTRAST

(i) Contrast can be defined as either;

$$C_1 = \frac{\text{measured character luminance}}{\text{background luminance}}$$

or as;

$$C_2 = \frac{\text{measured character luminance - background luminance}}{\text{background luminance}}$$

Both C1 and C2 are in common use however it is evident that:

$C_2 = C_1 - 1$

Consequently the method of calculation should be specified where ever contrast is specified. It is also essential that the ambient lighting illumination is also specified. Where ambient illumination is non zero then the luminance values should be inclusive of ambient light reflected from the display.

(ii) For Cathode Ray Tubes (CRTs), or other displays which produce a video type display, the display modulation at a specific spatial frequency should be specified.

$$\text{modulation} = \frac{L_{max} - L_{min}}{L_{max} + L_{min}} \quad \text{(at a specific spatial frequency).}$$

L_{max} = maximum display luminance when a sinewave modulated pattern, with a pre defined spatial frequency, is displayed.

L_{min} = minimum display luminance when a sinewave modulated pattern, with a pre defined spatial frequency, is displayed.

Both L_{max} and L_{min} should be measured under conditions of zero ambient illumination (ie below $0.3 Cd/m^2$).

(Chromate contrast can also play a large part in overall display legibility. Chromatic contrast, however, requires methods similar to the Discrimination Index, or Mathematical Modelling Technique, to quantify its contribution to display legibility.)

3.3 EMISSIVE DISPLAYS

3.3.1 These are displays relying on internally generated luminance emission throughout the expected ambient illumination.

3.3.2 Direct emissive displays directly modulate the luminance at source (e.g., CRT). Indirect emission displays modulate a fixed source by interference (e.g., projector map).

3.4 PASSIVE DISPLAYS

3.4.1 These are displays relying on reflected natural luminance supplemented as required by artificial luminance sources.

3.5 COLOUR

3.5.1 All colours should be defined in chromaticity co-ordinates related to Publication CIE No 2.2 (TC - 1.6) 1975 "Colours of Light Signals" or the 1976 CIE-UCS "u" - "v" colour co-ordinates.

(Chromatic separation falls with increasing ambient illumination (ie colours desaturate). Only Mathematical Modelling techniques provide a means of defining colour separation under these ambient lighting conditions.)

3.6 AUTOBRILLIANCE

3.6.1 Autobrilliance is the automatic control of the brightness of CL&D based on ambient conditions. It is important to monitor both the illumination level falling on the crewstation displays and the ambient illumination level falling into the pilots eyes. Monitoring both the display and the pilots eyes will allow the "flying into the sun" condition, where the displays are in sharp shadow whilst the pilots eyes are adapted to the solar disk, to be identified by the autobrilliance control which shall then select the maximum display brightness.

3.6.2 The illumination level range over which autobrilliance has control should be 10^2 to 10^5 lux.

3.6.3 The time response of the autobrilliance control (ie how quickly the autobrilliance function should respond to rises or falls in ambient illumination) shall be in the range of 0.5 to 2 seconds.

4 REQUIREMENT FOR, PASSIVE DISPLAYS

4.1 CONTRAST

4.1.1 Displays exposed to illumination levels in the "day" range should exhibit a contrast of 10 to 20 between white markings and the black background. Appropriate correction factors must be applied to coloured areas, either markings or as background.

4.1.2 Displays exposed to illumination levels in the "night" range will require supplementary lighting, either integral to the display or by floodlighting. The contrast, for displays being illuminated by supplementary lighting, should comply with that specified in para 4.1.1. During night operation some background illuminance should be provided to prevent auto-kinesis. (Auto-kinesis is a phenomena by which brightly lit displays, which are seen against a dark background, appear to drift.)

4.2 LUMINANCE

4.2.1 The generally accepted normal illuminance of white markings with the illumination sources operating at rated voltage is 3.43 ± 1.72 cd/m^2 (from STANAG 3224). There appears to be little purpose in amending this figure for the majority of military applications except, possibly, for rotorcraft frequently operating in brightly lit civil airports.

4.2.2 These values are applicable to all aircrew stations, however, the luminance range may be reduced if the range of ambient illumination is restricted (Note: errors in reading will start to increase at levels below 0.35 cd/m^2 - 10% of nominal luminance at rated voltage).

4.2.3 The illumination sources must be capable of being dimmed to produce display luminance typical of that required for night operations. This is in the region of 1Cd/m^2 (see Royal Aerospace Establishment, FS(F) Working Paper 6, para. 7.9.3). Dimming to levels much lower this will not be utilised by crewmembers and may constitute design criteria which is not achievable by some display technologies.

4.2.4 CL&D inside the crewstation should have a uniformity of illumination better than 2:1 throughout the entire dimming range.

4.2.5 CL&D should be designed, through careful control of stray light, such that CL&D are not reflected in crewstation glazing into the field of view of the crewmember. Where the crewstation is configured such that reflections cannot be eliminated in this way then those parts of the instrument panel which cause reflections in the glazing should be capable of being dimmed independently from those parts of the instrument panel which are not reflected in the glazing. Side consols and panels adjacent to glazing are particularly prone to causing such reflections. Separate dimming of "CL&D prone to reflection in the crewstation glazing" and "CL&D not prone to reflection in the crewstation glazing" may, in some instances, be more desirable than left side/right side dimming of crewstations.

4.2.6 Areas, in the crewstation instrument panels and consols, of very low luminance (ie 'black holes') should also be avoided (see para. 4.1.2).

4.3 COLOUR

4.3.1 The ideal illuminant colour, to ensure that any coloured segments, markings etc., retain their true colour, is equal energy white, x = 0.333; y = 0.333. These colour co-ordinates apply at the light sources rated voltage. The colour temperature shift experienced when tungsten lamps are dimmed (ie operating in the orange" band of the spectrum) is less significant when the lamps are blue filtered than when the lamps are unfiltered (Refer to Mil-C-25050 (for specification of lamp colour) and Annex A to STANAG 3224 (Edition 5)).[1]

4.3.2 If multicolour displays are used, colour should be reserved as follows in accordance with Chapter 105 para 13.[2]

Red	Dangerous conditions (e.g., enemy action, out of tolerance conditions, severe weather).
Yellow/Amber	Cautionary information.
Green/White	Normal conditions.

Note: Other colours, if available, may be used to enhance information presentation and to minimise clutter.

4.3.3 Where Night Vision Imaging Systems (NVIS) are likely to be used in the crewstation then the colour of crewstation lighting and displays that are operating whilst the NVIS are being used shall be dominated by NVIS design criteria (refer to Leaflet 116/1 - JAC Paper No.1131).

4.3.4 With Gen II NVIS this will generally mean that crewstation lighting and displays will be blue-green only with the possibility of yellow captions or warnings. Visual white and three colours CRT displays will be prohibited by Gen II NVIS criteria. Gen III crewstations utilising current filters may be paler blue-green (if required), and can include 3 colour CRT displays and probably red as well as amber and yellow warning captions. The above statements assumes the use of those filters specified in FS(F) Working Paper 6: "Cockpit Lighting Standards and Techniques for use with Night Vision Goggles".

4.4 STRAY LIGHT

4.4.1 It is essential that aircrew do not see illumination sources either directly or by reflection; this applies equally to floodlights or displays with integral lighting. It is difficult to lay down quantitative data but displays with integral lighting should cater for viewing at angles up to 60^{o} to the normal. The amount of stray light from displays with integral sources, and from floodlights and spotlights, requires control to minimize general aircrew station illumination and to ensure that the sources are not visible from outside the aeroplane. This may be achieved with floodlights by shading the floodlight to ensure that only the essential areas are illuminated. Where the location of the floodlight is such that stray light cannot be eliminated through shading then the floodlight should be replaced with a Spotlight (spotlights utilise a lens and aperture systems in order to precisely control the area being illuminated).

5 REQUIREMENTS FOR EMISSIVE DISPLAYS

5.1 CONTRAST

5.1.1 Contrast for monochromatic displays can be defined in accordance with the definition stated in para 3.2. However, when multicolour displays are considered, it is necessary to quantify colour contrast as well as luminance contrast. Colour differences can be measured in terms of 1931 CIE x and y chromaticity co-ords and transformed to 1976 USC u' and v' chromaticity co-ords. When the u' and v' values of the display background and the various colours of the display foreground are plotted on the CIE 1976 CIE diagram the distances between chromaticity co-ordinates will represent equal colour contrast irrespective of the area of the diagram used. Monochrome displays, except those which consist of a white on a black background, will contain an element of colour contrast which may be significant when the luminance contrast is low.

5.1.2 Colour contrast and its effect on sunlight legibility are more realistically dealt with by the Mathematical Modelling technique (refer to para 2.1.3). ARINC 725 provides a similar but much simplified method for defining colour contrast.

5.2 CONTRAST FOR MONOCHROMATIC EMISSIVE DISPLAYS

5.2.1 Where the definition for contrast given in para 3(i) is used then the quantitative data listed below should be adhered to:

(i) CRT displays. Contrast, under an ambient illumination of 10^3 and 10^5 lux, should be greater than 6.5. The contrast specified is C1 para 3.2(i). (To achieve sunlight legibility under high ambient illumination the display contrast under low ambient (ie ≤ 200 lux) should be greater than 70.)

The required contrast should be verified with the Rotorcraft Project Director prior to commencement of the Project (this is necessary in order to allow the current rapid development in CRT technology to be fully exploited).

Maximum peak line brightness (applies to both cursive and raster drawn displays) should exceed:

$$\left(\frac{\text{maximum ambient illumination (in lux)}}{300} \right) \text{ cd/m}^2$$

e.g., in 10^5 lux:

$$\frac{100{,}000}{300} = 330 \text{ cd/m}^2$$

Peak line brightness should be measured using a slit whose dimensions are such that the area being measured on the display surface is equivalent to, or greater than the Active Measured Area (AMA). The AMA is the area of the base of a cone which has a full angle of 4 minutes of arc and which has a 'base to apex' distance which is equal to the 'pilot eye to display surface' distance. (NB. 4 minutes of arc has been specified because even though the eye can detect lines finer than 1 minute of arc it can only distinguish brightness in lines which are greater than 4 minutes of arc.)

All filters, implosive shields, EMI coatings and anti reflection coatings shall be in place during the above measurements.

(ii) LED displays. Although the requirements stated above for CRTs would be ideal, current large, dot matrix, arrays achieve only C>4 in 10^5 lux; but the requirements of C>9.5 in 10^3 lux and below can be met. A high degree of contrast enhancement filtering is required because the LED junctions are extremely reflective. Circular polarising filters are particularly effective in this application. LED luminance should be used as stated for CRTs above when viewed through the filter.

(iii) Hidden Legends. Hidden Legends in illuminated push buttons and Central Warning Panels (CWP) captions etc., must achieve C>0.6 in 10^5 lux. Although this contrast appears low, it is the highest that is achievable with existing technology. Increasing the intensity of the illumination sources to improve the contrast results in excessive operating temperatures. Currently only tungsten filament lamps can provide sufficient light to achieve sunlight readability.

5.2.2 The modulation of monochrome CRT's shall be greater than 0.2 inside a central circle with a diameter equal to the width or height (which ever is smallest) of the display area. The contrast should be greater than 0.1mm over the entire display area. This contrast should be achieved whilst a sinusoidal test pattern with a spatial frequency of 600 lines per tube height is being generated on the display. The required contrast should be verified with the Rotorcraft Project Director prior to commencement of the Project (this is necessary in order to allow the current rapid development in CRT technology to be fully exploited.)

5.2.3 Para 4.2.3 to 4.2.6, 4.3.2 to 4.3.4 and 4.4 also apply to emissive displays.

6 REQUIREMENTS FOR FLOODLIGHTS

6.1 Because of the wide variety of floodlighting requirements, it is difficult to define quantitative data except for the specific applications listed below:

(i) Anti-Dazzle Lights. Where the primary flight instruments are of the passive type display, anti-dazzle lights should provide 1,000 lux minima over the area occupied by the primary flight instruments. Colour of illumination should be white. (Anti-Dazzle lights (refer also to Chapter 105, para 15.9) are switched on when a high brightness flash (such as a lightning flash) is anticipated. The Anti-Dazzle light allows the crewmembers to adapt their eyes to the anticipated high brightness environment. Anti-Dazzle light could be used to reduce the effect of an unexpected high brightness flash.)

(ii) NVIS Lights. Where there is a requirement for aircrew to use NVIS's, infra-red free floodlights will be required. Typical illumination levels would be 10 lux.

REFERENCES

Reference	ASCC Air Std	STANAG	BAe
1	)10/19	3224	-
2	)10/30	3370	-
3	-	-	GENODOC.408
4	-	-	BAe-WSE-RP-EFA-CPT-056

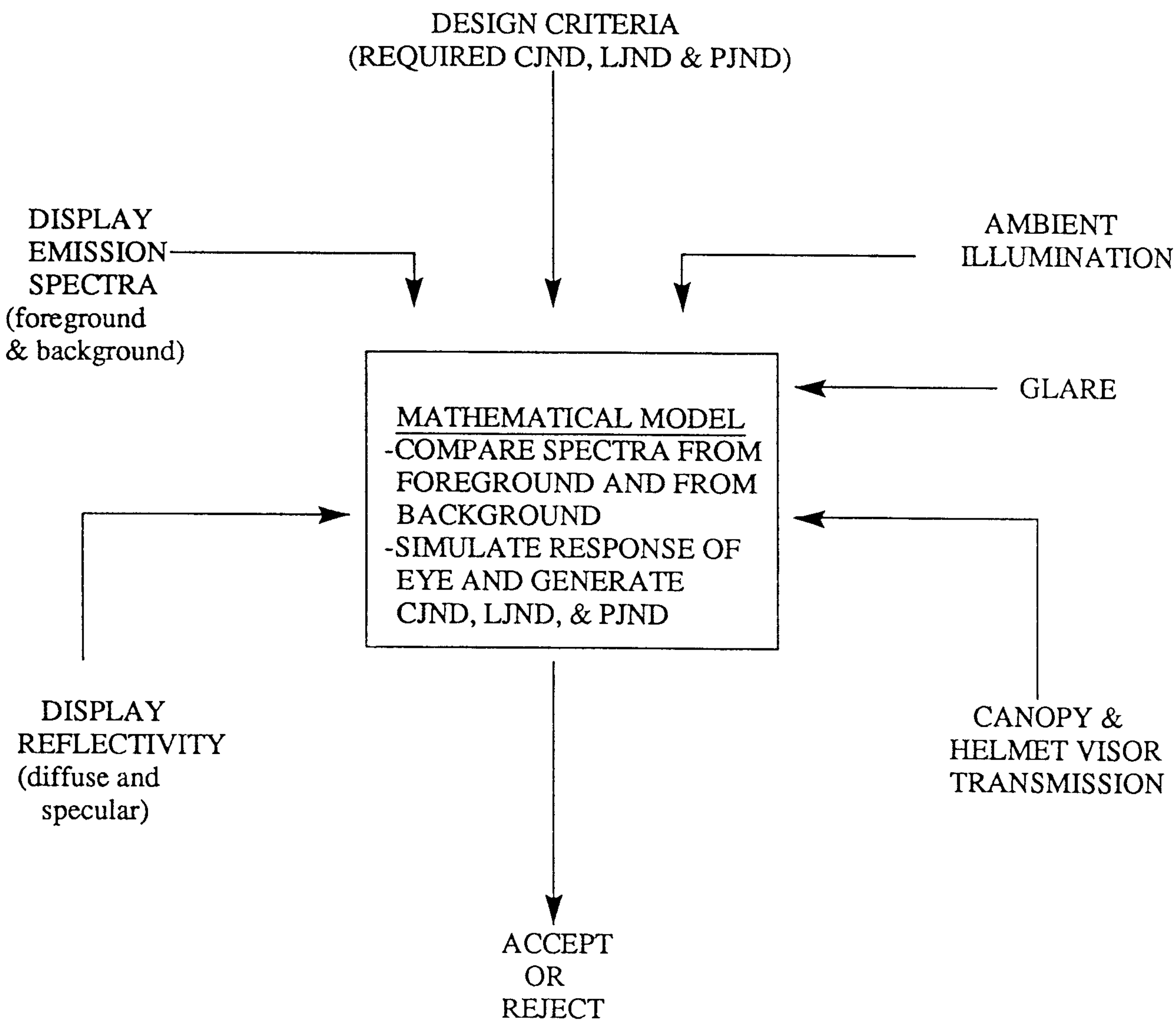
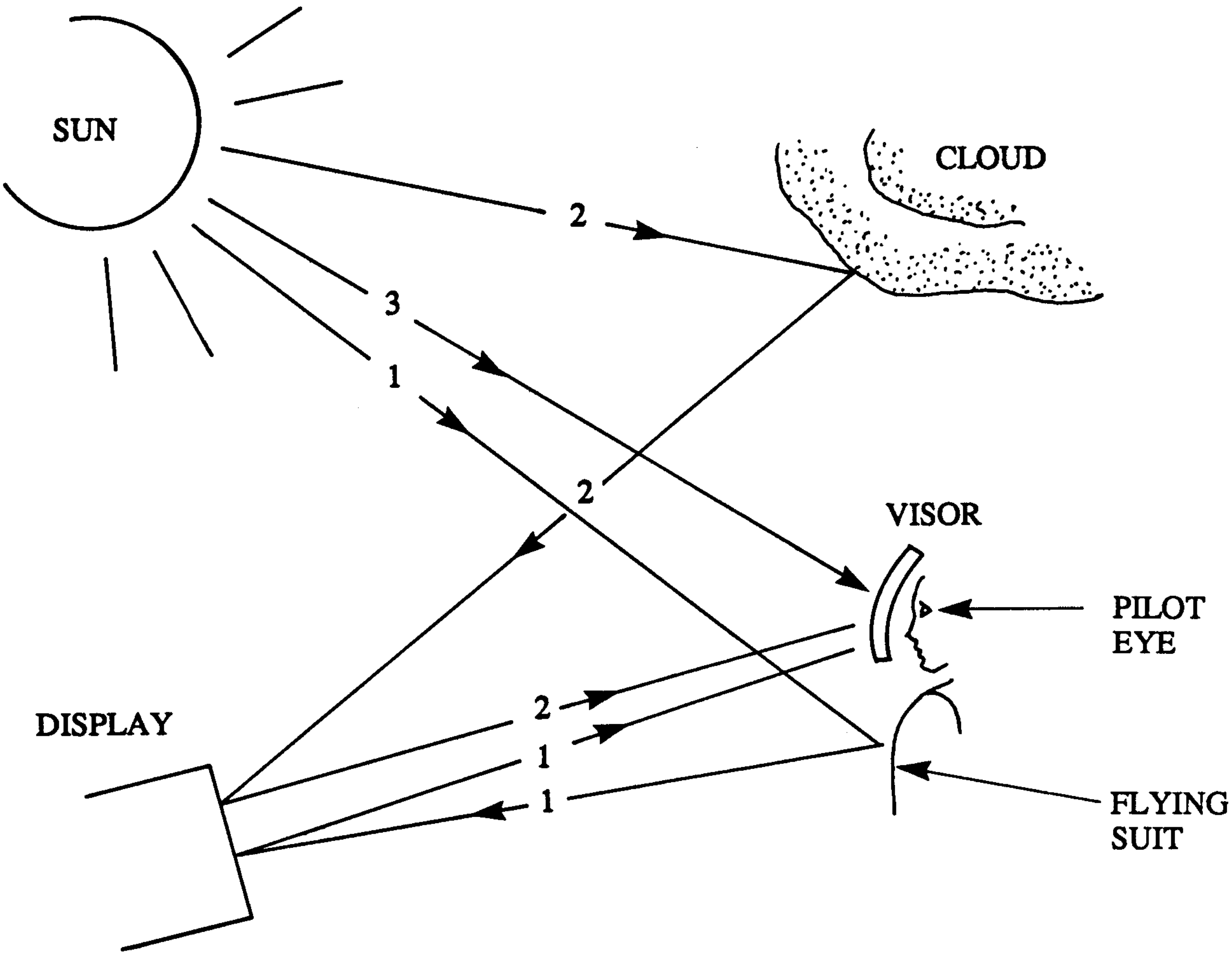


FIG 1 PREDICTIVE SUNLIGHT LEGIBILITY MATHEMATICAL MODELLING TECHNIQUE

SUN FORWARD



1 SUN - FLYING SUIT - DISPLAY - PILOT EYE

2 SUN - CLOUD - DISPLAY- PILOT EYE

3 SUN - PILOT EYE

FIG 2 DISPLAY LIGHTING CONDITIONS SIMULATED BY THE PREDICTIVE SUNLIGHT LEGIBILITY MATHEMATICAL MODELLING TECHNIQUE

CHAPTER 106

PILOT'S STATION - LAYOUT[1]

1 INTRODUCTION

1.1 The requirements of this chapter are, unless otherwise specified, applicable to all types of rotorcraft, and their purpose is to achieve standardization in the layout of pilots' stations.

1.2 Each new station, while in the mockup stage, shall be examined by the Rotorcraft Project Director and compared with the criteria contained in this chapter. Trials to determine the suitability of the pilot's station layout shall be held with personnel harnessed in the seat(s) and wearing the clothing and equipment detailed in the Aircrew Equipment Assembly Schedule (AEA) appropriate to the particular rotorcraft. Such personnel shall reflect the anthropometric limits determined for the particular rotorcraft.

2 DEFINITIONS

2.1 AIRCRAFT DESIGN EYE POSITION - (Fig.1)
2.1.1 A reference point, fixed with respect to the aircraft for the establishment of aircrew external and internal crew station vision and for crew station geometry (see also Chapter 104).

2.2 REFERENCE PLANE FOR VIEW ANGLES TO LEFT AND TO RIGHT
2.2.1 The plane through an aircraft design eye position parallel to the aircraft plane of symmetry (XZ plane).

2.3 FLIGHTPATH VISION PLANE
2.3.1 The plane through an aircraft design eye position parallel to the aircraft flightpath vector and perpendicular to the aircraft plane of symmetry.

2.4 LATERAL VISION REFERENCE PLANE
2.4.1 The plane through an aircraft design eye position parallel to the aircraft fuselage plane (XY plane).

2.5 PILOT'S EYE REFERENCE POINT
2.5.1 The midpoint between the centres of the pilot's eyes.

2.6 AIRCRAFT NEUTRAL DESIGN SEAT POSITION
2.6.1 A point, fixed with respect to the aircraft, which coincides with the seat reference point when the seat is adjusted so that the eye reference point for a 50th percentile pilot coincides with the aircraft design eye position.

3 FIRST PILOT'S STATION

3.1 In rotorcraft with a single pilot seated off-centre, the pilot shall sit on the right of the cockpit.[2]

3.2 In rotorcraft with side-by-side pilots, the first pilot shall occupy the right hand seat.[2]

3.3 In rotorcraft with the pilots seated in tandem, the specification will state which seat the first pilot shall occupy.

4 EXTERNAL VIEW

4.1 The external view from the pilot's station shall be in accordance with Chapter 104.

5 REACH

5.1 STICK REFERENCE POINT

5.1.1 The stick reference point is the location at which the operator's second (middle) finger will be in contact with the forward face of the stick grip with controls and trims neutral.

5.2 COLLECTIVE CONTROL REFERENCE POINT

5.2.1 The collective control reference point is the location at which the operator's second (middle) finger will be in contact with the downward face of the control grip.

5.3 NOMINAL CONTROL TRAVEL MIDPOINT

5.3.1 The nominal control travel midpoint is the location of the collective control reference point when midway between the full-up and full-down positions.

Note: Normally, the dimension of the eye height for a 50th percentile pilot is based on seats with backrests in a vertical position. In determining the neutral seat reference point, adequate allowance must be made for an angular position of the backrest.

5.4 All controls shall be within normal reach and sight of the pilot when harnessed in his seat and wearing the clothing and equipment detailed in the Aircrew Equipment Assembly Schedule appropriate to the particular rotorcraft. No controls, which are in regular use in flight, shall be positioned aft of the pilot's shoulder line.

5.5 The following factors shall be considered when deciding whether a control should be reached with or without harness locked:

(i) the rapidity with which the control needs to be operated e.g., all emergency controls shall be within reach with harness locked,

(ii) the period during flight when operation of the control is required, e.g., controls which need to be operated in the circuit or on an operational role which requires the harness to be locked, shall be within reach with harness locked, and

(iii) the frequency of operation of the control during a flight, i.e., the more frequently a control need be operated, the more necessary it is that it shall be operated with harness locked.

6 SEATING

6.1 SYMMETRY OF SEAT

6.1.1 The plane of symmetry of the pilot's seat shall be parallel to the plane of symmetry of the rotorcraft.

6.2 SEAT ADJUSTMENT

6.2.1 Sufficient vertical seat adjustment shall be provided to allow the specified percentile range aircrew members to place their eyes at the level of the aircraft design eye position. A method of confirming this position shall be provided. Fore-and-aft adjustment of the seat should normally be provided; the recommended range of adjustment is ± 50 mm about the neutral design seat position.

6.2.2 When provision is made for adjusting the seat in a fore-and- aft direction, a positive lock shall be provided to prevent fore-and-aft movement of the seat, when locked in any flying position, under the loads occurring during flight and/or the crash landing conditions of Chapter 307 (see also Chapter 107, Table 3, Item 6).

6.3 SIDE-BY-SIDE SEATS

6.3.1 In rotorcraft fitted with side-by-side pilot's seats, particular attention shall be paid to the ease of movement into and out of the first pilot's seat where such an interchange is possible during flight.

7 HEAD CLEARANCE

7.1 A minimum spherical envelope of 254 mm shall be provided from the aircraft design eye position to ensure a minimum of 50 mm head clearance from the canopy (see Fig.1).

8 DIRECTIONAL CONTROL PEDALS (See also Chapter 107 para 2.3)

8.1 RANGE OF ADJUSTMENT

8.1.1 The range of directional control pedal adjustment should be agreed with the Rotorcraft Project Director.

8.2 RANGE OF PEDAL MOVEMENT

8.2.1 The travel of the pedals should be ± 102 mm measured about their central position for any setting of the control pedal adjustment.

8.3 SIZE OF PEDALS

8.3.1 Pedals shall accommodate the largest flying boot with overshoe.

8.4 CLEARANCE BELOW PEDALS

8.4.1 The clearance between the bottom of the pedals and the floor shall be such as to ensure adequate directional control without the danger of inadvertent operation of the toe brakes or of trapping the pilot's heels.

9 CONSOLES

9.1 The height of the side and/or centre console shall be based on the smallest specified percentile aircrew member's effective arm reach with the seat in the full up position.

10 INSTRUMENT FLYING PRACTICE EQUIPMENT

10.1 Provision shall be made for the requisite instrument flying practice equipment.

11 STOWAGES

11.1 A map stowage, which shall also include a separate compartment for Aircrew Manuals/Pilot's Notes and Flight Reference Cards (FRCs), shall be provided for each pilot.

11.2 On all transport rotorcraft, a stowage within easy reach of either the first or second pilot shall be provided for a slide rule type load and centre of gravity computer.

12 CROSS REFERENCES TO OTHER CHAPTERS

12.1 A number of requirements directly related to the layout of the pilot's station appear elsewhere in this publication and the most important of these are listed below (see also the Alphabetical Index).

Chapter	Para	Subject
101	3	Clear vision
	5	Air conditioning
102	-	Emergency escape
103	-	Operational colouring and markings
104	-	View and clear vision
105	-	Crew stations - General requirements
107	-	Pilot's cockpit - Controls and instruments
111	-	Restraint and parachute harness for aircrew
307	-	Crash landing and ditching

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/55	3639
2	10/20	3225

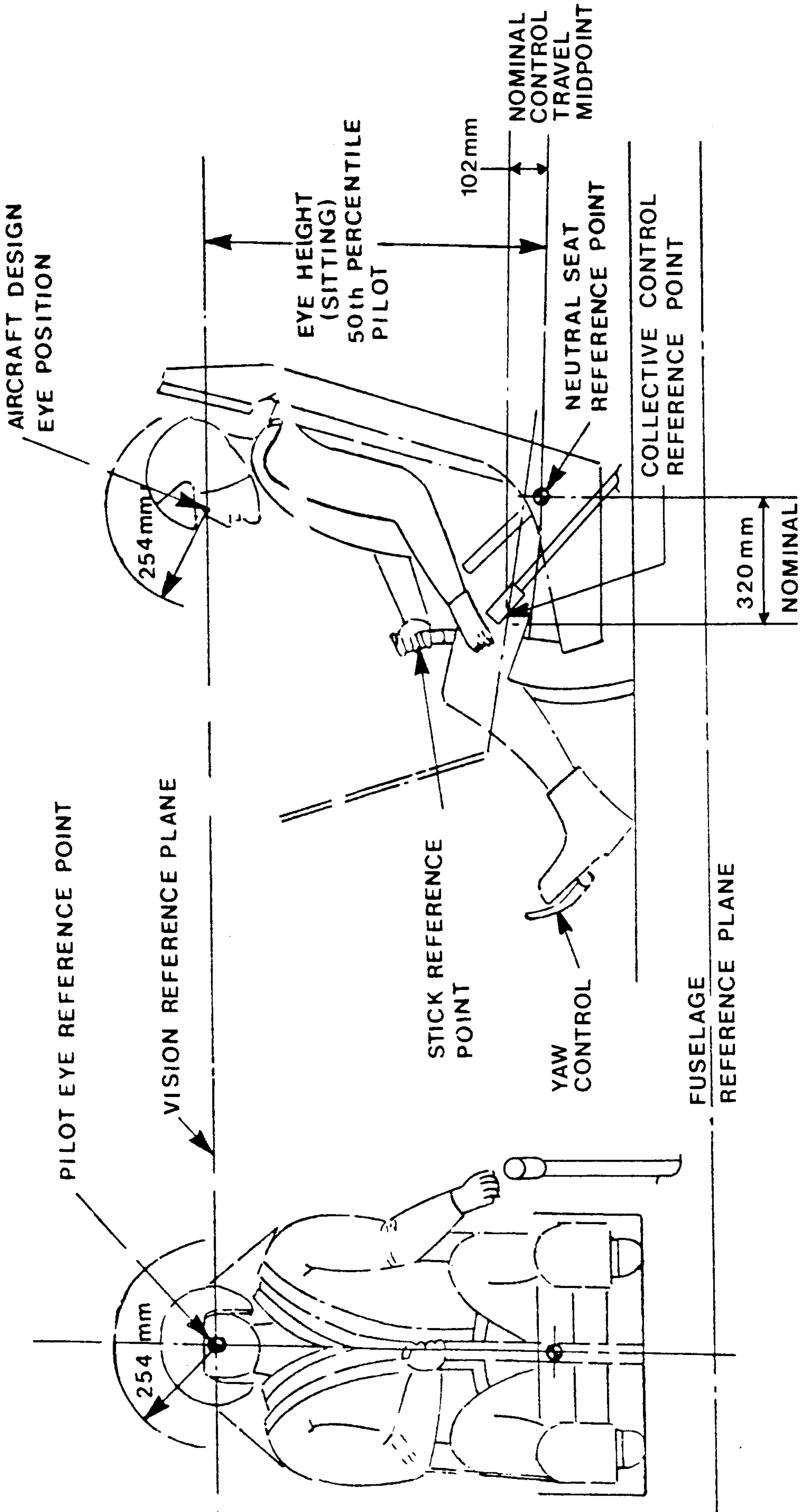


FIG.1
AIRCREW STATION DEFINITION AND GEOMETRY LAYOUT
FOR ROTORCRAFT

CHAPTER 107

PILOT'S COCKPIT CONTROLS AND INSTRUMENTS

1 INTRODUCTION

1.1 The requirements of this chapter are aimed at achieving standardization in the type, method of operation and positioning of the controls and instruments necessary for the operation of certain services. Those controls and instruments which are appropriate to the role of the rotorcraft shall be fitted at the pilot's station in accordance with the requirements.

1.2 Each new pilot's station shall be examined by the Rotorcraft Project Director at the mock-up stage so that any changes or additions required may be agreed. The examination shall be made with personnel wearing clothing and equipment detailed in the Aircrew Equipment Assembly Schedule (AEA) appropriate to the particular rotorcraft, and harnessed in the seat(s); such personnel shall approximate in size to the extreme limits specified for Service pilots (see Volume 1, Leaflet 105/3).

2 FLYING CONTROLS AND INDICATORS (see also Table 1)

2.1 CYCLIC PITCH CONTROL

2.1.1 A cyclic pitch control operable by the right-hand shall be provided for each pilot[1]. Each pilot's control shall, unless otherwise agreed, carry those of the controls listed below which are appropriate to the role of the rotorcraft:

	CONTROL	LOCATION
(i)	Trim switch when trimming is by an electrical system[2]	Top rear face of grip.
(ii)	Camera Control[2]	-
(iii)	Press to transmit/ICS switch[2]	Forward face of grip.
(iv)	Armament release/firing trigger[2]	Top forward face of grip.
(v)	Cargo release button or[2] parachute jump light switch, with guard	Top left-hand side of grip.
(vi)	Winch cable cutter button, with guard	Top right-hand side of grip.
(vii)	Trim release button[2]	Middle of grip left-hand side.
(viii)	Automatic flight control system disengage button [2,3]	Lower portion of forward face of grip.
(ix)	Winch control switch (see Note 2)	-

Notes 1 The location of those controls and switches for which no specific location is stated will be decided at the Mock-up Conference (see para 1.2).

2 This control may alternatively be located on the collective pitch control (see para 2.2).

2.2 COLLECTIVE PITCH CONTROL

2.2.1 A collective pitch control, operable by the left hand, shall be provided for each pilot[1]. Each pilot's control shall, unless otherwise agreed, carry those of the controls listed below which are appropriate to the role of the rotorcraft:

(i) landing lamp on/off switch,

(ii) landing lamp beam angle control switch,

(iii) powered flying control selector (see however Table 1, Item 2),

(iv) winch control switch (see Note 2),

(v) partial automatic flight control system disengage control,

(vi) external armament and drop tank jettison control (see however Table 9, Item 4),

(vii) emergency flare firing switch(es), with guard,

(viii) press to mute switch (see Note 2),

(ix) hold/haul down gear control.

Notes 1 The exact location of the above controls and switches on the collective pitch control will be decided at the Mock-up Conference (see para 1.2).

2 This control may alternatively be located on the cyclic pitch control (see para 2.1).

2.3 DIRECTIONAL CONTROL (see also Chapter 106, para 8)

2.3.1 Wide range quick-adjusting directional control pedals shall be provided for each pilot[1].

2.3.2 Both pedals shall be adjustable simultaneously. Each pilot shall be provided with his own independent adjustment control.

2.4 SUPPLEMENTARY FLYING CONTROLS[1]

2.4.1 When movement of any aileron, elevator, rudder or supplementary flying control surfaces is necessary during normal flight manoeuvring, then this control movement shall occur without the pilot taking any action other than the normal movements of cyclic pitch, collective pitch and directional controls. If these supplementary flying control surfaces are not required during normal manoeuvres but are used at the pilot's discretion, they shall be engaged and disengaged by a control within easy reach of the pilot.

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2.5 FRICTION CONTROLS[1]

2.5.1 These shall provide variable control friction, as demanded by the pilot, on the desired control. Their authority shall be such that no friction control shall be capable of locking any flying control in any position to the extent that movement of that control by any pilot is impossible. Each friction control shall be located adjacent to its relevant flying control, and shall be provided at the first pilot's station only.

2.6 AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) CONTROLS[3]
(see BS 185 Section 5: 1969 Item 5328 and Table 1, Item 1)

2.6.1 AFCS Failure Monitoring and Indication - Rotorcraft with AFCS shall be provided with signals in accordance with para 14 as follows:

(i) Means by which the pilot can check serviceability prior to take-off.

(ii) Automatic indication of in-flight malfunctions.

2.6.2 Disengaged Signal - Systems with safety monitoring features that automatically disengage one or more axes of control shall provide a positive indication of disengagement.

2.6.3 Manual Override - The AFCS shall be capable of being overridden by the pilot.

2.6.4 Trim Meter Indication - AFCS trim indications shall indicate the direction of the attitude change which the rotorcraft would experience at the disengagement of the AFCS.

2.6.5 System Interlocks - System interlocks shall be provided to prevent engagement or selection of incompatible functions or modes of operation.

2.6.6 Disengage Button (Emergency Disconnect). There shall be an AFCS electrical disengage button on the cyclic or collective control (see Table 1 Item 1 (b)).

2.6.7 Lighting. Controls and panels used in AFCS shall be lit in accordance with the applicable requirements outlined in Chapter 105 para 15 (see also Leaflet 105/4 and Chapter 116 (JAC Paper No 1131)).

2.6.8 Markings. Controls and panels used in AFCS shall be marked in accordance with DEF STAN 66-26.

2.6.9 Control Colour Schemes. Emergency control colour schemes shall be in accordance with Chapter 103, para 2 and AFCS panels and control colour schemes shall be in accordance with Chapter 103, para 6.

2.7 TRIM INDICATORS

2.7.1 When trim indicators are fitted, they shall be readily visible to each pilot and shall show clearly and without ambiguity the effect of movement of each trimming control.

2.8 UNDERCARRIAGE POSITION INDICATORS (see also Table 1, Item 4c)

2.8.1 An indicator shall be fitted to give visual indication for each retractable undercarriage unit as follows:

(i) a green lamp to be alight only when the unit is in the safe landing position and the locks are properly engaged, and

(ii) a red lamp to be alight at all times other than when:

(a) the unit is in the safe landing position and the locks are properly engaged, and

(b) the unit and its door(s) are in the safe flying position and the locks are properly engaged.

2.8.2 In addition, on rotorcraft having installations in which the door closes after lowering the undercarriage, the red lamp shall remain alight until the door is in the correct landing position.

2.8.3 It is desirable that a warning device should be fitted to guard against landing with the undercarriage retracted. The warning device, which shall not be incorporated in the standard warning system of para 14.2, shall be visible and/or audible to both pilots and shall operate with engine power both on and off.

3 WINCH CONTROLS (see Table 1, Item 9)

4 GROUND MANOEUVRING CONTROLS (see also Table 2)

4.1 WHEEL BRAKES CONTROL

4.1.1 Control of the wheel brakes system shall be affected by toe pedals integral with the directional control pedals[1] to provide differential operation in the natural sense. The brake pedals shall be so installed in the directional control pedals that it shall be possible use the brake pedals without moving the feet from the directional control pedals irrespective of the position of the directional control pedals.

5 ENVIRONMENT AND ESCAPE CONTROLS AND INSTRUMENTS (see Table 3)

6 CONTROLS AND INDICATORS FOR SHIPBORNE OPERATION (see Table 4)

7 ENGINE POWER CONTROLS (see also Table 5)

7.1 DEFINITIONS

7.1.1 Throttle - A twist grip and/or lever which controls the fuel supply to the engine.

7.1.2 Engine condition control - A control which has the multiple function of selection OFF, ground idle and flight settings. In the "flight" setting it may also act as a rotor speed governor.

7.1.3 Rotor speed select control - A control which has the function of setting or altering the governed speed of the engine/rotor throughout the full range from ground idle to flight condition.

7.1.4 Rotor speed governor - A control to maintain automatically rotor rpm in the flight condition.

7.1.5 Flight idle control - A control for preventing a turbine engine compressor from idling in flight below a predetermined minimum figure.

7.1.6 Drive selector - A control to enable an engine to be coupled to either the rotor transmission system or the accessories only.

7.2 NUMBERING OF ENGINES AND THEIR ASSOCIATED CONTROLS AND DISPLAYS[10]

7.2.1 Engines shall be numbered in accordance with BS2 M41, and their associated controls and displays shall follow the same convention.

7.2.2 Every engine which has any individual pilot-operated control (e.g., starter control, HP cock or throttle control) shall have an individual number even though two or more engines may be coupled together under a single control during some stage of flight. Where two or more engines in a group are always controlled by a single set of pilot-operated controls, these engines and their controls shall have a single number and each engine of such a group shall be identified by a capital letter suffix. (see Fig.2(b) of BS2 M41).

7.2.3 Engines whose relative position can vary during flight shall be numbered in accordance with their geometric position during normal cruise flight. No two engines in an airframe shall bear the same number. In twin engined rotorcraft, with one engine on each side of the plane of symmetry, the engines may be designated left and right as alternatives to 1 and 2.

7.2.4 It is desirable that all engine controls and indicators should follow as closely as possible the nomenclature of the engines and the pilot's thrust controls. It is recognised, however, that unified controls or indicators may be provided for example during V/STOL flight, and these may be related to some or all of the individual engines or groups of engines.

7.2.5 In the numbering of engines, auxiliary power units (APUs) which are, for instance, required for starting the engines, are not to be included in the count.

8 FUEL SYSTEM CONTROLS (see also Table 6)

8.1 GROUPING

8.1.1 All fuel system management controls, flow meter indicators, fuel gauges, fuel jettison and flight refuelling controls shall be, as far as possible, positioned on one panel and be superimposed on a fuel system diagram to give the pilot an indication of the state of the system. Suitable indicators shall be located adjacent to each selector or switch to show the position of the appropriate fuel cock (open/shut) in relation to the system diagram. The panel shall be located so that it is accessible to the first pilot's left hand[1] with harness locked and is easily visible.

8.2 FUEL COCK CONTROLS

8.2.1 Handwheels shall not be used to control the operation of fuel cocks. Where an "On/Off" fuel cock under the pilot's control is provided for jettisonable fuel tanks, the cock shall not be combined with the jettison or any other control. Any fuel cock control, other than the combined low/high pressure, cross-feed and fuel transfer cocks, shall register in two positions only, i.e., forward or up for on and rearward or down for fuel off.

8.3 DROP TANK RELEASE CONTROL (see also Table 6, Item 3)

8.3.1 The release switch shall control the release of all drop tanks in flight. The electrical circuit for this switch shall also be connected with the EXTERNAL ARMAMENT AND DROP TANKS JETTISON switch (see also Table 9, Item 4 and Chapter 105, para 11.3), if fitted, so that the drop tanks will be released also, in addition to the external armament, by operation of this latter switch.

8.4 IN-FLIGHT REFUELLING COCKPIT CONTROLS AND DISPLAYS[5]

8.4.1 The following controls and displays shall be provided for in-flight refuelling operations in receiver rotorcraft.

(i) Controls (see also Table 6 Item 6):

(a) Air refuel selector.

(b) Reset button (where applicable).

(c) Disconnect button (where applicable).

(ii) Display:

(a) Tanks full indication.

(b) Disconnect indication (Master caution) (where applicable).

9 ELECTRICAL CONTROLS (see also Table 7)

9.1 GROUPING

9.1.1 The following controls shall be grouped together:

(i) AC and DC supply switches, but not the master electrical switch,

(ii) all switches and rheostats associated with instrument and cabin lighting, except the emergency lighting switch,

(iii) all switches and rheostats for miscellaneous items such as defrosting, deicing, cockpit air conditioning and windscreen wiping, and

(iv) any other switches and controls specifically for electrical services.

10 RADIO CONTROLS (see also Table 8)[7]

10.1 On radio frequency cards, pre-set channels shall be displayed in ascending order of frequency.

11 ARMAMENT CONTROLS (see also Table 9)

11.1 GROUPING
11.1.1 In general, armament selectors shall be grouped functionally and shall be easily accessible to the pilot with harness locked.

11.2 MASTER ARMAMENT SWITCH
11.2.1 When the armament selectors are at any other crew station, a guarded master armament switch shall be provided under the control of the pilot. Consideration shall be given to the provision of an indicator, visible from outside the rotorcraft, to show when the installation is safe.

12 FLIGHT INSTRUMENTS (see also Table 10)

12.1 LAYOUT[9]
12.1.1 The layout of flight instruments shall conform to the pattern shown in Fig.1 below. This arrangement applies to conventional instruments and to flight data in electronic displays. Deviations from this pattern may be permitted, after agreement with the Rotorcraft Project Director, where it is necessary to include special displays e.g., radar and television scopes, large integrated displays etc., on the pilot's instrument panel.[9]

AIR SPEED	HEADING (2)	ALTITUDE (3)
	ATTITUDE	
ANGLE OF ATTACK (1)	HEADING (2)	VERTICAL SPEED

NOTES:
(1) Angle of attack if required.
(2) Heading may be located at the top or bottom of the display.
(3) Altitude may be either barometric, radar, or both.

FIG.1 LAYOUT OF FLIGHT DATA IN PILOT'S DISPLAYS

12.2 POWER FAILURE INDICATION IN FLIGHT INSTRUMENTS
12.2.1 All electrically driven flight instruments shall have an indication of power failure incorporated. The indication shall be such that the instrument cannot be used in the event of power failure.

13 ENGINE INSTRUMENTS[11]

13.1 LOCATION OF DEDICATED DISPLAYS
13.1.1 Single or Tandem. Displays shall be located to the left of the pilot's basic flight instrument group.

13.1.2 Side-by-side. Displays shall be located between the first and second pilot's flight instruments so that they are legible to both crew members.

13.2 LOCATION OF STANDBY DISPLAYS

13.2.1 In rotorcraft using standby engine displays in conjunction with a time-shared engine multifunction display, the standby display may be located in any area of the forward panel that is visible and legible to both crew members but preferably on the lower portion of the panel. The layout of a multifunction standby display shall mimic the layout of the primary display where possible.

13.3 ARRANGEMENT OF DISPLAYS

13.3.1 Vertical Grouping. The engine display parameters shall be arranged so that the primary or most important display for a particular engine and rotorcraft (thrust, torque, RPM, etc) be located at the top of the display group if a vertical grouping is provided. The next most important display parameter shall be positioned under the primary display progressing down the panel with the least important at the bottom.

13.3.2 Horizontal Grouping. If the engine displays are grouped in a horizontal format such as when vertical scale tape displays are used, the primary display shall be on the end of the group nearest the first pilot progressing to the least important at the further end.

13.3.3 Arrangement in Relation to Engines. The displays shall be arranged and numbered in relation to the rotorcraft engines (see para 7.2). The displays to the left of the group shall correspond to the left side engines and the displays on the right, to the right side engines.

13.3.4 Multifunction Displays. When engine parameters are shown on multifunction displays the guidelines outlined in paras 13.3.1, 13.3.2 and 13.3.3 shall apply. Multifunction displays shall also include a prime power display, eg, Rotor Torque or Power Turbine RPM display adjacent to the attitude display.

13.3.5 Scale Alignment and Marking. Where possible engine parameter display scales shall be aligned so that pointers on round dial displays are all at the 12 o'clock position or, the 9 o'clock position when normal conditions exist (not mixed). When vertical scale format displays are used the pointers or the tape readouts shall approximate a straight horizontal line across the display group when normal conditions exist. In all cases instrument scales shall be marked in accordance with DEF STAN 66-26 (part 7).

13.3.6 The fuel flow meter may be placed on a fuel panel subject to the Rotorcraft Project Director's agreement.

13.3.7 Other groupings may be used when required for operational reasons or to accommodate new technically advanced instrumentation subject to the Rotorcraft Project Director's agreement.

13.3.8 Any additional instruments demanded by a particular engine installation shall be discussed with the Rotorcraft Project Director before being included in the main instrument grouping.

13.3.9 Notwithstanding the requirements of paras 13.3.1 to 13.3.8, contractors may propose alternative arrangements of engine displays to enhance data transfer from display to crew. The Rotorcraft Project Director may permit deviations from the requirements when the alternative arrangements proposed materially enhance operational capability and/or flight safety.

14 WARNING, CAUTIONARY AND ADVISORY SIGNALS (see also Chapter 105, paras 12 and 13)

14.1 GENERAL

14.1.1 The pilot shall be provided with a system of warning, cautionary and advisory signals (as defined in Chapter 105, para 12) in accordance with the requirements of paras 14.2 to 14.4 inclusive, unless otherwise agreed by the Rotorcraft Project Director.

14.1.2 The system of signals at other than the first pilot's station shall be agreed with the Rotorcraft Project Director at an early stage of the design.

14.1.3 All magnetic indicators used in the system shall conform to Chapter 105, para 13.6.

14.2 WARNING SIGNALS

14.2.1 A Standard Warning System in accordance with the following requirements shall be fitted to give warning of each of the following emergencies, when applicable:

(i) fire (in conjunction with the lights in para 14.5),

(ii) oxygen failure,

(iii) generator channel failure,

(iv) powered flying control system failure,

(v) loss of engine, coupling gearbox or transmission oil pressure, and

(vi) any folding component not locked in the flight condition.

No other warnings shall be incorporated in the system without prior agreement by the Rotorcraft Project Director.

14.2.2 The system shall consist of red flashing attention lights, a master audio warning signal and a central panel which indicates the particular emergency. The system shall be designed in accordance with Specification EL 1960 (see however para 14.2.3).

14.2.3 The flashing attention lights shall be placed just below the bottom edge of the windscreen, one on either side about 230 mm from the centre of the pilot's line of sight when he is looking straight ahead. Two lights only shall be used in rotorcraft with only one pilot's station, or where the pilots sit in tandem; three lights shall be used where the pilots are seated side-by-side.

14.3 CAUTIONARY SIGNALS

14.3.1 An amber, resettable Master Caution Light situated on a central panel together with, and operating in parallel with, selected indicators, shall be fitted to denote secondary types of malfunction or failure.

14.3.2 It shall be agreed at the Mock-up Conference which cautionary indicators shall be grouped on the central panel. As a guide, the secondary types of malfunction or failure to be indicated should be those which do not involve the risk of immediate catastrophe, but which if ignored indefinitely, will lead to loss of, or damage to, the rotorcraft (see definition of cautionary signals given in Chapter 105, para 12.1.3(ii)).

Note: In order to avoid complicated electrical circuits when only a small number of systems require inclusion in the cautionary category, individual amber lights, centrally grouped, may be provided, with the agreement of the Rotorcraft Project Director instead of the full system described in para 14.3. 1. Less important signals may, in this case, be denoted by magnetic indicators.

14.4 ADVISORY SIGNALS

14.4.1 Advisory signals should not normally be placed centrally, but should be situated adjacent to the control or instrument to which they apply. These include:

(i) undercarriage position indicator,

(ii) hold/haul down gear lights,

(iii) radio altimeter limit lights,

(iv) marker beacon lights,

(v) armament installation lights,

(vi) camera installation lights,

(vii) external cargo hook release,

(viii) parachute jump light,

(ix) intercommunication crew call light,

(x) sonar installation lights, and

(xi) tele-briefing lights.

14.5 FIRE INDICATOR LIGHTS

14.5.1 A red light for each fire zone (see Chapter 712) shall be provided at the pilot's station so that in the event of fire it will give continuous illumination until the fire is extinguished, when the light will switch automatically. The light shall be incorporated in the fire extinguisher button, which shall be located so that there is the least possibility of confusing it with any other light. A marking shall be incorporated in the light to indicate the fire zone to which it is connected. These markings shall be about 5mm high and appear translucent (as opposed to transparent) white on a red background.

14.5.2 The fire extinguisher button shall be given emergency markings in accordance with Chapter 103[8].

15 SECOND PILOT'S STATION

15.1 During the design stages of dual seat rotorcraft, consideration shall be given to the following points:

(i) for side-by-side pilot's stations, the controls and instruments provided for the first pilot shall in general be duplicated at the second pilot's station, except for those controls which can be used and those instruments which can be seen by each pilot while harnessed in his seat. (Guidance on the extent of this duplication will be found in the Tables at the end of the Chapter, where the requirements are stated for specific controls and instruments)

(ii) for tandem pilot's stations, the same layout of controls and switches shall be provided at each station as far as possible, and

(iii) for dual seat training versions of operational rotorcraft, the cockpit layout for the pilot under training shall conform as closely as possible to that of the operational type.

16 CROSS REFERENCES TO OTHER CHAPTERS

16.1 A number of requirements directly related to the pilot's station appear elsewhere in this publication and the most important of these are listed below (see also the Alphabetical Index).

Chapter	Para	Subject
101	3	Clear vision
	5	Air conditioning
102	-	Emergency escape
103	-	Operational colouring and markings
104	-	View and clear vision
105	-	Crew stations - General requirements
106	-	Pilot's cockpit - Layout
111	-	Restraint and parachute harness for aircrew
700		Engine Controls
711	2.5	Air Data Sensors - indication of heater serviceability

REFERENCES

Reference	ASCC Air Std	STANAG
1	10/20	3225
2	10/22	-
3	10/16	3221
4	10/30	3370
5	10/49	-
6	10/19	3224
7	10/21	3258
8	10/23	3341
9	10/11	3216
10	10/39	3593
11	10/11	3359
12	10/14	3219
13	10/13	3218

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TABLE 1

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
1	AUTOMATIC FLIGHT CONTROL SYSTEM[3]			
(a)	CONTROL PANEL Single or Tandem	On left hand side of the cockpit where control stick steering is employed	-	-
	Side-by-side	On the centre pedestal and accessible to both pilots.	-	-
(b)	DISENGAGE BUTTON (EMERGENCY DISCONNECT)	Lower portion of forward face of cyclic pitch control. In wheel controlled rotorcraft, the disengage button shall be located on the side of the wheel opposite the throttle hand.	Press to disengage	Shall have emergency markings if emergency disconnect is incorporated[8]
(c)	PARTIAL DISENGAGE CONTROL	On collective pitch control.	-	-
2	POWERED FLYING CONTROL SELECTORS			
	Single or tandem	On collective pitch control, but if impracticable, should be accessible to pilot's left hand with harness locked.	Forward or up for POWER ON or for PRIMARY POWER ON.	Warning of control failure shall be provided on the Standard Warning System (see para 14.2).
	Side-by-side	On collective pitch control, but if impracticable, should be accessible to both pilot's with harness locked.		

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
3 (a)	TRIMMING CONTROLS[1] (PRIMARY FLIGHT CONTROLS)	Manual: Wheel(s) or Knob(s) easily accessible to each pilot's left hand with shoulder harness in place and locked.	This shall be in direct relation to the desired motion of the rotorcraft.	-
		Electrical: An operating switch located on the cyclic pitch control(s) for cyclic control trim; on the collective pitch lever(s) for collective pitch trim; and within easy reach of each pilot for other trim controls. Both cyclic and collective pitch trim switches shall be so located that they are easily operated by the thumb of each pilot.	This shall be in direct relation to the desired motion of the rotorcraft.	-
(b)	DIRECTIONAL TRIM			
	Single or tandem	Operable by pilot's left hand.	) As for pitch and roll trim.	-
	Side-by-side	On centre console, accessible to first to first pilot with harness locked.	)	
(c)	RELEASE BUTTON	Middle of cyclic pitch control, left hand side	Press to release.	-

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
4	UNDERCARRIAGE			
(a)	NORMAL CONTROL[1] Single or tandem	For single pilot operation: there shall be a single control mounted to the left of the pilot adjacent to collective lever grip. Within easy reach of the pilot with shoulder harness in place and locked (or armed). For tandem operation: similar but each pilot shall be provided with a control.	Lever control: shall operate upwards to retract and downwards to extend.	Lever control handle shape shall be in accordance with DEF STAN 16-24.
	Side-by-Side	Control shall be mounted on the centre console adjacent to the first pilot's collective lever grip. Within easy reach of first pilot with his shoulder harness in place and locked (or armed) and within reach of the other pilot.	Push button operation: top button for under-carriage UP, bottom button for undercarriage down	
(b)	STANDBY LOWERING[1]	Lever control shall be located adjacent to the normal control.	Lever control shall operate in the same sense.	-
		Third button integral with normal control.	-	-
(c)	POSITION INDICATOR (see also para 2.8)	Adjacent to undercarriage control visible to each pilot.	-	-

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
5	AIR BRAKES[1]	Any control which operates fuselage or aerodynamic surface mounted airbrakes or spoilers shall be located in a suitable position on the pilot's collective pitch lever. Where there is two pilot operation, a control shall be provided at both pilot stations.	The direction of operation shall be aft for airbrakes out or spoilers open and forward for airbrakes or spoilers closed.	-
6	INTERNAL FLIGHT CONTROL LOCKS[1]	Adjacent to the engine and/or rotor speed controls located at first pilot's station only.	-	The design shall be that it shall be impossible to obtain take-off power with locks engaged.
7	HYDRAULIC SYSTEM CONTROLS[1]	All hydraulic system selectors which may control hydraulic supplies to the primary or supplementary flying controls shall be mounted such that they may be operated by the pilot's left hand. For two pilots in seated tandem, the control shall be mounted as above and provided at both pilot stations. For two pilots seated side by side the control(s) shall be accessible to both pilots	-	-

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
8	ROTOR BRAKE CONTROL[1] Single or Tandem	Depending on cockpit configuration, this shall be located on the left of the pilot, preferably in the overhead and/or adjacent to the engine and rotor speed controls, within easy reach of the pilot's left hand with the shoulder harness in place and locked. For two pilots seated in tandem: as specified above but shall be located on the left of each pilot	-	Design of the controls shall be such that inadvertent operation of the controls shall not be possible. A safety device and/or warning system (alarm) shall be incorporated in this system to guard against engagement of the rotor when the system is locked.
	Side-by-side	As specified above but located within easy reach of the first pilot with shoulder harness in place and locked and within reach of the other pilot.	-	

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
9	WINCH CONTROLS[1] Single pilot operation	The controls shall be located on the cyclic or collective stick grip but may also be on console when an automatic hover facility is provided.	) On rotorcraft where the winch is not permanently installed on the outer structure, moving a pivoted switch right or left shall extend or retract the winch. Moving a pivoted switch up or down (or back/forward) shall retract or extend the winch cable.	(i) a "winch cable cut" device should provided for both the winch operator and the pilot(s). For the winch operator this device should be within easy reach and not require him to take his eyes off the winch loop/hook/basket to operate the device; for the pilot(s) the device should be located on each cyclic or collective stick grip.
	Pilot and winch operator in tandem	With minimum crew of one pilot plus one winch operator in tandem, the controls shall be located as above.	)	
	Two pilots seated side by side	Controls shall be located on each cyclic or collective stick grip or on the console.	)	(ii) The pilot(s) controls should feature override facilities for winch operator controls.
	Pilot in cockpit and Winch operator in Cargo Hold or in Rear Cabin	The controls shall be located at the rear winch operator's station and the pilot(s) is (are) provided with a winch override control and a winch cable cut device. A second winch control may be provided for the pilot(s), the primary winch override switch becoming a flight station selector.	)	(iii) Winch control may permit variation of winch cable speed.

TABLE 1 - continued

FLYING CONTROLS (see also para 2)

Item No	Service	Location	Actuation	Remarks
10	FLOAT CONTROLS[1] (Emergency Float or Similar Devices)	Float controls (release button or switch) shall be provided on each cyclic or collective stick grip or on the console regardless of pilot seat arrangements. The float system is to be protected by a circuit breaker located within reach of the left hand of the first pilot.	With the circuit breaker closed, the floats will be activated directly through the immersion switches or the float control button (or switch).	When immersion switches are fitted the float control may be on the console.

TABLE 2

GROUND MANOEUVRING CONTROLS (see also para 4)

Item No	Service	Location	Actuation	Remarks
1	PARKING BRAKE/WHEEL LOCK CONTROLS[1] Single or tandem Side-by-side	 Located within easy reach of the pilot(s). Located within easy reach of the first pilot and also within reach of the other pilot.	Operation of either control shall not require movement of the pilot(s) right hand from the cyclic cyclic pitch control.	-
2	WHEEL BRAKE CONTROL (see also para 4.1)	A brake pedal on directional control pedal[1] at first pilot's station and if decided at Mock-up Conference at second pilot's station also.	Toe operation [1].	-
3	WHEEL CASTORING LOCK CONTROL	Within reach of first pilots left hand and shall be easily operated with shoulder harness in place and locked (or armed)[1].	Switch or lever, forward or up for LOCK.	-
4	STEERING CONTROL[1]	Undercarriage steering should be controlled through the rudder pedals and a locking system shall be provided. The control shall be within easy reach of the first pilot with his shoulder harness in place and locked (or armed).	Indication of steering system engagement shall be provided for the pilot(s).	-
5	WHEEL DRIVE SPEED CONTROL[1]	Where internally supplied power can be directed to the landing gear wheel units, to provide a ground taxying capability with rotors stopped, the controls for this power shall be located within reach of the first pilot when seated. The drive controls, shall not reduce the external view of any crew member from that available when taxying with rotors rotors turning.	- -	- -

TABLE 3

ENVIRONMENT AND ESCAPE CONTROLS (see also para 5)

Item No	Service	Location	Actuation	Remarks
1	ABANDON AIRCRAFT COMMAND VISUAL SIGNAL CONTROL (see also Chapter 105, para 12.4)			
	All multi-seat rotorcraft	At first pilot's station, operable by left hand with harness locked[4].	2-Position guarded switch[4]. Labelled ABANDON AIRCRAFT. Forward or on position labelled ABANDON.	Shall have emergency markings[8].
2	AIRFRAME DE-ICING CONTROLS	Grouped together and accessible to first pilot's left hand.	-	-
3	CABIN CONDITIONING SELECTOR	Accessible first pilot	-	-
4	EMERGENCY FLOTATION INFLATION CONTROL	Easily accessible to both pilots, with harness locked.	-	Shall have emergency markings.
5	SAFETY HARNESS LOCKING CONTROL	Convenient to pilot's left hand.	2-Position lever UNLOCKED forward and LOCK rearward.	The pilot(s) shall be safe-guarded by an inertia lock or a device giving similar protection
6	SEAT ADJUSTMENT CONTROL			
(a)	VERTICAL ADJUSTMENT	On right hand side of seat, or for side-by-side seats when access to seats is restricted, on out-board side of each seat.	Mechanical control - lever operating in natural sense. Electrical control - 3-position switch, operating in natural sense, with centre off position.	-
(b)	FORE-AND-AFT ADJUSTMENT (see also Chapter 106 para 6.2.2)	Adjacent to vertical adjustment control.	Forward to unlock.	-

TABLE 4
CONTROLS AND INDICATORS FOR SHIPBORNE OPERATION (see also para 6)

Item No	Service	Location	Actuation	Remarks
1	HOLD/HAUL DOWN			
(a)	CONTROL	On collective pitch control.	To be agreed at Mock-up Conference.	-
(b)	INDICATOR LIGHT	Visible to each pilot by day and by night.	To be agreed at Mock-up Conference.	-
2	ROTOR BLADES FOLD AND SPREAD			
(a)	CONTROL	To be agreed at Mock-up Conference.	A minimum of two controls, one or more to lock and unlock and one or more to actuate the system with an interlock to ensure correct sequence of operation.	(i) A safety device to be fitted to to prevent inadvertent operation. (ii) A similar system with separate controls to be provided for any other folding part.
(b)	INDICATOR	At rotor head, visible to pilot by day and by night.	Mechanical indicator to give warning when blades not locked in flight position. Indicator shall fail safe.	(i) Unsafe condition shall also be indicated on Standard Warning System (see para 14.2). (ii) Indicator shall derive its information from a different source to that of the Standard Warning System. (iii) A similar system with separate indicator to be provided for any other folding part.

TABLE 5

ENGINE POWER CONTROLS - TURBINE ENGINED ROTORCRAFT (see also para 7)

Item No	Service	Location	Actuation	Remarks
1	DRIVE SELECTOR Single or tandem Side-by-side	 Accessible to pilot(s). Accessible to both pilots.	)) Forward or up for) MAIN DRIVE.	(i) A safety device shall be fitted to prevent inadvert-ent operation. (ii) A visual indicator shall be provided to show the position of the drive actuator. (iii) In tandem seat case, a separate control for the second pilot may not be necessary.
2	ENGINE CONDITION CONTROL Single or tandem Side-by-side	 Within comfortable reach of pilot's left hand with harness locked. Accessible to both pilots, with harness locked.	))) Forward or up to increase) RPM))	(i) A safety device shall be fitted to prevent inadvert-ent operation. (ii) If used as a rotor speed select control, it shall incorporate some form of fine trimming device. (iii) In tandem seat case, a separate control for the second pilot may not be necessary.
3	ROTOR SPEED GOVERNOR CONTROL Single or tandem	 Accessible to pilot(s) left hand with harness locked.		(i) In tandem seat case, a separate control for the second pilot may not be necessary.

TABLE 5 - continued

ENGINE POWER CONTROLS - TURBINE ENGINED ROTORCRAFT (see also para 7)

Item No	Service	Location	Actuation	Remarks
3 cont	Side-by-side	Accessible to both pilots, with first pilot's harness locked.		
4	ROTOR SPEED SELECT CONTROL Single or tandem Side-by-side	 Within comfortable reach of pilot's left hand, with harness locked Accessible to both pilots, with harness locked.	) Forward, up or clockwise to increase RPM.	(i) When an electrical control is used, it should be in the form of an inching switch, spring loaded to the neutral position. (ii) If the control is located on the collective pitch control, it shall be in such a position that the pilot can operate it without moving his hand from the throttle twist grip. (iii) In tandem seat case, a separate control for the second pilot may not be necessary.
5	STARTING CONTROL-GROUND OR IN FLIGHT	Within reach of first pilot's left hand, with harness locked.	Forward, up or clockwise for ON	A push button control may be used.
6 (a)	THROTTLES (where applicable) MAIN	 On collective pitch control.	 Twist grip; clockwise, when viewed from free end of collective pitch control, to increase power.	 (i) Duplicated controls shall be interconnec-ted. (ii) The relationship between throttle angle and collective pitch to be decided at Mock-up Conference.

TABLE 5 - continued

ENGINE POWER CONTROLS - TURBINE ENGINED ROTORCRAFT (see also para 7)

Item No	Service	Location	Actuation	Remarks
6 (b)	ADDITIONAL OR EMERGENCY			
	Single or tandem	On a console by pilot's left hand, with harness locked.	))) Lever, forward to increase power.	(i) In tandem seat case, a separate control for the second pilot may not be necessary.
	Side-by-side	On console by first pilot's left hand, accessible to both pilots with harness locked.	))	
7	THROTTLE AUTO/ MANUAL SELECTOR	On collective pitch control in such a position that pilot can operate it without moving his hand from the throttle twist grip.	-	A positive indication shall be provided to show when the throttle is in the manual mode.
8	THROTTLE, FRICTION CONTROL			
(a)	MAIN THROTTLE	At rear end of throttle twist grip.	Knurled ring; rotate clockwise when viewed from free end of collective pitch control, to increase friction.	-
(b)	ADDITIONAL OR EMERGENCY THROTTLES	Adjacent to throttle lever.	Knurled knob; rotate clockwise to tighten.	-

TABLE 6

FUEL SYSTEM CONTROLS (see also para 8)

Item No	Service	Location	Actuation	Remarks
1	BOOSTER PUMP CONTROLS	On fuel system panel (see para 8.1).	In, forward or up for ON.	-
2	CROSS FEED AND TRANSFER COCKS	On fuel system panel (see para 8.1).	In the natural sense, according to disposition of tanks.	-
3	DROP TANKS RELEASE CONTROL (see also para 8.3)	Within easy reach of first pilot's left hand, with harness locked.	2 position guarded switch, forward or up for SAFE, aft or down for RELEASE. Alternatively, recessed pushbutton, push for RELEASE.	Control shall be labelled DROP TANKS/RELEASE.
4	FUEL JETTISON CONTROLS	On fuel system panel (see para 8.1)	Double action control, pull and twist to jettison.	-
5	HIGH AND LOW PRESSURE COCKS	On fuel system panel (see para 8.1).	Forward or up for fuel ON.	A safety device shall be fitted to prevent inadvertent operation.
6	FLIGHT REFUELLING CONTROLS			
	(a) Selection/Reset Controls	In single or tandem seat rotorcraft, operated by the left hand. In multi-seat rotorcraft, near to and operated by the second crew member or flight engineer.	-	-
	(b) Disconnect Control	In single or tandem seat rotorcraft, on the control column and operated with the right hand. In multi seat rotorcraft, on the control column and operated by the pilot.	-	-

TABLE 7

ELECTRICAL CONTROLS - (see also para 9)

Item No	Service	Location	Actuation	Remarks
1	AC AND DC SUPPLY CONTROLS	In electrical group (see para 9.1).	Forward or up for ON. Switches for generator/ actuator/TRU to be guarded.	Warning of supply failure shall be provided on the Standard Warning System (see para 14.2)
2 AL5	ANTI-DAZZLE LIGHTING SWITCHES (see also Chapter 105, para 15.9)	In electrical group (see para 9.1), in such a position that pilot(s) can find and operate it quickly and easily when blind and with harness locked.	3-position guarded switch. Forward or up for BRIGHT, centre for OFF, aft or down for DIM[6]	Switch shall be labelled ANTI-DAZZLE[8].
3	MASTER ELECTRICAL SWITCHES[12]		-	-
	Single or tandem	Together and easily accessible to the left hand.		
	Side-by-side	Together and, if possible, accessible to both pilots and on the overhead panel, if fitted.	-	-
4	EMERGENCY LIGHTING SWITCH	On the lower edge of the main instrument panel in a position immediately accessible to pilot's left hand.	Forward or up for ON.	See also Chapter 103, para 2.3
5	LANDING LAMP SWITCHES			
(a)	BEAM ANGLE CONTROL SWITCH	On collective pitch control, in such a position that the pilot can operate it without moving his grip on the collective pitch control.	Forward for DOWN, and aft for UP.	When an azimuth control is fitted, it should operate in the natural sense using the same switch.
(b)	LANDING LAMP SWITCHES	On collective pitch control.	Forward or up for ON.	-
(c)	INTENSITY CONTROL	Combined with ON/OFF switch.	-	-

TABLE 7 - continued

ELECTRICAL CONTROLS - (see also para 9)

Item No	Service	Location	Actuation	Remarks
6	LIGHTING, DE-ICING, ANTI-ICING, PITOT HEAD AND ENVIRONMENTAL CONTROL SWITCHES [12]			
	Single or tandem	Functionally grouped together on the aft end of the left console.	-	-
	Side-by-side	Functionally grouped together, and, if possible, accessible to both pilots, and on the overhead panel, if fitted.	-	-

TABLE 8

RADIO CONTROLS - (see also para 10)

Item No	Service	Location	Actuation	Remarks
1	COMMUNICATION, NAVIGATION IDENTIFICATION AID CONTROLS			
(a)	CONTROLS USED DURING CRITICAL MODES OF FLIGHT, INCLUDING TAKE-OFF AND LANDING	Operable by pilot with minimum interference with his primary task. Within easy reach with harness locked.	Any radio change-over switch shall operate in natural sense.	A remote channel and/or frequency indicator may be used, located in pilot's forward field of view but not below the level of the primary flight instruments.
(b)	CONTROLS NOT USED DURING CRITICAL MODES OF FLIGHT			
	Single or tandem	On the right console		
	Side-by-side	On centre console or overhead panel accessible to both pilots	Any radio change-over switch shall operate in natural sense.	A remote channel and/or frequency indicator may be used, located in pilot's forward field of view but not below the level of the primary flight instruments.
2	MUTING SERVICE	1. On cyclic pitch control or alternatively on collective pitch controls AND 2. In cockpit location other than on cyclic or collective pitch controls.	Press to MUTE.	- Location to be decided at mock-up conference, but shall be easily accessible during critical modes of flight with harness locked.
3	PRESS-TO-TRANSMIT /ICS SWITCHES	1. On cyclic pitch control AND 2. In cockpit Location other than on cyclic or collective pitch controls.	Press to TRANSMIT	Location to be decided at mock-up conference, but shall be easily accessible during critical modes of flight with harness locked.
4	RADIO ALTIMETER SELECTOR	Preferably on or adjacent to the radio altimeter indicator.	-	-

TABLE 9

ARMAMENT CONTROLS - (see also para 11)

Item No	Service	Location	Actuation	Remarks
1	ARMAMENT RELEASE/ FIRING TRIGGER	Top forward face of cyclic pitch control[2].	By current standard armament release trigger.	Trigger shall also operate cine camera, if fitted.
2	BOMB DOORS			
(a)	CONTROL	On armament panel (see para 11.1).	2 or 3-position control. For 3-position control, forward or up for OPEN, centre CLOSED, aft or down for AUTO.	-
(b)	POSITION INDICATOR	On armament panel (see para 11.1) adjacent to bomb door control.	3-Position magnetic indicator, labelled BOMB DOORS POSITION, showing black for CLOSED, striped for TRANSIT and white with black 0 for OPEN.	-
3	CINE CAMERA CONTROL	On cyclic pitch control[2].	Push button - push and hold to actuate.	-
4	EXTERNAL ARMAMENT AND DROP TANK JETTISON CONTROL			
	Single	(i) On collective pitch control, or	For (i); guarded push button. For (ii); recessed push button, unguarded. Push to jettison. Labelled EXTERNAL ARMAMENT AND DROP TANK JETTISON.	Shall have emergency MARKINGS[8].
	Tandem	(ii) Within easy reach of first pilot's left hand with shoulder harness in place and locked, while maintaining normal vision from the rotorcraft.[1] As for 'Single' but at each pilot station[1]		
	Side-by-side	As for 'Single' but within easy reach of the first pilot, with shoulder harness in place and locked, and within reach of the other pilot.[1]		
5	INTERNAL ARMAMENT JETTISON CONTROL	On armament panel (see para 11.1)	3-position guarded switch. Forward or up for JETTISON, centre for OFF, aft or down for OVERRIDE. Labelled INTERNAL ARMAMENT JETTISON.	(i) Shall control operation of bomb door and jettisoning bombs in sequence. (ii) Shall have emergency emergency markings.

TABLE 10

FLIGHT INSTRUMENT CONTROLS - (see also para 12)

Item No	Service	Location	Actuation	Remarks
1	STAND-BY ARTIFICIAL HORIZON POWER SELECTOR SWITCH	Preferably adjacent to artificial horizon.	Labelled STAND-BY ARTIFICIAL HORIZON SUPPLY. NORMAL AND STAND-BY.	) Consideration shall be given to the classification of these switches as
2	TURN AND SLIP STAND-BY POWER SWITCH	Adjacent to each turn and slip indicator	Labelled T AND S SUPPLY. NORMAL AND STAND-BY.	) emergency controls with emergency markings.

CHAPTER 108

INTERNAL NOISE

1 INTRODUCTION

1.1 The requirements of this Chapter are concerned with the protection of aircrew passengers from exposure to internal noise, presented in the form of design aims, specifications and evaluation procedures.

1.2 Whilst the levels themselves are not mandatory, the rotorcraft designer shall be able to demonstrate to the Procurement Agency that noise control has been given proper consideration throughout all phases of helicopter design and development. This demonstration shall be in accordance with the environmental management plan set out in Leaflet 501/ 2. Every effort shall be made to ensure that noise levels are kept to a minimum consistent with other design requirements placed upon the helicopter design.

1.3 The design aim shall be for the total noise levels at the ear not to exceed an 8 hour L_{eq} (equivalent continuous noise level) of 85 dB(A) in order to provide adequate protection against permanent hearing losses, and, if these levels are achieved other factors such as intelligibility and some aspects of auditory monitoring will generally be acceptable.

1.4 The criteria are divided into two parts:

(i) A noise environment for aircrew protected by an aircrew helmet.

(ii) A noise environment for unprotected passengers.

2 CRITERION (i) A NOISE ENVIRONMENT FOR AIRCREW PROTECTED BY AN AIRCREW HELMET

2.1 Aircrew should be protected by the Mk 4 series flying helmet, with this level of protection being provided for 97% of the aircrew population (the helmet attenuation is defined in the last column of Table 1, Leaflet 108/2). Should alternative helmets be used, they must have equal or better acoustic attenuation and variance that is not statistically significantly different from that specified in Table 1, when assessed by the same semi-objective method.

2.2 With the Mk 4 helmet in use the following aspects are taken into account to provide specification levels.

2.2.1 A minimum Articulation Index (AI) of 0.5 shall be attained using the whole communication system fitted to the production standard aircraft. The AI shall be calculated from measurements made at the aircrew's ear using a Mk 4 helmet. This shall be calculated in accordance with ANSI S.3.5.1969 (Rl973) and shall properly take into account the limitations on testing of systems noted in this document. The 20 band method shall be used. An AI of 0.5 should allow a minimum speech intelligibility of 95% in rotorcraft when using military jargon and phrases.

2.2.2 Auditory monitoring tasks such as passive and active sonics returns, electronic warfare returns in the form of derived and "true audio" spectra and auditory warning signals have been taken into account in deriving this specification. However, there are certain incompatibilities between helicopters noise spectra and auditory listening tasks and these are further discussed in Leaflet 108/1 para 4.

2.2.3 Long term hearing damage risk, based on an L_{eq} of 85 dB(A) is calculated for a 4 hour flight with an allowance for the additional risk caused by communications and auditory monitoring levels.

2.2.4 At frequencies below 25 Hz, at sufficiently high levels, non-auditory aspects of noise can affect crew performance. Measurements should be taken as described in para 4.2 and the results discussed with the Procurement Authority.

3 CRITERION (ii) A NOISE ENVIRONMENT FOR UNPROTECTED PASSENGERS AND CREW

3.1 The environment is based on a flight of 2 hour maximum length with an 8 hour L_{eq} of 85 dB(A) and the ability to carry out unaided special communications. The following aspects are relevant to this particular part of the specification.

3.2 The specified levels shown are for just adequate non-aided voice communication over a distance of 2 feet (0.62m) with a shouted voice. The Speech Interference Level (SIL) calculated in accordance with ANSI S.3.14-1977 shall not exceed 80 dB(A), and if this criteria is met, 95% sentence intelligibility is probable.

3.3 Temporary Threshold Shift (TTS) should not be excessive and this specification is intended to restrict TTS levels to between 5 and 10 dB(A)

3.4 The criterion for prevention of long term hearing impairment is identical to the protected aircrew criterion, which is an L_{eq} Of 85 dB(A) without the allowances for communications and auditory monitoring.

3.5 The effects of non-auditory aspects of noise shall be treated as stated in para 2.2.4.

4 SPECIFICATION LEVELS

4.1 The following Table sets out the design aims of cabin noise level in each octave band and is applicable to each rotorcraft.

Condition	OCTAVE BAND CENTRE FREQUENCY (Hz)									
	8 16	31.5	63	125	250	500	1K	2K	4K	8K
(a)dB(A)	See para 2.2.4 and 4.2	111	101	91	87	84	82	81	83	82
(b)dB(A)		117	107	97	90	84	81	80	75	75

Condition (a) is for protected aircrew.
Condition (b) is for unprotected crew and passengers.

4.2 The lowest frequency band that shall be measured shall be related to the actual rotorcraft main rotor blade passing frequency, nR. Where 'nR' is below the 31.5 Hz octave band lower limiting frequency (22.4 Hz), then measurements shall be taken down to a frequency which allows the levels of 'nR' to be accurately defined ('nR' is defined as the number of rotor blades (n) multiplied by normal operating rotor rpm (R) and calculated in Hz.)

4.3 When applying the 'trade-offs' described in Leaflet 108/1 para 3 the sum of the 5 bands from 31.5 Hz to 500 Hz inclusive shall not exceed 87 dB(A) for condition (a) para 4.1 or 89 dB(A) for condition (b) para 4.1, and the individual levels of these 5 bands should not be exceeded by more than 5 dB(A) SPL.

5 MEASUREMENT OF ROTORCRAFT INTERIOR SOUND PRESSURE LEVELS

5.1 The measurement procedures described shall be used to ensure uniformity of acoustical measurement from one helicopter to another and from one measurement system to another.

5.2 Cabin noise measurements shall be made at:

(i) All crew member stations.

(ii) A representative number of passenger positions (seated);

(One position per every three passengers shall be measured with measurement points to be evenly spread over the passenger area. Where conditions are such that noise levels may vary significantly from seat to seat, the maximum noise levels shall be recorded and reported along with their position in the rotorcraft).

5.3 Measurements shall be made with passengers absent and the minimum number of crew members present. The microphone shall be hand held vertically in a fixed position, with the diaphragm facing upward at seated head height. Care shall be taken to minimise the effects of the sound field caused by the person holding the microphone and the presence of the crew member. A description of the technique used, if different from that described above, shall be reported.

5.4 OPERATING CONDITIONS

5.4.1 Flight conditions:

(i) Stabilised hover in ground effect (IGE).

(ii) Level flight at 0.9 V_{no} in temperate conditions at low altitude, (V_{no} is maximum level flight speed).

Measurements shall be made at within ±10% of maximum all-up weight at normal operational rotor speed.

5.4.2 Cabin configuration
Doors, windows and vents shall be closed. Acoustic treatments applicable to the production rotorcraft shall be intact and the interior fully furnished as per the production standard rotorcraft (where applicable). Where normal operation involves the operation of the rotorcraft with door(s) open, a measurement shall be made in this condition, at the operators position and reported.

5.4.3 Subsystems/equipment. All subsystems and equipment which are normally operated in flight for more than 5 minutes per hour, shall be operated during the acoustic measurements. This will include conditioning and clear vision systems.

5.5 Data recordings shall be made only when the conditions required have been obtained and stabilised and should have a minimum duration commensurate with the requirements of the subsequent analysis, particularly where low frequency components are being measured (typically 40 seconds minimum). Wherever possible, the recordings should commence with a statement of conditions or test serial number or, alternatively, some form of event marking shall be used, to enable the start and duration of the recording to be subsequently identified. The following flight data shall be observed and noted during the acoustic measurements, at intervals commensurate with the length of flight and the flight condition requirements:

(i) Take-off gross weight.

(ii) Fuel burn off.

(iii) Indicated air speed.

(iv) Rotor speed.

(v) Engine speed and power/torque settings.

Care shall be taken that oversaturation or overrecording of the recording system or medium shall not take place.

5.6 INSTRUMENTATION.

5.6.1 All test instrumentation and procedures shall conform to appropriate specifications and standards. Instrumentation shall be to precision standards and comply with the relevant sections of IEC651 Type 1 and/or ANSI S1.4 Type 1S.

5.6.2 The measurement system shall consist of a microphone which is fed through an interface unit used to convert the signal to match the characteristics of the recorder where the information is stored on magnetic tape or other suitable medium.

5.6.3 The microphone used shall have an adequate dynamic range to cover the sound pressure levels encountered and a frequency amplitude response which is linear over the frequency range of interest. The microphone shall be selected from a range of devices which have been developed for noise measurement purposes and whose transfer characteristics (sound pressure to voltage) have been related to approved standards. The microphone shall be fitted with random incidence correctors where necessary and windshields when used and the calibrations for these devices will also be required. The standard of performance of the microphones shall be verified by the results of a calibration within the previous 12 months and preferably by a check calibration prior to the commencement of the trials.

5.6.4 The recorder shall be calibrated to show the adequacy of dynamic range, frequency response, linearity and speed stability of the tape transport system.

Calibration of the total recording system in both amplitude and frequency shall be carried out to demonstrate frequency response, linearity, resolution, stability, noise floor and repeatability. In addition, measurements of the electrical noise floor of the total recording system shall be required to establish an adequate working range above this noise floor when the recordings are made.

5.6.5 The recording system performance shall not be degraded by vibration and electrical interference.

5.6.6 The details of the trial shall be recorded at the beginning of the tape for subsequent identification purposes.

5.6.7 Additionally, a constant amplitude sine wave of known amplitude and frequency shall be recorded at the beginning of each reel of tape used during testing, and this shall be relatable to the absolute sound pressure levels within the cabin.

5.7 The results of Data Reduction/Analysis obtained shall be presented in a manner such that the acoustic conditions within the rotorcraft at the various crew and passenger positions for the variations in flight conditions may be easily determined and compared with the relevant part of the specification. The recorded data shall provide the octave band, linear (unweighted) and 'A' weighted sound pressure levels for each of the measured locations and operating conditions. The time constants used for the integrating process shall be recorded and reported. Speech interference level shall be calculated, in the case of passenger carrying rotorcraft, by averaging the average SPL's in the 500, 1000, 2000 and 4000 Hz octave bands in accordance with ANSI S.3.14-1977.

5.8 Where relevant or appropriate, analysis in narrower frequency bands than octaves shall be shown. The analysis equipment shall be calibrated in a similar fashion to the noise measuring system and care must be taken to work within the linear range of the equipment and at least 10 dB above the noise floor.

LEAFLET 108/1

INTERNAL NOISE
INTERNAL NOISE LEVEL SPECIFICATIONS

1 INTRODUCTION

1.1 This leaflet sets out the design aims to be specified for maximum noise levels in rotorcraft cockpits and cabins for both rotorcraft and unprotected passengers.

2 NOISE LEVELS

2.1 The specification levels in Chapter 108 para 4 provide noise levels which are low enough to allow the aircrew to maintain effective and efficient performance throughout the sortie without undue noise interference with communications or auditory monitoring tasks, without undue risk of hearing impairment and mental or physical fatigue and without allowing the non-auditory effects to unduly interfere with their performance.

2.2 For unprotected passengers the specification provides for protection against long and short term hearing impairment and for the ability to communicate adequately without the support of aided communications systems.

2.3 The overall design aim is for the total noise levels at the ear not to exceed an 8 hour L_{eq} of 85 dB(A). For this single number criterion of 85 dB(A) L_{eq} an expansion can then be calculated in terms of octave band levels to provide a target specification for noise control purposes.

2.4 For the purposes of this document, the expansion has taken the form of a typical rotorcraft noise spectrum. However, this spectrum has been expanded on the basis of the current shapes of rotorcraft noise (i.e., high levels of low frequency noise rolling off rapidly as frequency increases). If, however, a rotorcraft is produced with a radically different cabin noise spectrum, then some adjustments to levels may be necessary between the 31.5 Hz and 500 Hz bands.

2.5 The use of octave bands is by no means ideal for helicopter noise, which comprises narrow band components, but the use of the narrower bands, such as one-third octaves, or less, complicates the derivation to such an extent that it becomes impracticable to provide a controlling noise specification.

3 ALLOWABLE TRADE-OFFS OF LEVELS OUTSIDE THE SPECIFIED BAND LEVELS

3.1 In the specification different octave band levels are controlled by different requirements. Where possible it is intended to allow certain trade-offs, of levels between octave bands, to allow a greater flexibility in meeting the overall specification, generally allowing a trading of the higher frequencies (e.g., gear noise) for the lower frequency bands (e.g., rotor noise), or vice versa.

3.2 The levels of frequency bands between, and including, the 31.5 Hz and 500 Hz bands are controlled by the limitations of long term hearing impairment. This is based on an 'A' weighted spectrum and, since this overall 'A' weighted level is a spectrum summed across all the relevant frequency bands, it is possible to obtain identical 'A' weighted levels with a variety of spectral shapes. Thus by trading between these 5 frequency bands it may well be possible for the rotorcraft to have a cabin noise spectrum which exceeds the levels in certain octave bands but still meets the total 'A' weighted criteria. (Chapter 108 para 4.3).

4 AUDITORY MONITORING

4.1 Where a rotorcraft is to be used for an auditory monitoring task or where auditory monitoring is considered an operational requirement during the sortie, the following aspects concerning auditory listening should be considered.

4.2 The use of octave band levels is inappropriate for analysis of cabin noise interference with an auditory monitoring task, due to the mechanism of auditory masking in humans. A rigorous consideration of the complexities of auditory monitoring has shown that the derivation of a fully satisfactory specification at low frequencies is impracticable. Consequently cabin noise levels below the 500 Hz band are considerably higher than would be permissible if auditory monitoring criteria alone were used. Thus, where detection or classification of low frequency signals is essential to the operational role, consideration should be given to both providing a more detailed analysis of the cabin noise than octave band levels and to alternative technologies, such as heterodyning, to transform the auditory signals to frequencies above 500 Hz.

LEAFLET 108/2

INTERNAL NOISE

Mk4 FLYING HELMET ATTENUATION

TABLE 1

The figures given were measured by semi-objective techniques
(Ref: RAE Tech Memo FS 171:1978).

1/3 Octave Centre Freq (Hz)	Mean Attenuation (dB)	Standard Deviation (dB)	Mean Minus 2 s.d. (dB)
<50*	-	-	-
50	5.9	2.4	1.1
63	4.9	1.7	1.5
80	5.2	1.7	1.8
100	6.0	1.6	2.4
125	6.2	1.7	2.8
160	6.3	2.0	2.3
200	6.4	2.1	2.2
250	8.3	2.8	2.7
315	11.1	3.0	5.1
400	22.5	3.3	15.9
500	27.0	3.1	20.8
630	27.0	3.7	19.6
800	32.2	4.4	23.4
1000	35.3	3.5	28.3
1250	29.5	3.1	23.3
1600	28.6	3.6	21.4
2000	35.3	4.4	26.7
2500	40.3	3.4	33.6
3150	40.3	3.6	33.1
4000	44.0	3.8	36.7
5000	43.0	3.2	36.6
6300	42.9	3.2	36.5
8000	45.1	3.7	37.7
10000	45.9	5.5	34.9

* Helmet attenuation is assumed to be zero below
the 50Hz one-third octave band.

CHAPTER 109

NORMAL ENTRANCE AND EXIT

1 INTRODUCTION

1.1 This chapter states the operational design and strength requirements for doors and hatches, which provide the normal entrances to a rotorcraft. Other aspects of the design which should also be considered are contained in the chapters listed in Table 1.

2 BASIC OPERATIONAL REQUIREMENTS

2.1 The doors and hatches shall be large enough to provide satisfactory entrance to and exit from the rotorcraft for large aircrew (as defined in Volume 1, Leaflet 105/3), wearing, the clothing and equipment detailed in the Aircrew Equipment Assembly Schedule appropriate to the particular rotorcraft.

2.2 Positive means shall be provided to retain the doors or hatches, in an open position, with rotors turning or stationary.

2.3 Means shall be provided to drain overboard any water which might run off doors or hatches, secured in the open position.

2.4 It shall be possible to operate the doors and hatches (including securing in and releasing from the open position) without undue difficulty from inside and outside the rotorcraft, and in all winds of up to 28 m/s (55 kn), from any direction with rotors turning or stationary.

2.5 No door shall be located with respect to any rotor disc or exhaust efflux so as to endanger persons leaving or entering the rotorcraft by day or night.

2.6 The design shall be such as to minimise the possibility of jamming of doors, hatches, external handgrips or steps due to any foreseeable cause. Particular attention shall be paid to the effect of ice or snow in cold weather or when operating on or near to snow covered ground and to the effect of rapid change, in ambient temperature. Protection shall also be provided against the abrasive effects of sand when operating in a sandy environment.

2.7 It shall be possible to enter and leave the rotorcraft without the use of ground support equipment, unless otherwise permitted by the Rotorcraft Specification.

2.8 The suitability of the entrance and the means provided to reach the entrance shall be demonstrated, to the satisfaction of the Rotorcraft Project Director, at the mock-up stage. This shall be demonstrated using aircrew approximating to both the large and small limits of size specified in Volume 1, Leaflet 105/3 and wearing the appropriate clothing and equipment, as in para 2.1.

2.9 Measures shall be taken to ensure against protrusions and sharp edges likely to damage aircrew survival equipment (e.g. immersion suits).

2.10 Provision shall be made to lock all entrance doors and hatches from the outside to prevent the entry of unauthorised persons and the means of locking shall be designed in such a way that deliberate action with a key is necessary to engage the lock. The doors shall be openable from the inside when locked.

3 ENTRANCE DOORS AND HATCHES

3.1 Means shall be provided for locking the entrance doors or hatches in the closed positions such that there is no possibility of a hazardous opening occurring during flight, either inadvertently or as the result of a single mechanical or electrical failure. If a crew member cannot see an entrance or check that it is correctly secured, a 'doors locked/unlocked' indicator shall be fitted in the cockpit.

3.2 Entrance doors and hatches which serve as emergency exits shall comply with the appropriate requirements of Chapter 102.

4 HANDGRIPS AND STEPS

4.1 External handgrips and steps may be provided to assist the aircrew. If these are designed to retract into or fold flush with the surface of the airframe, it shall be possible to operate them without the use of tools, and without any risk of grazing or jamming the hand or foot, even when cold weather clothing is worn. There shall be no risk of damage to aircrew clothing or equipment. Minimum dimensions of hand and footholds are given in SIS 6001. Air flow disturbance resulting from such holes shall not adversely affect the air conditioning or handling of the rotorcraft.

5 STRENGTH REQUIREMENTS

5.1 GROUND LOADS

5.1.1 Doors, hatches and associated structure shall have proof and ultimate factors of not less than 1.125 and 1.5 respectively, on the loads applied under the following conditions:

(i) The rotorcraft stationary with the doors, and hatches secured in any position in winds of up to 28 m/s (55 kn) from any direction with the rotors turning and with the rotors stationary.

(ii) With the doors, and hatches secured in any permissible combination of open and shut positions, for the landing and take-off cases in winds from ahead to 30° either side, up to the maximum normal service operating wind speed quoted in the Rotorcraft Specification.

5.2 FLIGHT LOADS

5.2.1 Doors and hatches shall have proof and ultimate factors of not less than 1.125 and 1.5 respectively, on the loads arising during any flight condition for which the rotorcraft is designed. All permissible combinations of doors and hatches, open or shut, shall be considered.

5.2.2 Measures shall be taken to minimise gaps occurring at the leading edges of doors, or hatches, when locked and under any flight condition likely to be encountered. Gaping which may occur elsewhere shall be such as to minimise stresses, buffet or vibration.

TABLE 1

LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	Leaflet	Para	Subject
100		7	Prevention of incorrect assembly
102		-	Emergency escape
	102/1	-	Ground test schedule for jettisonable doors and hatches
	102/2	-	Door and hatch locking mechanisms
103		-	Marking of doors, exits and canopies
104		-	View and clear vision
105		11.3.2	Emergency controls
106		7	Head clearance
407		-	Protection from weather
714		2.6.4	Paratroop doors
715		-	Transport panels

CHAPTER 110

NAVIGATION AND ANTI-COLLISION LIGHTS

1 INTRODUCTION

1.1 This chapter contains requirements for rotorcraft navigation and anti- collision lights based on an international agreement within NAT0[1].

AL/4

1.2 All military rotorcraft shall be provided with navigation light systems which will provide illumination completely around the normal plan of flight of the rotorcraft. The navigation light system shall include anti-collision lights as well as the side and tail lights or their equivalent unless otherwise agreed with the Rotorcraft Project Director. The coloured side and white tail lights or their equivalent, provide direction of flight information to pilots of other aircraft in the vicinity. The anti-collision light system provides a signal which generally permits rotorcraft to be seen at greater distances than rotorcraft provided only with the side and tail lights.

2 ANTI-COLLISION LIGHT SYSTEM

2.1 LOCATION

2.1.1 The anti-collision light(s) shall be located so that the emitted light shall not be detrimental to the crews' vision and will not detract from the conspicuity of the navigation lights.

2.2 COLOUR

2.2.1 Red and/or Aviation White (while operating).

2.3 FIELD OF COVERAGE

2.3.1 The system shall consist of such lights as will afford coverage of all areas around the rotorcraft. The field of coverage shall extend in all directions within 30° above and 30° below the horizontal plane of the rotorcraft, except that obstructed visibility totalling not more than 0.03 steradian shall be permissible within a solid angle of 0.15 steradian centred about the longitudinal axis in the rearward direction.

2.4 FLASHING CHARACTERISTICS

2.4.1 The arrangement of the system, such as the number of light sources, beam width, speed of rotation, etc., shall be such as to give an optimum flash frequency of 90 cycles per minute. The flash frequency for any single light source shall not be less than 40 cycles per minute. The flash frequency shall not be more than 100 cycles per minute except when the system includes overlaps created by more than one light source. In overlaps, the effective flash frequency shall not exceed 180 cycles per minute. The effective flash frequency is to be established as that frequency at which the rotorcraft's complete anti-collision light system is observed from a reasonable distance.

2.5 LIGHT INTENSITY

2.5.1 The minimum effective intensities in all vertical planes, measured with the red filter, shall be in accordance with Table 1. If a higher intensity is desired, a colourless glass may be used. In this case, the value of the effective intensity of the white light must be at least four (4) times higher than the minimum intensity of the red light (Table 1). The following relation shall be assumed:

where
$$I_e = \frac{\int_{t_1}^{t_2} I_t dt}{0.2 + (t_2 - t_1)}$$

I_e = effective intensity (candelas),and is the maximised value of the righthand side of this equation.

I_t = instantaneous intensity as a function of time.

$t_2 - t_1$ = flash duration (seconds).

Note: The maximum value of I_e is obtained when t_2 and t_1 are so chosen that the effective intensity is equal to the instantaneous intensity at t_2 and t_1.

TABLE 1

MINIMUM EFFECTIVE INTENSITIES FOR ANTI-COLLISION LIGHTS

Angle Above and Below Horizontal Plane	Effective Intensity Ie (Candelas)	
	Imin < 0.30 Imax	Imin > 0.30 Imax
0° to 5°	100	
5° to 10°	60	See sub-para 2.6
10° to 20°	20	
20° to 30°	10	

Imin = minimum intensity during "OFF" period.

Imax = maximum intensity during 'ON" period.

Note: It is desirable to increase the angle above and below the horizontal plane to 60°. Between 20° and 60° the minimum effective intensity shall be 10 candelas.

2.6 FLASH FREQUENCY v EFFECTIVE INTENSITY

2.6.1 The rise and decay characteristic of high current lamps flashed by electrical means are such that the intensity may not decay during, the "OFF" period to an acceptable level of less than 0.30 times the peak intensity. In such cases the flash frequency may be reduced to obtain an adequate decay provided that the effective light intensity (see para 2.5 above) is increased by twice the percentage of flash frequency reduction below 90 cycles per minute. As an example, if the flash frequency is 45 cycles per minute (a decrease from 90 cycles per minute of 5O percent), the effective intensity requirements of para 2.5 shall be increased by 100 percent.

3 NAVIGATION LIGHT SYSTEM

3.1 WING LIGHTS (SIDE EXTREMITY LIGHTS)

3.1.1 Location - The side lights shall be spaced laterally and as far apart as practicable on the fuselage. Such lights may be installed on other than the fuselage as agreed with the Rotorcraft Project Director. Supplementary lights may be installed in any location as necessary to meet minimum light distribution requirements. Each light as installed shall show unbroken light in accordance with Figs. 1 and 2.

3.1.2 Colour - The side lights shall be Aviation Red for the left side and Aviation Green for the right side.

3.1.3 Candlepower - Candlepower requirements shall be as shown in Figs. 1 and 2.

3.2 TAIL LIGHTS (AFT EXTREMITY LIGHTS)

3.2.1 Location - The tail light shall be located as near as practicable to the rear extremity of the fuselage. Supplementary lights may be installed if necessary to meet the minimum distribution requirements of Figs. 1 and 2.

3.2.2 Colour - Aviation White.

3.2.3 Candlepower - The candlepower requirements shall be as shown in Figs. 1 and 2.

3.3 Rotorcraft not provided with anti-collision lights shall be furnished with flashing navigation lights. The flash rate for the navigation lights shall be 85 $\pm$ 15 flashes per minute with an "on" to "off" ratio of between 3:1 and 1.857:1. The complete system, side and tail lights, shall be flashed simultaneously.

3.4 Naval rotorcraft shall be fitted with flashing navigation lights controlled by a FLASHING/OFF/STEADY switch and by a BRIGHT/DIM switch. They shall, where possible, be fitted with a flashing white light(s) visible in all directions and be capable of independent operation.

4 COLOURS

4.1 Chromaticity - Colours reference in this document shall have the applicable International Commission on Illumination (CIE) chromaticity co-ordinates as follows (see also Fig. 3):

(i) Aviation Red:

y is not greater than 0.335,

z is not greater than 0.020.

(ii) Aviation Green:

x is not greater than 0.440 - 0.320y,

x is not greater than y - 0.170,

y is not less than 0.390 - 0.170x.

(iii) Aviation White:

x is not less than 0.300,

x is not greater than 0.540,

y- y_o is not numerically greater than 0.01. y_o is the y co-ordinate of the Plankian radiator for which "x_o = x".

(iv) Red for Anti-collision Lights:

y is not greater than 0.350,

z is not greater than 0.020.

4.2 Colour Limits - Fig. 3 graphically represents limit co-ordinates of Aviation colours.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	-	3153

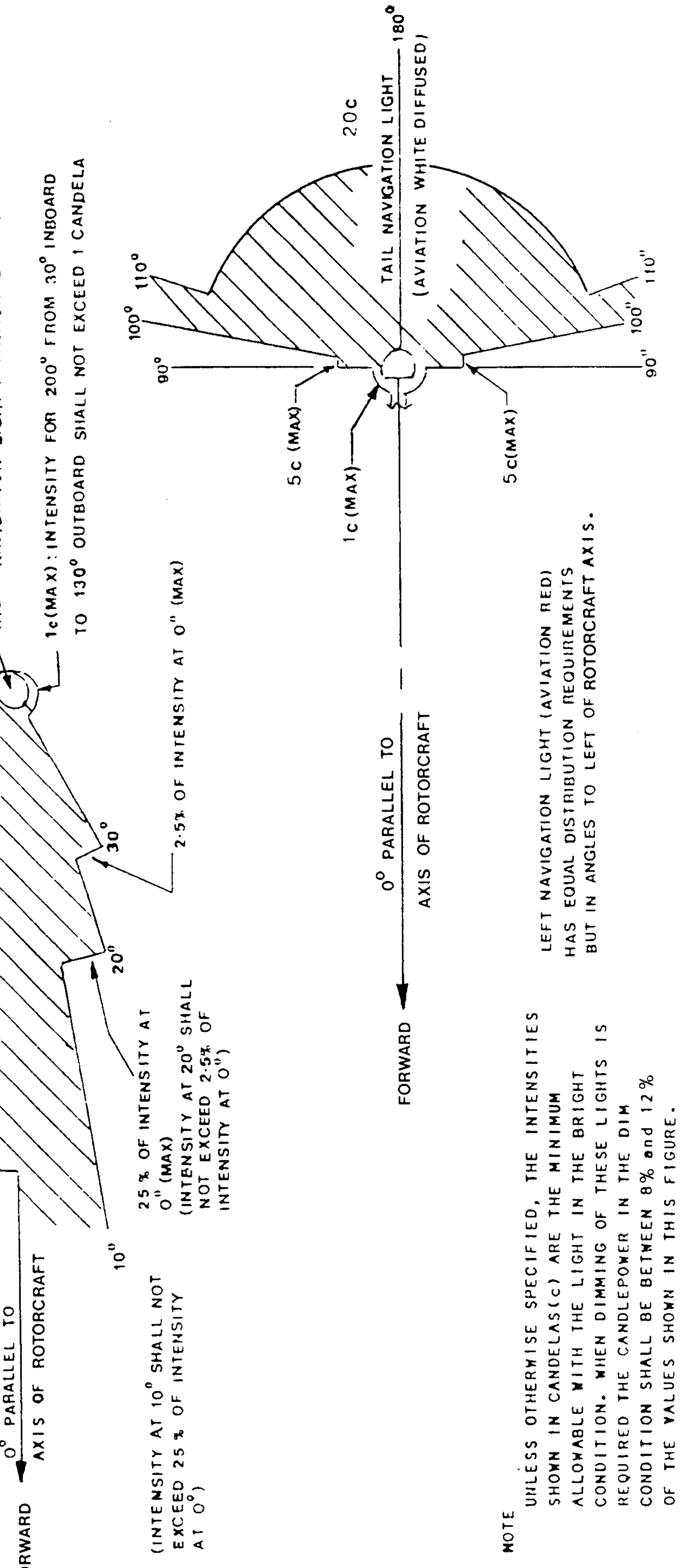
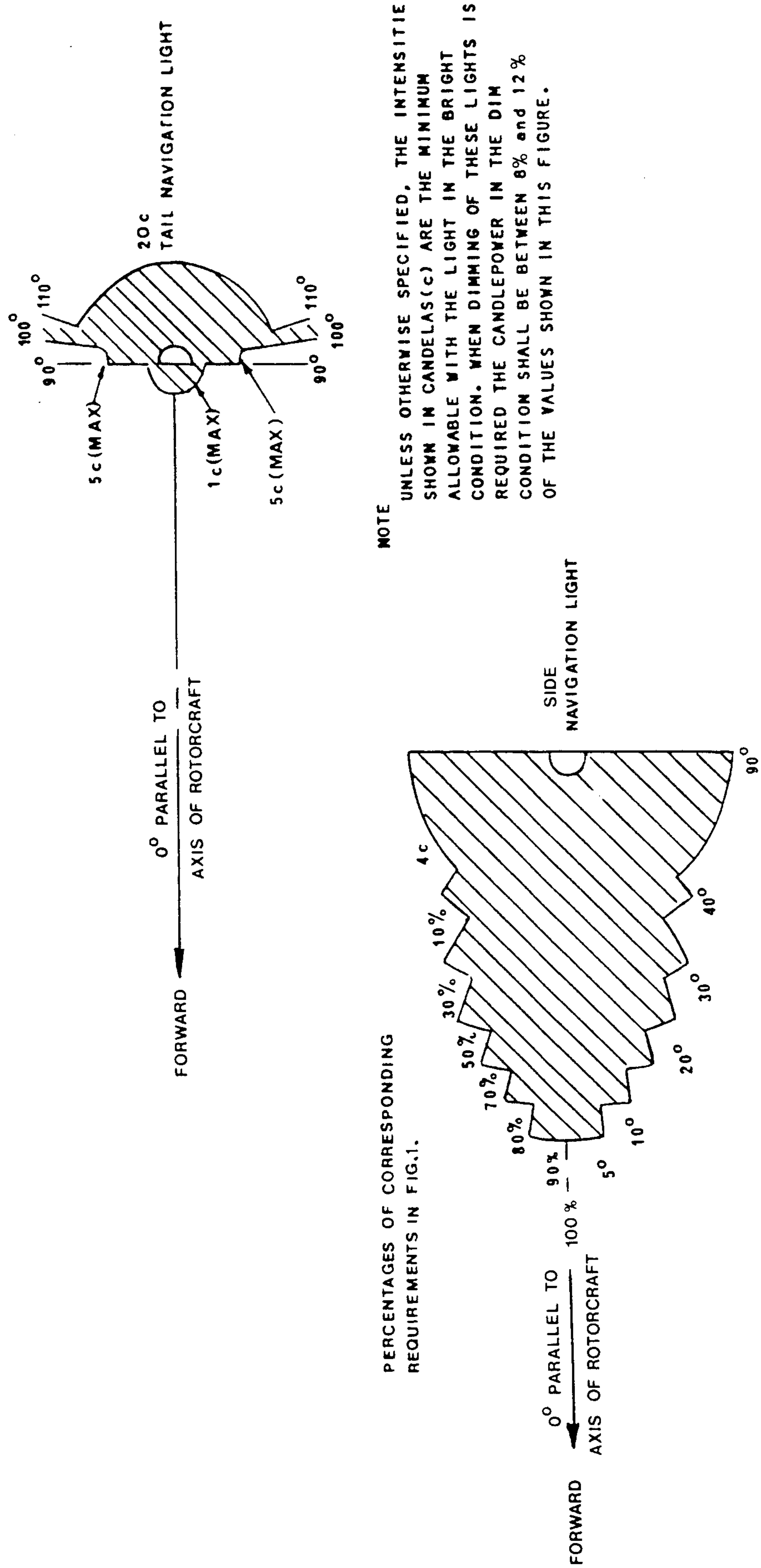


FIG.1 INTENSITY DISTRIBUTIONS IN THE HORIZONTAL PLANES, THROUGH THE CENTRES OF THE LIGHTS



NOTE

UNLESS OTHERWISE SPECIFIED, THE INTENSITIES SHOWN IN CANDELAS(c) ARE THE MINIMUM ALLOWABLE WITH THE LIGHT IN THE BRIGHT CONDITION. WHEN DIMMING OF THESE LIGHTS IS REQUIRED THE CANDLEPOWER IN THE DIM CONDITION SHALL BE BETWEEN 8% and 12% OF THE VALUES SHOWN IN THIS FIGURE.

FIG.2 INTENSITY DISTRIBUTIONS IN THE VERTICAL PLANES THROUGH THE CENTRES OF THE LIGHTS

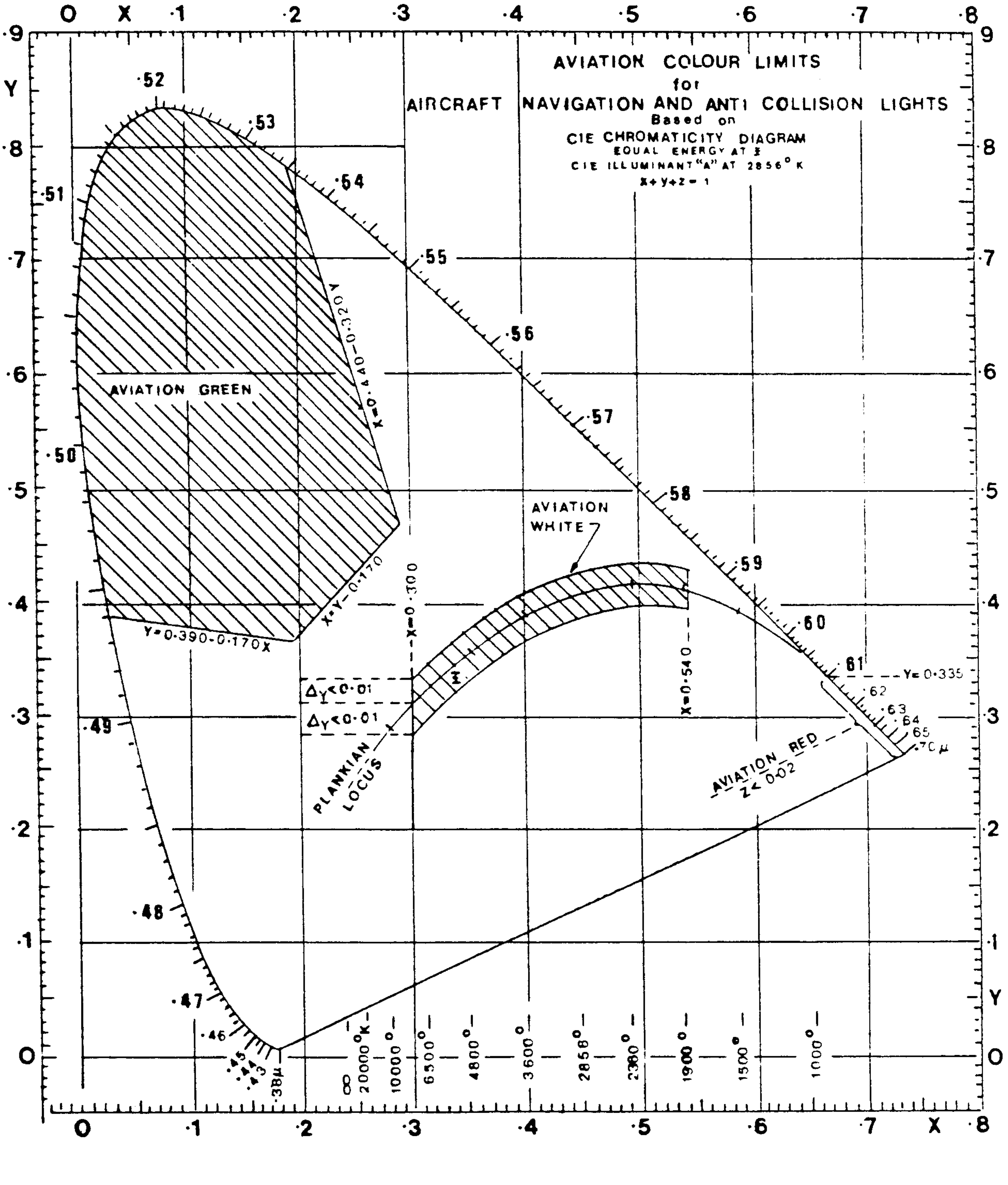


FIG. 3 CIE CHROMATICITY DIAGRAM SHOWING AVIATION COLOUR LIMITS

CHAPTER 111

AL/1

RESTRAINT AND PARACHUTE HARNESS FOR AIRCREW[1]

1 INTRODUCTION

1.1 The requirements of this Chapter shall apply to restraint and parachute harnesses of all combat rotorcraft.

2 DESIGN REQUIREMENTS

2.1 The harness shall prevent injury to the user caused by multi-directional forces acting singly or together up to the level of human tolerance. Minimum static strength requirements are given in Chapter 307.

2.2 The dimensions of the crew and occupants used for the design of the harness, harness to seat or rotorcraft, shall provide for 'large' and 'small' airmen as defined in Volume 1, Leaflet 105/3 and full allowance shall be made for all items of clothing and equipment in the applicable Aircrew Equipment Assembly Schedule (AEA).

2.3 Provision should be made if possible for the attenuation of energy in the harness, seat or rotorcraft so that the peak accelerations applied to the harness user is minimised.

2.4 Maximum practical area shall be provided by the harness webbing in order to distribute the loads without injury and it shall prevent secondary impact of the head and torso with cockpit structures. The harness should apply restraint to strong parts of the body (e.g., pelvis and chest) but should not cause specific injuries.

2.5 The harness shall be lightweight and comfortable in use. Hardware should not be sited on the harness so that it is placed over bony prominences of the user. It shall be easily adjustable for all sizes of personnel and designed so that all service personnel intended to use the harness can do so correctly and comfortably. The use of the widest selection of up-to-date and appropriate anthropometric data should be made to ensure that this requirement is met. The harness should be designed so that it can be adjusted and worn firmly in flight so that dynamic amplification of acceleration is avoided.

2.6 The harness should be easy to put on and release. It should be possible for an aircrew member to fit his harness without aid. The harness shall be simple in form and avoid complex routing. It should be impossible to fasten or use the harness incorrectly. Suitable stowages for the components of the unfastened harness should be provided as an aid to strapping in, especially in confined cockpits. Release of the crew member from the seat without parachute or survival equipment shall be provided by activation of a single control to permit immediate ground escape. Retention of the survival equipment may be required when ditching.

2.7 The harness shall provide sufficient freedom for all sizes of aircrew to operate and reach all controls and maintain adequate external vision. An inertia reel may be used with the harness to provide both restraint and minimum restriction.

3 RESTRAINT HARNESS GEOMETRY

3.1 The harness shall incorporate a pair of over-the-shoulder straps, a pair of lap straps and a negative g strap.

3.2 The shoulder straps should start from a point on the seat back 674mm ±15mm above the compressed sitting platform measured along the back tangent line. The straps should be restrained at that point so that not more than 15 mm lateral movement of each strap is allowed. Where cushions are used, the line of the compressed sitting platform is the tangent line to the upper surface of the cushion of the sitting platform when compressed by an occupant wearing full AEA of the maximum body weight specified by Volume 1, Leaflet 105/3.

3.3 The lap straps should start from the side of the seat such that the strap centre line is placed at an angle of 45° to 55° relative to the centre line above the compressed sitting platform. The lap straps should be placed at this angle so that under impact the straps remain over the anterior superior spines of the pelvis for transmission of acceleration force to the body. It should be possible to vary the length of each lap and shoulder strap with a force not exceeding 134 N by means of a simple self locking buckle.

3.4 The negative g strap should start vertically from a point on the centre line of the compressed sitting, platform 350 mm ±25 mm forwards of the front surface of the seat back and should connect to the central release point of the harness. When the harness is connected, the strap should be designed so that the centre of the harness release fitting is 180 mm ±25 mm above the top of the compressed sitting platform, and the lap straps intersect the compressed sitting platform at 45° to 55° . If the sitting position is not varied, a correctly designed negative g strap should require no length adjustment. The harness release fitting should be permanently attached to the negative g strap.

4 PARACHUTE HARNESS GEOMETRY

4.1 Where restraint and parachute harnesses are combined into one assembly, either man (torso) or seat mounted, the restraint harness requirements will cover both applications, with the proviso that the harness geometry shall positively exclude inadvertent escape from the harness. Where parachute harnesses are used separately from restraint harnesses, the same requirements apply. When harnesses are so combined, it is essential that the parachute function of the harness is not compromised.

5 QUICK RELEASE FITTINGS

5.1 The restraint and preferably the parachute harness and release fitting shall be designed so that all components of the harness can be released from a single point. In order to prevent inadvertent release, a release action requiring two separate, differing and sequenced actions shall be incorporated. Quick release fittings which eject the lugs or attachments on release to prevent inadvertent re-locking are preferable. Variations of the types of release action should be minimised to reduce confusion in an emergency. The action should be either a rotary, depressing or lifting type and the particular actions required should be clearly described on the fitting. In addition, the action should be simple, capable of being effected with either hand in hot, cold or wet environments and in darkness wearing equipment appropriate to the crew task. Aircrew should be able to determine by feel alone whether the quick release fitting is locked or released.

6 INERTIA REELS

6.1 Multi-directional rather than uni-directional sensitive inertia reels shall be used and a control placed on the left-hand side of the seat shall be provided so that the reel can be mechanically locked at will. The stroke of the reel should be adequate to allow all sizes of aircrew specified to reach all crew station controls and maintain internal and external vision appropriate to their task. The force retracting the strap into the reel should not be less than 22N at 50 mm extension of the strap and not less than 10 N at full retraction. Inertia reels incorporating powered retraction devices shall be capable of retracting the most critical aircrew member and equipment specified into the correct position with minimal time delay against a minimal acceleration load of 3 g acting in any direction.

7 TESTING

7.1 Harnesses, inertia reels and quick release fittings shall be subjected to both static and dynamic tests in accordance with Chapter 307. Where energy attenuating devices are incorporated, specific examination of their performance shall be made under dynamic conditions. These tests shall ensure that the additional movement provided by these devices do not introduce additional hazards for the crew (e.g., contact with cockpit structures).

REFERENCES

Reference	ASCC Air Standard	STANAG
1	61/2	-

AL/1

CHAPTER 112

REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

1 INTRODUCTION

1.1 This chapter contains design aims and requirements which enhance the survivability of the rotorcraft and its crew by reducing their vulnerability to battle damage.

1.2 See Leaflet 112/1 for general background information on Reduction of Vulnerability to Battle Damage, Design Aims, Vulnerability Analysis, Protection Measures, Battle Damage Repair, and Kill Categories. See Chapter 307 for Crash Landing and Ditching requirements and Chapter 113 for Protection Systems for Aircrew.

1.3 Table 1 gives the Defined Threat Effects and also contains some data which may be used in a Vulnerability Analysis if more specific information is not available. The Volume 1 Chapter 600 aircraft classes are directly applicable to Aeroplanes and may also be applied to Rotorcraft to obtain probabilities of occurrence if more relevant data is not given in the Rotorcraft Specification. The directional qualifiers are applicable only to Aeroplanes. For Rotorcraft the full 360° in elevation and azimuth should be considered and an appropriate set adopted for the project under consideration .

2 DEFINITIONS

NOTE: The following definitions apply to terms used in this Chapter and in the Leaflet. They may not be the same as definitions of the same terms in JSP-110 or Mil-Std-2089.

2.1 SURVIVABILITY The capability of a rotorcraft to avoid and/or withstand the effects of a combat environment.

2.2 VULNERABILITY The degree to which the Defined and Specified Threat Effects (see below) will degrade flight or mission capability.

2.3 SUSCEPTIBILITY The degree to which a rotorcraft, equipment or weapon system is open to effective attack from a threat or threats.

2.4 THREATS Those hostile elements of a combat environment which could reduce the ability of a rotorcraft, its systems, and crew, to perform its mission.

2.5 THREAT EFFECT The definition of a Threat in terms of those physical characteristics which affect rotorcraft design.

2.6 SPECIFIED THREAT EFFECTS Those Threat Effects referred to in the Rotorcraft Specification.

2.7 DEFINED THREAT EFFECTS Those Threat Effects defined in Table 1.

2.8 SORTIE An operational flight by one rotorcraft.

2.9 MISSION The task to be performed during a sortie.

2.10 BATTLE DAMAGE REPAIR The best repair that can be done to make the rotorcraft missionworthy after suffering battle damage, taking into account the time and resources available and the operational requirements.

2.11 PROBABILITY OF OCCURRENCE (see Table 1) A function of three factors:

(i) Probability of encountering a particular threat.

(ii) The susceptibility of the rotorcraft to the threat.

(iii) Threat lethality.

These are combined in Table 1 to give a single probability for each class of rotorcraft that indicates the importance of each type of threat.

3 DESIGN

3.1 The principle design aim shall be to maximise the probability that no single Threat Effect defined in Table 1 or in the Rotorcraft Specification will degrade the Flying Qualities of the Rotorcraft below Level 3 of DEF STAN 00-970 Volume 1 Part 6.

3.2 The vulnerability of all systems shall be considered. The vulnerability of critical systems shall be reduced by application, where relevant, of the following techniques:

(i) Configuration adjustment.

(ii) Redundancy.

(iii) Separation.

(iv) Change of Dimensions.

(v) Additional protection measures.

(vi) Use of components designed to tolerate battle damage.

3.3 Consideration shall be given to isolation and suppression of fire and sources of ignition shall be separated effectively from flammable fluids and gases. See also Chapters 702 and 712.

3.4 Consideration shall be given to the following when assessing any material for use in the design:

(i) Nuclear, biological and chemical effects.

(ii) Repairability.

(iii) Residual structural strength.

3.5 Consideration shall be given to the probability of a hit on each pressurised gas storage vessel or gas/oil hydraulic accumulator, which makes a hole in the vessel releasing the energy, and to the effects this would have on the vulnerability of structure and systems in the vicinity. (See also Chapter 719).

4 VULNERABILITY ANALYSIS

4.1 The Chief Designer shall consult with the Rotorcraft Project Director and establish whether, and how, the vulnerability of the rotorcraft to the Defined and Specified Threat Effects will be assessed, and consider how consequent design changes, if any, will be introduced.

4.2 The Vulnerability Analysis shall include a Casualty Reduction Analysis (See Chapter 113).

5 BATTLE DAMAGE REPAIR

5.1 The designer shall consider and provide for the repair of structure and of flight and mission critical systems following battle damage and shall incorporate such design features as will facilitate battle damage repair.

TABLE 1

TABLE OF DEFINED THREAT EFFECTS

		THREAT	SOURCE	FORM (Hit density, impact area, velocity, mass).	PROBABILITY OF OCCURENCE BY AIRCRAFT CLASS (NOTE 2)				DIRECTIONAL QUALIFIERS (NOTE 4)		GENERAL EFFECT
					I	II	III	IV	Elevation ∝ (deg)	Azimuth β (deg)	
	a	INERT BULLETS	GUNS	0.2/m²; over aircraft; 600m/s; 7g	0.40	0.30	0.05	0.05	+5 -15	+60 -60	PENETRATION
	b	INERT FRAGMENTS	MISSILES	20/m²; over 2 m x 7m; 2000m/s; 5g	0.05	0.15	0.40	0.60	(See Note 3)		PENETRATION
Group 1 Threats (Conventional)	c	SHELL	GUNS	0.05/m²; over aircraft; 23mm) (Note 1)	0.40	0.40	0.40	0.15	+5 -10	+60 -60	DISRUPTION
	d	WARHEAD	MISSILES	Fragments and blast (Note 1)	0.15	0.15	0.15	0.20	+10 -10	+170 -170	DISRUPTION
	e	INCENDIARY BULLETS	GUNS	(23mm) one part in four of threats a and c	-	-	-	-	-	-	FIRE
		TOTAL FOR GROUP 1 THREATS			1.0	1.0	1.0	1.0			
		HEAT									
	f	- DIRECTED	LASER	See Aircraft Specification	0.0	0.2	0.3	0.2	+0 -90	+180 -180	PENETRATION
	g	- GENERAL	NUCLEAR	" " ")							DEGRADATION
Group 2 Threats	h	NEUTRON FLUX	NUCLEAR	" " ")							DISRUPTION
(Non-Conventional)	i	GAMMA RADIATION	NUCLEAR	" " ")	0.8	0.4	0.5	0.3			DEGRADATION
	j	ELECTROMAGNETIC	NUCLEAR	" " ")							DISRUPTION
	k	BLAST	NUCLEAR	" " ")							DISRUPTION
	l	BIOLOGICAL	-)	See NGASR No. 562 for crew and	0.1	0.2	0.1	0.25			INCAPACITATION
	m	CHEMICAL	-)	personnel protection	0.1	0.2	0.1	0.25			INCAPACITATION
		TOTAL FOR GROUP 2 THREATS			1.0	1.0	1.0	1.0			

NOTES:
1. See RAE Report No. 79123.
2. See Chapter 600 for definitions.
3. From any direction perpendicular to the axis of the aircraft at any point within the length of the aircraft.
4. Not applicable to Rotorcraft for which the full 360° circle in elevation and azimuth should be considered.

LEAFLET 112/1

AL/3

REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

GENERAL REQUIREMENTS

1 GENERAL (See also Ref 1)

1.1 The survivability of a rotorcraft in battle conditions depends on its susceptibility (Detection, Acquisition, Tracking, and Threat Avoidance) and on its Vulnerability.

1.2 Detection, Acquisition and Tracking are functions of energy emitted by, or reflected by the rotorcraft and its components, the predictability of the flight path of the rotorcraft, and its behaviour. Contributing to detection signatures are:

(i) Radar reflection (Radar Cross Section).

(ii) Infra red radiation.

(iii) Ultra violet radiation.

(iv) Visual contrast.

(v) Smoke emission.

(vi) Noise radiation.

(vii) Electromagnetic radiation.

1.3 Threat avoidance is related to:

(i) Warning devices.

(ii) Countermeasures

(iii) Speed.

(iv) Height.

(v) Manoeuvre

(vi) Size.

(vii) Shape.

1.4 Vulnerability, in an analysis, is quantified as the probability that the rotorcraft will be degraded to one of the defined kill levels (see para 5 below), after being subjected to the Defined or Specified threats.

2 DESIGN AIMS

2.1 The objective of the design aim stated in Chapter 112 para 3.1 is to set a standard for flight control, structural integrity and operation of all systems. The general strategy for the reduction of vulnerability to battle damage is based on:

(i) Design for tolerance of battle damage at component level.

(ii) Rotorcraft layout to minimise weapon effects.

(iii) System/Component redundancy.

(iv) Damage control equipment (e.g. fire and explosion suppression).

(v) Shielding by non-flight-essential components, armour or other structure.

(vi) Reduction of airframe vulnerability.

2.2 The Vulnerability Analysis of Chapter 112 para 4 is an iterative process, the results of which will show how far the design meets the requirements of Chapter 112. However, the results of the analysis will be influenced by trade-off effects, and the impact of these results on the design will be influenced by the time and money available for re-design after completion of the analysis.

3 PROTECTION MEASURES

The following is a list of measures which should be considered. For any particular project some will be more important than others and some may be omitted. The order of priority will be determined by the Vulnerability Analysis.

3.1 FUEL SYSTEM

(i) Design of components to tolerate battle damage.

(ii) Continuance of supply.

(iii) Protection against leakage, explosion and fire.

(iv) Low volatility fuels.

3.2 POWER SYSTEMS AND SIGNALLING FOR FLYING CONTROLS

(i) Design of components to tolerate battle damage.

(ii) Redundancy.

(iii) Use of fire resistant hydraulic fluid.

(iv) Jam proofing.

(v) Leak sensing hydraulic systems.

3.3 ENGINE(S)

(i) Design of components to tolerate battle damage.

(ii) Redundancy.

(iii) Separation of engines and/or vital components.

(iv) Protection of ancillaries.

3.4 CREW (See also Chapter 113 and Leaflet 113/1).

(i) Redundancy.

(ii) Close armour protection.

3.5 ROTOR AND TRANSMISSION SYSTEMS

(i) Design of components to tolerate battle damage.

(ii) Redundancy.

(iii) Separation of vital components.

(iv) Protection of ancillaries.

(v) Stability of rotor when controls are severed.

3.6 STRUCTURE

(i) Design to tolerate battle damage.

(ii) Design for repair to battle damage.

(iii) Modular Construction.

(iv) Maximisation of accessibility.

(v) Reduction of airframe vulnerability by consideration of all design possibilities:

(a) Fail safe/multi load path structure.

(b) Material selection.

4 BATTLE DAMAGE REPAIR

4.1 When designing for battle damage repair the designer should consider not only how to restore full airworthiness when the rotorcraft is back at base but also how to get the rotorcraft airborne and back to base following a forced landing.

4.2 The designer should also consider the information to be put into the Repair Manual(s). After combat damage, a decision will have to be made about each damaged area in the structure and systems, as to whether or not it must be repaired immediately. The manuals should therefore contain the information necessary for such decisions which will normally, but not always, be made by Engineering Staff.

4.3 The designer should incorporate such features as removable panels, line replaceable units, and cable and pipe line identification, to assist with the battle damage repair task.

5 UK KILL CATEGORIES

F(t)- Within time (t) following the damaging strike the rotorcraft will become permanently incapable of controlled flight (periods of (t) are normally 0, 15 secs, 5 mins, 20 mins, 30 mins)

C(t)- Within time (t) following the damaging strike the rotorcraft will become unable to perform the stated mission (periods of time (t) are normally 2, 5 and 30 secs).

E(t)- The rotorcraft receives damage which will keep it grounded for repairs for time (t). The preferred periods for assessment purposes are 8, 24, 48 hours and infinity (i.e. write off).

Note: The times quoted are typical of those used in comparative studies. Times used for a particular analysis will depend on the type of project and the mission profile under consideration.

6 US KILL CATEGORIES

These have been taken from Ref 2 and are given in Table 1 in this leaflet with the UK Kill Categories for comparison.

7 FURTHER INFORMATION

7.1 Where the information contained in Chapter 112 or in this Leaflet is not sufficient for the designer's purpose, particularly with regard to the Vulnerability Analysis, and with regard to reports or further advice on Biological and Chemical effects, reference should be made to the Rotorcraft Project Director in the first instance.

7.2 American information on Design for Survivability and Vulnerability Reduction is contained in the MIL-HDBKS and MIL-STDS listed below. The Standards contain requirements and definitions. The Handbooks contain background information and acceptable methods of compliance.

REFERENCES

1 MIL-STD-2069: 24 August 1981: Requirements for aircraft non-nuclear survivability program.

2 MIL-STD-2089: 21 July 1981: Aircraft non-nuclear survivability terms.

3 RAE TR 79123 - September 1979 - by R.G.E. Mallin.

4 MIL-HDBK-268(AS) 5 Aug 82: Survivability Enhancement, Aircraft, Conventional Weapon Threats, Design and Evaluation Guidelines.

5 MIL-HDBK-336-1, 2 and 3. 25 October 1982; 26 August 1983; 31 January 1983; Survivability, Aircraft, Non-Nuclear, VOL 1 - General Criteria, VOL 2 - Airframe, VOL 3 - Engine.

6 AGARD - CP 212 - P 218396 - Design for reduction of aircraft vulnerability - by Lt Col R.T. Remers USAF.

TABLE 1 BRITISH AND AMERICAN KILL LEVELS

Definition	American (MIL-STD-2089 Para 5.3.3.2)	British (RAE)
Aeroplane is lost from the inventory by:	Attrition kill	F(t)
- disintegrating immediately on being hit	KK - kill (Catastrophic kill)	t = 0 sec
- falling out of control		
- in 15 seconds	-	t = 15 sec
- in 30 seconds	K - kill	-
- in 5 minutes	A - kill	t = 5 min
- in 20 minutes	-	t = 20 min
- in 30 minutes	B - kill	t = 30 min
- before completing mission	C - kill	-
Unable to complete stated mission within time (t) following a strike	Mission abort kill	C(t)
	-	t = 2 sec
	-	t = 5 sec
	-	t = 30 sec
Unable to complete part of mission	Mission limiting condition	-
Mission completed but repairs required before next mission	Mission available kill	-
Special Rotorcraft Category in which damage or a warning required a forced landing	Forced landing kill	-
Damage causes aeroplane to miss its next scheduled mission	Repair time kill	
	-	E(t)
	-	t = 8 hrs
	-	t = 24 hrs
	-	t = 48 hrs
		t = Infinity (write-off)
Damage which makes structural damage on landing probable (e.g. a burst tyre)	E - kill	-
VTOL aeroplane only. Damage which causes the loss of vertical operational ability.	V - kill	-
	NUCLEAR	
No appreciable damage - normal turnround	Sure safe	-
Mission cannot be completed but continued controlled flight is possible	Mission kill	-
Damage which causes aeroplane to fall immediately out of control	Sure kill	-

CHAPTER 113

PROTECTION OF AIRCREW AGAINST CONVENTIONAL WEAPONS[1]

1 GENERAL

1.1 This chapter contains requirements for the provision of efficient ballistic protection systems for aircrew, particularly armour, and are additional to those requirements relating to aircrew protection contained in Chapter 112.

1.2 The requirements are applicable to all types of rotorcraft except research, primary trainer, basic trainer and transport rotorcraft unless required by the Rotorcraft Specification.

2 DESIGN

2.1 Structure and non-critical components shall be used as much as possible for shielding the aircrew.

2.2 Multiple pilot stations shall be separated as much as practicable and consideration shall be given to the provision of shielding between them.

2.3 Aircrew protection systems shall not restrict aircrew mobility and vision, nor access to controls to the extent that the accomplishment of the mission is affected. They shall also not preclude the use of survival equipment nor Nuclear, Biological and Chemical protection equipment.

2.4 Aircrew protection systems shall not interfere with Normal Entrance and Exit (Chapter 109), nor Emergency Escape (Chapter 102), nor constitute a hazard in the event of a Crash Landing or Ditching (Chapter 307) .

2.5 Materials used for armour protection or fragment suppression shall be selected to provide the most suitable characteristics for their purpose at each location. (See Leaflet 113/1 para 3).

2.6 The Vulnerability Analysis of Chapter 112 shall include a Casualty Reduction Analysis.

2.7 Consideration shall be given to the need for tests to verify the protection provided against Conventional Defined and Specified Threats (see Chapter 112) and where tests are considered to be necessary, a programme shall be agreed with the Rotorcraft Project Director.

3 ARMOUR

3.1 Where armour is provided for protection of the crew, the following requirements apply:

3.1.1 Where the armour is an integral part of the rotorcraft structure, the structure including the armour shall meet all relevant design requirements.

3.1.2 Where the armour is an integral part of the seat, it shall be capable of sustaining the loads applied to the seat, and the seat, including the armour, shall meet all relevant design requirements.

3.1.3 Where the armour is attached to the rotorcraft structure, it shall be treated as an item under Chapter 307 para 3.1.1(ii).

3.1.4 Where the armour is attached to the seat or mounted on the pilot or crewman, the mass of the armour shall be included in the mass of the seat when designing to meet the requirements of Chapter 307.

3.2 The forces on the armour attachments and back-up structure arising from direct projectile impact shall also be considered.

3.3 Materials which generate spall particles shall not be used unless suitable provision is made to suppress the spall, to avoid injury to aircrew and to prevent increase of vulnerability to an unacceptable level.

3.4 Body armour shall be designed for quick release in an emergency.

4 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

4.1 Any equipment specifically required to protect the crew against Defined and Specified Threat Effects (see Chapter 112) will be listed in the Rotorcraft Specification. Where space provision, fixed fittings and/or support systems for this equipment are required, this also will be stated in the Rotorcraft Specification.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/68	-

LEAFLET 113/1

AL/3

PROTECTION OF AIRCREW AGAINST CONVENTIONAL WEAPONS[1]

GENERAL REQUIREMENTS

1 GENERAL

1.1 This Leaflet contains non-mandatory requirements, background information, and advice on the provision of protection systems for aircrew. It covers those parts of the requirements that are not already incorporated in Chapters 112 and 114.

2 DESIGN

2.1 The protection of aircrew stations and personnel against Defined and Specified Threat Effects (see Chapter 112) will be achieved by suitable design of appropriate protection systems.

2.2 The following protection systems should be considered:

(i) Increased separation of crew members.

(ii) Shielding by structure or non-critical components.

(iii) Armour protection or fragment suppression materials.

2.3 The systems selected and the amount of protection provided, will be determined by the Vulnerability Analysis of Chapter 112.

3 ARMOUR

3.1 The effectiveness of armour protection for a particular crew member will be improved, or the mass penalty reduced, by the following:

(i) Location of the armour as close as possible to the crew member.

(ii) Location and shaping of the armour to provide protection for more than one crew member and/or for vital equipment at the same time.

(iii) Use of armour as structure.

(iv) Use of removable armour as special role equipment.

3.2 The effectiveness of material for armour protection will be influenced by the following:

(i) Mass and bulk of the material.

(ii) Deflection characteristics.

(iii) Spall characteristics.

(iv) Multi-hit capability.

(v) Durability

(vi) Ease of replacement.

(vii) Cost and availability.

3.3 Armour may be provided as:

(i) body armour worn by the crew member,

(ii) armoured seats with integral or bolt-on armour,

(iii) armour added to or integrated into floors, sidewalls bulkheads and instrument panels,

(iv) transparent armour where visibility is required, and

(v) external armour attached to the airframe in the vicinity of aircrew.

3.4 Where body armour and structural armour are both provided, the designs should be integrated to eliminate gaps and overlaps.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	10/68	-

PART 1 APPENDIX No. 1
GENERAL AND OPERATIONAL REQUIREMENTS
MILITARY DERIVATIVES OF CIVIL ROTORCRAFT

1 INTRODUCTION

1.1 This Appendix covers general and operational requirements for military derivatives of civil rotorcraft.

1.2 Each paragraph of the Appendix consists of a table and amplifying notes comparing military and civil requirements for individual chapters from Part 1 of DEF STAN 00-970 Volume 2. In general, only those parts of the DEF STAN in which military and civil requirements differ are included in the Appendix and where no reference is made to DEF STAN chapter/paragraph. numbers it is considered that the military and civil requirements are compatible and adequately covered by the military requirements (unless explained otherwise in the amplifying notes).

1.3 The issue or change number of the civil airworthiness requirements referred to in this Appendix are recorded in paragraph 9.5 of the main introduction to DEF STAN 00-970 Volume 2.

2 COMPLIANCE CHECKS/ASSESSMENT

2.1 Throughout this Appendix reference is made to the need for ‘Compliance Checks or Assessments’ of the existing civil rotorcraft against the military requirements of DEF STAN 00-970. A full explanation of the aims of compliance checks or assessments is given in paragraph 9 of the main Introduction to DEF STAN 00-970 Volume 2.

3 GENERAL REQUIREMENTS

3.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 100	BCAR’S		FAR’S		JAR’S	
Para Item	Section G	29	27	29	27	29
1. Mock-ups	--	--	--	--	--	--
2. Standard items	--	--	--	--	--	--
3. Installation Information for Items of Equip’t	6-1,2 and 3	Sub Part F	Sub Part F	Sub Part F	Sub Part F	Sub Part F
4. Strength	3-1,4	303 305	303 305	303 305	303 305	303 305

DEF STAN 00-970 CHAPTER 100	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
5. Vibration	See Appendix No.1 to DEF STAN 00-970 Volume 2 Part 5.					
6. Tests	3-1,4 4-1,4	307	307	307	307	307
7. Prevention of Incorrect assembly of systems	4-1,11 4-8,2 5-1,2 6-4,2	671	671	671	671	671
9. Power-operated systems	4-1,3 4-8,3 6-2,2	672	672 673	672 673	672 673	672 673
10. Alignment of Directional Sensitive Equipment & Weapons	--	--	--	--	--	--
11. External Vision	4-2,3	773 775	773	773	773	773
12. External Vision	6-7	1383	1383	1383	1383	1383
13. Destruction of Rotorcraft	--	--	--	--	--	--
14. Loose Article Hazards-Control Systems	4-8,2	685	685	685	685	685
15. Prevention of Accidental Damage	4-8,2	685	685	685	685	685

DEF STAN 00-970 CHAPTER 100 Para Item	BCAR'S Section G	BCAR'S 29	FAR'S 27	FAR'S 29	JAR'S 27	JAR'S 29
16. Jettisoning of Stores	--	--	--	--	--	--
17. Folding Installations (Naval Rotorcraft)	--	--	--	--	--	--
18. External Communications	--	--	--	--	--	--
19. NBC Equipment	--	--	--	--	--	--
20. Carriage of Underslung Loads	4-12	865	865	865	865	865
23. Sonar Locator	--	--	--	--	--	--
24. Design Requirements for Aircraft Equipment	6-1		Sub Part F			
26. Reliability	6-1	1309	1309	1309	1309	1309

3.2 There are no equivalent civil requirements in BCAR's, FAR's or JAR's to the military requirements defined under paras 1,2,10,13,16,17,18,19 or 23.

However, it is expected that civil rotorcraft will comply with 18.

3.3 In defining strength Chapter 100, paragraph 4 refers to strength clauses in systems and installation chapters.

3.4 The military requirements defined under paragraphs 5,12 & 14 are generally compatible with civil requirements. In respect of paragraph 26 Reliability the civil requirements address this differently but in general the civil aircraft reliability in normal operation will be compatible with DEF STAN 00-970.

3.5 Chapter 100, paragraph 3 refers to Aircraft Equipment Installation (AE 11), Radio Installation Memorandum (RIM) or other equivalent document (DEF STAN 05-123, Chapter 205) Civil Aircraft requirements are contained in the various sections of the civil airworthiness requirements.

4 OPERATION IN VARIOUS CLIMATIC CONDITIONS

4.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 101	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Temperature Limits						
1.1. Basic operation Requirements	1-1 1-2-1	45	45	45	45	45
1.2. Requirements for Design	1-2 App 1	1-2 App 1	--	--	--	--
1.3. Test Requirements	--	--	--	--	--	--
2. Humidity Limits	2-4	2-4	1525 1527	1525 1527	1525 1527	1525 1527
3. Clear Vision	See Appendix 1 Volume 1 Chapter 104					
4. Flight in Icing Conditions	) Detail Requirements in Appendix) 1 Chapter 711					
5. Weatherproofing	4-1,8	609	609	609	609	609
6. Dust and Sand Proofing	5-5	1105	1105	1105	1105	1105

4.2 A military derivative will generally have been designed and certificated to civil requirements very similar to the requirements in Chapter 101. The climatic conditions of operation will be defined in the civil model specification which may refer to BS(3)G100 or MIL STD 210. A compliance assessment will, nevertheless, be needed of the existing rotorcraft in relation to the requirements of Chapter 101.

Note: BS(3)G100 is being superseded by DEF STAN 00-970 and DEF STAN 00-35

5 EMERGENCY ESCAPE

5.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 102	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. General Principles	4-3,3 4-10	561 563 783 801 803 805	561 563 783 801	561 563 783 801 803	561 563 783 801	561 563 783 801 803
3. Emergency Exits	4-3	805 807 809 811	807	805 807 809 811	807	805 807 809 811
4. Emergency Escape/Evacuation Illumination	4-3	801 803 805 809 811	801 807	801 803 805 809 811	801 807	801 803 805 809 811
5. Abandon Aircraft Alarm Signal	--	--	--	--	--	--
6. Emergency Alighting	3-8 4-10 6-12	561 785	561 785 785	561 785 785	561 785 785	561 785 785
7. Tests	4-3,5	803	803	803	803	803

5.2 A compliance assessment for escape on the ground may be required if the military derivative rotorcraft specification calls for members of the armed forces to be fully equipped for combat on landing.

6 OPERATIONAL COLOURING AND MARKINGS

6.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 103	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. Emergency Controls	8-3	1557 1561	1557 1561	1557 1561	1557 1561	1557 1561
3. Canopy or Hatch Controls	--	--	--	--	--	--
4. Emergency Exits and their Operating Controls	4-3,5	1555 1561	1555 1561	1555 1561	1555 1561	1555 1561
5. Liferaft Release and Flotation Controls	6-6	1561	1561	1561	1561	1561
6. Airframe Notice	4-4,3 6-6,2	853 1541 to 1565	853 1541 to 1565	853 1541 to 1565	853 1541 to 1565	853 1541 to 1565
7. Colour Standards at Crew Positions	4-2	1555	1555	1555	1555	1555
8. Maintenance Areas	--	--	--	--	--	--

6.2 A review of military and civil requirements shows that different colour standards are specified. It will be necessary to resolve the differences to a standard acceptable to the service for the rotorcraft in its military role(s).

7 VIEW AND CLEAR VISION

7.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 104	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. General Requirements for Aircrew Stations	4-2))))					
3. Basic Requirements for Pilots Stations	4-2))))))	773 775	773 775	773 775	773 775	773 775
4. Vision Requirements for Pilots	4-2))))					
5. Basic Operational Requirements for Clear Vision	4-2) 4-3)))					

8 CREW STATIONS - GENERAL REQUIREMENTS

8.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 105	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. Nomenclature	8-3,2	1541 to 1565	1541 to 1565	1541 to 1565	1541 to 1565	1541 to 1565
3. Layout	4-2	771	771	771	771	771
4. Freedom of Movement	--	--	--	--	--	--

DEF STAN 00-970 CHAPTER 105 Para Item	BCAR'S Section G	BCAR'S 29	FAR'S 27	FAR'S 29	JAR'S 27	JAR'S 29
5. Noise	--	771	771	771	771	771
7. Design of Seats and Harnesses	4-4	785	2 785	2 785	2 785	2 785
8. Restraint and Parachute Harness for Aircrew	4-4	785	785	785	785	785
9. Armour Protection	--	--	--	--	--	--
10. Cockpit Structure	3-8 4-3	771 785 801	771 785 801	771 785 801	771 785 801	771 785 801
11. Controls	4-8	779 1555 1557	779 1555 1557	779 1555 1557	779 1555 1557	779 1555 1557
12. Warning Cautionary and Advisory Signals	4-5,2 8-2 8-3	1322	1322	1322	1322	1322
13. Indicators	6-1,3	1322	1322	1322	1322	1322
14. Instruments	6-1	1322	1322	1322	1322	1322
15. Lighting	4-2	771 1381	771 1381	771 1831	771 1381	771 1831
16. Blackout Curtains	--	--	--	--	--	--
17. First Aid Kits	ANO's Sch.4	--	--	--	--	--
18. Personal Survival Packs	--	--	--	--	--	--

DEF STAN 00-970 CHAPTER 105	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
19. Aircrew Overnight Kit Stowages	--	--	--	--	--	--
20. Hand Fire Extinguishers	4-3,9	851 853		851 853		851 853
21. Axes and Heat Resisting Gloves	--	--	--	--	--	--
22. Sanitation	4-3,5	--	--	--	--	--
23. Cup holders	--	--	--	--	--	--
24. Signal Pistols	4-2,2	--	--	--	--	--
25. Cross References to Other Chapters	--	--	--	--	--	--

8.2 The requirements for crew stations layouts are, generally, similar between military and civil rotorcraft, and those from DEF STAN 00-970 will apply in principle to military derivatives, subject to compliance checks, as necessary. Paragraph 3.1 of the military requirements refer to the 'Aircrew Equipment Assembly' (AEA), which would normally not have a civil requirement counterpart. However, when the military derivative operates in its military role, some type of AEA might be required. This will be defined in the military derivative specification, and the DEF STAN 00-970 requirement will then apply.

8.3 Where colour coding of illuminated indicators e.g., paragraph 13.4 of Chapter 105 is called for in the military requirements, there is no relevant civil requirement, an assessment of the suitability of the civil installation against the military requirements will be needed.

8.4 Paragraph 4 of Chapter 105 may initiate a compliance assessment if the military derivative is required to operate in severe turbulence for military necessity.

8.5 Paragraphs 7.1 and 10.1 of Chapter 105 refer to Chapter 307 of DEF STAN 00-970 which specifies the static design loads for crash landing and ditching. A compliance assessment may be required dependent on the issue/change No of the civil requirement to which the military derivative has been certificated in its civil role.

8.6 Paragraphs 16 to 19 and 21 to 23 may have similar provision as part of a constructors basic specification. The rotorcraft civil model specification should be reviewed.

9 PILOT'S STATION LAYOUT

9.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 106	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1.2 Introduction See note 9.2	4-2,1	771	771	771	771	771
6.3.1 Seats	4-2,1	785	785	785	785	785
6.4 Ejection Seat	--	--	--	--	--	--
6.5 Side by Side Seats	4-4,1	771	771	771	771	771
7. Head Clearance	4-4,1 and App.1	785	785	785	785	785
See Note 9.2						

9.2 The civil requirements for Pilot's Station Layout are written in very general terms, and there are no comparable requirements in detail for paragraphs 1-6 & 8-12 of Chapter 106. A compliance assessment of the existing rotorcraft in relation to the requirements of Chapter 106 will be needed.

9.3 Re-paragraph 11 of Chapter 106 stowages will already have been provided in the military derivative to comply with Air Navigations Orders Schedule 4, Scale A and its foreign equivalent.

10 PILOT'S COCKPIT - CONTROLS AND INSTRUMENTS

10.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 107	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Introduction	--	--	--	--	--	--
2. Flying Controls and Indicators	4-2 4-5 4-6 4-8 6-4	671 to 696 729 777 779 1321 to 1337	671 to 695 729 777 779 1321 to 1337			
3. Winch Controls	4-12	252 865	865	865	865	865
4. Ground Manoeuvring Controls	4-2,4 4-8 1.1.6	777	777	777	777	777
5. Environment and Escape Controls and Instruments	4-2 4-4	771 777 785	771 777 785	771 777 785	771 777 785	771 777 785
6. Controls and Indication for Shipborne Operation	--	--	--	--	--	--
7. Engine Power Controls		777 1141 to 1147 1157 1159 1555	777 1141 to 1147 1555	777 1141 to 1147 1157 1159 1555	777 1141 to 1147 1555	777 1141 to 1147 1157 1159 1555

DEF STAN 00-970 CHAPTER 107 Para Item	BCAR'S Section G	29	FAR'S 27	29	JAR'S 27	29
8. Fuel Systems Controls	4-2 5-2	777 1141 1142 1143 1147 1157	777 1141 1142 1143 1147	777 1141 1142 1143 1147 1157	777 1141 1142 1143 1147	777 1141 1142 1143 1147 1157
9. Electrical Controls	See Note 10.5					
10. Radio Controls	(Section R)					
11. Armament Controls	--	--	--	--	--	--
12. Flight Instruments	6-1 6-10	1321 to 1335	1321 to 1335	1321 to 1335	1321 to 1335	1321 to 1335
13. Engine Instruments	6-1,2 2-11	1337	1337	1337	1337	1337
14. Warning Cautionary Advisory Signals	See Appendix 1 Chapter 105 paras 12 & 13					
15. Second Pilot's Station	4-2	771 773 777	771 773 777	771 773 777	771 773 777	771 773 777
16. Cross References to other Chapters	--	--	--	--	--	--

10.2 The civil requirements for pilot's cockpit are much more generalised, but the main civil references are listed.

10.3 There is no specific reference in the civil requirements for winches except BCAR Section G winch controls although some civil rotorcraft do have such a facility.

10.4 Civil requirements do not directly refer to shipborne operation.

10.5 Civil requirements for electrical controls are included in the specific systems.

10.6 In-flight refuelling is not yet included in civil airworthiness requirements.

11 INTERNAL NOISE

11.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 108	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Introduction	--	771	771		771	
2. Criterion (i) A Noise Environment for Aircrew Protected by an Aircrew Helmet	--	--	--	--	--	--
3. Criterion (ii) A Noise Environment for Unprotected Passengers and Crew	--	771	771		771	
4. Specification Levels	--	--	--	--	--	--
5. Measurement of Rotorcraft Interior Sound Pressure Levels	--	--	--	--	--	--

11.2 Large civil passenger rotorcraft may, no doubt, comply with the requirements of this chapter and therefore so will the military derivative. However, military modifications introduced for the military role(s) may need to be assessed for trainer rotorcraft, an assessment of the existing rotorcraft in relation to the requirements of this chapter may also be necessary.

12 NORMAL ENTRANCE AND EXIT

12.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 109	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
3 Entrance Doors and Hatches	4-3,5	805 807	807	805 807	807	805 807
4. Handgrips and Steps	--	--	--	--	--	--
5. Strength Requirements	3-1 3-2	301 to 309 321 to 341	301 to 309 321 to 341	301 to 309 321 to 341	301 to 309 321 to 341	301 to 309 321 to 341
Table 1: List of Other Important Requirements	--	--	--	--	--	--

12.2 In principle, the military requirements of DEF STAN 00-970 Volume 1 Chapter 109 apply to the military derivative, which will have been designed and certified to the civil requirements listed in the table. A compliance assessment of the existing rotorcraft in relation to Chapter 109 may be needed.

12.3 For paragraph 5 of Chapter 109 refer also to paragraph 3.2 of Appendix No 1 to Part 2 of DEF STAN 00-970 Volume 2.

13 NAVIGATION AND ANTI-COLLISION

13.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 110	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Introduction	6-7,4	1385	1385		1385	
2. Anti-Collision Light System	6-7,3 6-7,5	1401	1401		1401	
3. Navigation	6-7,4	1385 to 1397	1385 to 1397		1385 to 1397	

13.2 Chapter 110 is based on STANAG 3153, the contents of which are compatible with and similar to the civil requirements. However, a compliance assessment of the existing rotorcraft in relation to Chapter 110 will be needed.

14 RESTRAINT AND PARACHUTE HARNESS FOR AIRCREW CHAPTER 111

14.1 This Chapter is primarily concerned with combat Aeroplanes for which there is no civil equivalent. Reference should be made to Chapter 307 for strength requirements. If, however, a military derivative such as a trainer has a combat role then the contents of Chapter 111 will apply and an assessment of its effect made.

15 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE CHAPTER 112

15.1 There are no civil requirements applicable to Reduction of Vulnerability to Battle Damage. If this is a requirement in the military derivative rotorcraft's specification, a study will need to be undertaken.

16 PROTECTION OF AIRCREW AGAINST CONVENTIONAL WEAPONS CHAPTER 114

16.1 There is no civil airworthiness requirement for the protection of aircrew against conventional weapons an assessment will need to be made of the military derivative should this be a requirement for its military role(s).

PART 1 APPENDIX No. 2
GENERAL AND OPERATIONAL REQUIREMENTS
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 100: GENERAL REQUIREMENTS

100	MIL-STD-101	COLOUR CODE FOR PIPE LINES AND COMPRESSED AIR CYLINDERS
	MIL-HDBK-244	GUIDE TO AIRCRAFT/STORES COMPATIBILITY
	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-765	COMPASS SWINGING, AIRCRAFT GENERAL REQUIREMENTS FOR
	MIL-STD-781	RELIABILITY TESTING FOR ENGINEERING DEVELOPMENT, QUALIFICATION, AND PRODUCTION
	MIL-STD-783	LEGENDS FOR USE IN AIRCREW STATIONS AND ON AIRBORNE EQUIPMENT
	MIL-STD-1247	MARKINGS, FUNCTIONS AND HAZARD DESIGNATIONS OF HOSE, PIPE AND TUBE LINES
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-1530	AIRCRAFT STRUCTURAL INTEGRITY PROGRAM , AIRPLANE REQUIREMENTS
	MIL-STD-1763	AIRCRAFT/STORES CERTIFICATION PROCEDURES
	MIL-STD-2124	FLIGHT DATA RECORDER, FUNCTIONAL STANDARDS FOR
	MIL-L-6503	LIGHTING EQUIPMENT, AIRCRAFT, GENERAL SPECIFICATION FOR INSTALLATION OF
	MIL-L-006730	LIGHTING EQUIPMENT, EXTERIOR, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-A-8591	AIRBORNE STORES, SUSPENSION EQUIPMENT AND AIRCRAFT INTERFACE (CARRIAGE PHASE): GENERAL DESIGN CRITERIA FOR
	MIL-A-8650	MOCK UPS, GENERAL SPECIFICATION FOR
	MIL-1-8671	INSTALLATION OF DROPPABLE STORES AND ASSOCIATED RELEASE SYSTEMS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-H-46855	HUMAN ENGINEERING REQUIREMENTS FOR MILITARY SYSTEMS, EQUIPMENT FACILITIES
	MIL-P-85034	POWER PACKAGE, EMERGENCY AC/DC GENERATOR, RAM AIR TURBINE DRIVEN
	AFGS-87221	AIRCRAFT STRUCTURES, GENERAL SPECIFICATION FOR

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. MOCK-UPS

100 1.	MIL-H-46855B	PARA: 3.2.2.1.1
	MIL-M-8650	

2. STANDARD ITEMS

100 2	MIL-STD-1472C	PARA: 4.2

3. INSTALLATION INFORMATION FOR ITEMS OF EQUIPMENT

4. STRENGTH

100 4	MIL-STD-1472C	PARA: 4.5
	MIL-D-23222A	PARA: 3.1.10, 3.3.5.2.3, 3.4.2.2

5. VIBRATION

100 5.	MIL-STD-1472C	PARA: 4.4.(b), 5.8.4
	MIL-H-8501A	PARA: 3.7
	MIL-S-8698	PARA: 3.6.1
	MIL-D-23222A	PARA: 3.7.4
	MIL-H-46855B	PARA: 3.2.2.3.(d)
	AFGS-87221	

6. TESTS

100 6.	MIL-STD-1530A	PARA: 5.3.1.1, 5.3.2.1
	MIL-T-8679	PARA: 3.2.1, 3.2.9.3, 3.4, 3.5, 3.6, 3.7
	MIL-D-23222A	PARA: 3.1.3

7. PREVENTION OF INCORRECT ASSEMBLY OF SYSTEMS

100 7.	MIL-STD-101	
	MIL-STD-783	
	MIL-STD-1247	
	MIL-STD-1472C	PARA: 5.9.1.9

8. CONDITIONS OF OPERATION

100 8.

9. POWER OPERATED SYSTEMS

100 9.	MIL-D-23222A	PARA: 3.13.2.4
	MIL-P-85034	

10. ALIGNMENT OF DIRECTIONALLY SENSITIVE EQUIPMENT AND WEAPONS

100 10.	MIL-HDBK-244	
	MIL-STD-765	

11. EXTERNAL VISION

100 11.	MIL-D -23222A	PARA: 3.12.12, 3.12.13

12. EXTERNAL LIGHTS

100 12.	MIL-L-6503H	PARA: 3.3.7.2, 3.4
	MIL-L-006730	PARA: 3.2.3
	MIL-D-23222A	PARA: 3.13.2.7

100 12.	MIL-L-81174	
13.	DESTRUCTION OF ROTORCRAFT	
100 13.		
14.	LOOSE ARTICLE HAZARDS - CONTROL SYSTEMS	
100 14.		
15.	PREVENTION OF ACCIDENTAL DAMAGE	
100 15.	MIL-STD-250D	PARA: 4.6.(F)
16.	JETTISONING OF STORES.	
100 16.	MIL-HDBK-244	
	MIL-STD-1763	PARA: 5.1.4.5.5
	MIL-A-8591	PARA: 6.3 1.4
	MIL-I-8671	PARA: 3.13
	MIL-D-23222A	PARA: 3.11.6.1, 3.12.9
17.	FOLDING INSTALLATIONS (NAVAL ROTORCRAFT)	
100 17.	MIL-D-23222A	PARA: 3.12.14
18.	EXTERNAL COMMUNICATIONS	
100 18.	MIL-D-23222A	PARA: 3.14.1.7
19.	NBC EQUIPMENT	
100 19.	MIL-STD-1472C	PARA: 4.4.(k), 4.10
	MIL-D-23222A	PARA: 3.12.2.4
	MIL-H-46855B	PARA: 3.2.2.3
20.	CARRIAGE OF UNDERSLUNG LOADS	
100 20.		
21.	ACCIDENT DATA RECORDERS	
100 21.	MIL-STD-2124A	PARA: 1.1, 4
22.	COCKPIT VOICE RECORDERS	
100 22.		
23.	SONAR LOCATOR BEACONS	
100 23.		

24. DESIGN REQUIREMENTS FOR AIRCRAFT EQUIPMENT

100 24.	MIL-STD-781	PARA: 1, 4, 5
	MIL-STD-1472C	PARA: 4

25. DAMAGE CONTAINMENT - COMPONENTS (EXCEPT ENGINES) INCORPORATING HIGH ENERGY ROTORS

100 25.	MIL-STD-1472C	PARA: 4.4.(c), 5.13.7.2.1

26. RELIABILITY

100 26.	MIL-STD-781	PARA: 1, 4, 5
	MIL-D-23222A	PARA: 3.17

REFERENCES

CHAPTER 101: OPERATION IN VARIOUS CLIMATIC CONDITIONS

101	MIL-STD-210	CLIMATIC INFORMATION TO DETERMINE DESIGN AND TEST REQUIREMENTS FOR MILITARY SYSTEMS AND EQUIPMENT
	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-STD-850	AIRCREW STATION VISION REQUIREMENTS FOR MILITARY AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-C-5778	COVERS, AIRCRAFT COMPONENTS, GENERAL REQUIREMENTS FOR
	MIL-W-006729	WATERTIGHTNESS OF AIRCRAFT, TESTING, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS

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1. TEMPERATURE LIMITS

101 1.	MIL-STD-210D	PARA: 4.5, 5
	MIL-STD-810	PARA: FIG 1

2. HUMIDITY LIMITS

101 2.	MIL-STD-210D	PARA: 4.5, 5
	MIL-STD-810D	PARA: FIG 1

3. CLEAR VISION

101 3.	MIL-STD-850B	PARA: 4
	MIL-STD-1472C	PARA: 5.12.5
	MIL-D-23222A	PARA: 3.12.12, 3.123

4. FLIGHT IN ICING CONDITIONS

101 4.	MIL-STD-210D	PARA: 4.5, 5
	MIL-STD-810D	PARA: FIG 1
	MIL-D-23222A	PARA: 3.12.5

5. WEATHERPROOFING

101 5.	MIL-STD-210D	PARA: 4.5
	MIL-STD-810D	PARA: FIG 1
	MIL-C-5778	
	MIL-W-6729	PARA: 3
	MIL-W-006729	

6 DUST AND SAND PROOFING

101 6.	MIL-STD-210D	PARA: 4.5
	MIL-STD-810D	PARA: FIG 1

CHAPTER 102: EMERGENCY ESCAPE

102	MIL-STD-872	TEST REQUIREMENTS AND PROCEDURES FOR AIRCRAFT EMERGENCY GROUND AND DITCHING ESCAPE PROVISIONS
	MIL-STD-1290	LIGHT FIXED AND ROTARY WING AIRCRAFT CRASH RESISTANCE
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-L-18276	LIGHTING, AIRCRAFT INTERIOR, INSTALLATION OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	AFGS-87235	EMERGENCY ESCAPE, AIRCRAFT

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1. INTRODUCTION

102 1.	MIL-STD-1472C	PARA: 4.4.(i), 5.14.4

2. GENERAL PRINCIPLES

102 2.	MIL-STD-1472C	PARA: 4.4.(i), 5.14.4.1, 5.14.4.2.2, 5.14.4.2.5
	AFGS-87235	

3. EMERGENCY EXITS

102 3.	MIL-STD-1290	PARA: 5.4
	MIL-STD-1472C	PARA: 4.4.(i), 5.14.3.1.1, 5.14.4.1.2, 5.14.4.2
	MIL-D-23222A	PARA: 3.12.11

4. EMERGENCY ESCAPE/EVACUATION ILLUMINATION

102 4.	MIL-STD-1472C	PARA: 5.14.4.2.1
	MIL-L-18276	PARA: 3.7

5 ABANDON AIRCRAFT COMMAND SIGNAL

102 5.	MIL-STD-1472C	PARA: 4.4.(i), 5.14.4

6. EMERGENCY LIGHTING

102 6.	MIL-STD-1290	PARA: 4.3, 5.1.1.3, 5.1.6
	MIL-STD-1472C	PARA: 5.14.4.2.3, 5.14.4.3.8.3

7. TESTS

102 7.	MIL-STD-872	PARA: 4.2.1, 4.2.2
	MIL-STD-1290	PARA: 4.5
	MIL-D-23222A	PARA: 3.3.5.2.4, 3.12.11

REFERENCES

CHAPTER 103: OPERATIONAL COLOURING AND MARKINGS

103.	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
	MIL-C-8779	COLORS, INTERIOR, AIRCRAFT, REQUIREMENTS FOR
	MIL-M-18012	MARKINGS FOR AIRCREW STATION DISPLAYS, DESIGN AND CONFIGURATION OF

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1. INTRODUCTION

103 1.	MIL-STD-1472C	PARA: 5.7.9, 5.5.1.3
	MIL-STD-2161	PARA: 1
	MIL-C-8779D	PARA: 3.3

2. EMERGENCY CONTROLS

103 2.	MIL-STD-250D	PARA: 4.5, 4.6
	MIL-STD-1472C	PARA: 5.5, 5.7.9
	MIL-C-8779D	PARA: 3
	MIL-M-18012B	PARA: 3.1.9

3. CANOPY OR HATCH CONTROLS

103.3

4. EMERGENCY EXITS AND THEIR OPERATING CONTROLS

103 4.	MIL-STD-250D	PARA: 4.5
	MIL-STD-1472C	PARA: 5.5.5, 5.7.9, 5.14.4.2.1
	MIL-C-8779D	PARA: 3

5. LIFERAFT RELEASE AND FLOTATION CONTROLS

103 5.	MIL-STD-1472C	PARA: 5.5.5, 5.7.9
	MIL-C-8779C	PARA: 3

6. AIRFRAME NOTICES

103 6.	MIL-STD-1472C	PARA: 5.5.1.1, 5.5.1.2, 5.5.3, 5.5.4, 5.7.9
	MIL-C-8779D	PARA: 3

7. COLOUR STANDARDS AT CREW STATIONS

103 7.	MIL-STD-1472C	PARA: 5.7.9
	MIL-C-8779D	PARA: 3.4

8. MAINTENANCE AREAS

103 8.	MIL-STD-1472C	PARA: 5.7.9
	MIL-C-8779D	PAPA: 3.5

REFERENCES

CHAPTER 104: VIEW AND CLEAR VISION

104	MIL-STD-850	AIRCREW STATION VISION REQUIREMENTS FOR MILITARY AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-T-5842	TRANSPARENT AREAS, ANTI-ICING, DEFROSTING, DEFOGGING SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-H-46855	HUMAN ENGINEERING REQUIREMENTS FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-R-81367	RAIN REMOVAL SYSTEMS, AIRCRAFT WINDSHIELD, JET AIR BLAST
	MIL-R-81589	RAIN REPELLENT FLUID APPLICATION SYSTEM, AIRCRAFT WINDSHIELD

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1. INTRODUCTION

104 1.	MIL-STD-850B	PARA: 1.1
	MIL-STD-1472C	PARA: 5.14.1.1

2. GENERAL REQUIREMENTS FOR AIRCREW STATIONS

104 2.	MIL-STD-850B	PARA: 4.1.1, 4.1.3
	MIL-STD-1472C	PARA: 5.14.1.1
	MIL-D-23222A	PARA: 3.12.13

3. BASIC REQUIREMENTS FOR PILOT'S STATIONS

104 3.	MIL-STD-850B	PARA: 8.1
	MIL-STD-1472C	PARA: 5.14.1.1

4. VISION REQUIREMENTS FOR PILOTS

104 4.	MIL-STD-850B	PARA: 8
	MIL-STD-1472C	PARA: 5.14.1.1

5. BASIC OPERATIONAL REQUIREMENTS FOR CLEAR VISION

104 5.	MIL-STD-850B	PARA: 3.4
	MIL-T-5842B	PAPA: 3.4, 3.4.1, 3.4.5
	MIL-D-23222A	PARA: 3.12.13
	MIL-H 46855B	PARA: 3.2.2.3
	MIL-R 81367	PARA: 3
	MIL-R-81589	PARA: 3

REFERENCES

CHAPTER 105: CREW STATIONS - GENERAL REQUIREMENTS

105	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-411	AIRCREW STATION SIGNALS
	MIL-STD-783	LEGENDS FOR USE IN AIRCREW STATIONS AND ON AIRBORNE EQUIPMENT
	MIL-STD-875	TYPE DESIGNATION SYSTEMS FOR AERONAUTICAL AND SUPPORT EQUIPMENT
	MIL-STD-1288	AIRCREW PROTECTION REQUIREMENTS, NON-NUCLEAR THREAT
	MIL-STD-1290	LIGHT FIXED AND ROTARY WING AIRCRAFT CRASH RESISTANCE
	MIL-STD-1333	AIRCREW STATION GEOMETRY FOR MILITARY AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
	MIL-L-5057	LIGHTS, INSTRUMENT, INDIVIDUAL, GENERAL SPECIFICATION FOR
	MIL-H-5364	HARNESS, SHOULDER SAFETY, GENERAL SPECIFICATION FOR
	MIL-E-5627	EXTINGUISHERS, FIRE, CARBON DIOXIDE, PORTABLE
	MIL-L-5667	LIGHTING, EQUIPMENT, AIRCRAFT INSTRUMENT PANEL, GENERAL SPECIFICATION FOR, INSTALLATION OF
	MIL-L-6484	LIGHTS, COCKPIT, UTILITY, AIRCRAFT GENERAL SPECIFICATION FOR
	MIL-L-6503	LIGHTING EQUIPMENT, AIRCRAFT GENERAL SPECIFICATION FOR INSTALLATION OF
	MIL-S-6904	SIGNALS, EMERGENCY WARNING, GENERAL SPECIFICATION FOR
	MIL-P-7788	PANELS, INFORMATION, INTEGRALLY ILLUMINATED
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-M-18012	MARKINGS FOR AIRCREW STATION DISPLAYS, DESIGN AND CONFIGURATION OF
	MIL-L-18276	LIGHTING, AIRCRAFT INTERIOR, INSTALLATION OF
	MIL-N-18307	NOMENCLATURE AND IDENTIFICATION FOR AERONAUTICAL SYSTEMS INCLUDING JOINT ELECTRONICS TYPE DESIGNED SYSTEMS AND ASSOCIATED SYSTEMS
	MIL-I-18373	INSTRUMENTS AND NAVIGATION EQUIPMENT, AIRCRAFT, INSTALLATION OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-K-25049	KNOBS, CONTROL, ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-C-25050	COLORS, AERONAUTICAL LIGHTS AND LIGHTING EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-S-25073	SEAT, AIRCRAFT
	MIL-L-25467	LIGHTING, INTEGRAL, RED, AIRCRAFT INSTRUMENT, GENERAL SPECIFICATION FOR
	MIL-L-27160	LIGHTING, INSTRUMENT, INTEGRAL, WHITE, GENERAL SPECIFICATION FOR
	MIL- S-38039	SYSTEMS, ILLUMINATED, WARNING, CAUTION, AND ADVISORY, GENERAL SPECIFICATION FOR
	MIL-H-46855	HUMAN ENGINEERING REQUIREMENTS FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-S-58095	SEAT SYSTEM, CRASHWORTHY, NON-EJECTION, AIRCREW, GENERAL SPECIFICATION FOR
	MIL-R-81729	RESTRAINT SYSTEMS, AIRCREWMANS
	MIL-S-81771	SEATS; AIRCREW, ADJUSTABLE; AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-81774	CONTROL PANEL, AIRCRAFT, GENERAL REQUIREMENTS FOR

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	MIL-STD-1789	SOUND PRESSURE LEVELS IN AIRCRAFT
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1.	INTRODUCTION		
105 1.		MIL-STD-250D	PAPA: 1.1
		MIL-STD-1333B	PARA: 1.1
		MIL-STD-1472C	PARA: 5.14, 5.14.2
		MIL-D-23222A	PARA: 3.12.12
		MIL-H-46855B	PARA: 3.2.2.1.1, 3.2.2.3
2.	NOMENCLATURE		
105 2.		MIL-STD-783D	PARA: 5
		MIL-STD-875	
		MIL-STD-1472C	PARA: 5.5
		MIL-M-18012B	PARA: 3.1.2.3, 3.1.2.4
		MIL-N-18307	
		MIL-D-23222A	PARA: 3.12.12
3.	LAYOUT		
105 3.		MIL-STD-250D	PARA: 4.9
		MIL-STD-1333B	PARA: 4.1
		MIL-STD-1472C	PARA: 4.4, 5.1.2, 5.14
		MIL-D-23222A	PARA: 3.12.12
		MIL-H-46855B	PARA: 3.2.2.3
4.	FREEDOM OF MOVEMENT		
105 4.		MIL-STD-1333B	PARA: 4.5.1.2, 4.6
		MIL-STD-1472C	PARA: 4.4, 5.14
		MIL-D-23222A	PARA: 3.12.12
		MIL-H-46855B	PARA: 3.2.2.3
5.	NOISE		
105 5.		MIL-STD-1472C	PARA: 4.4, 5.14
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		MIL-D-23222A	PARA: 3.12.2.5
		MIL-H-48655B	PARA: 3.2.2.3
6.	INTERCOMMUNICATION		
105 6.		MIL-STD-1472C	PARA: 4.4
		MIL-D-23222A	PARA: 3.14.1.2, 3.14.1.7
		MIL-H-46855B	PARA: 3.2.2.3
7.	DESIGN OF SEATS AND HARNESSES		
105 7.		MIL-STD-1290	PARA: 5.2.1.2.3, 5.3.2.1
		MIL-STD-1472C	PARA: 4.1.4, 4.4, 5.14.2.4
		MIL-T-8679	PARA: 3.2.9.2.1

105 7.	MIL-D-23222A	PARA: 3.12.12
	MIL- S-25073	PARA: 3.5.3
	MIL-H-46855B	PARA: 3.2.2.3
	MIL-S-58095A	PARA: 3.4, 3.12
	MIL- S-81771	PARA: 3.7, 3.9

8. RESTRAINT AND PARACHUTE HARNESS FOR AIRCREW

105 8.	MIL-STD-1472C	PARA: 4.4, 5.14.2.4
	MIL-H-5364	
	MIL-D-23222A	PARA: 3.12.12
	MIL-H-46855B	PARA: 3.2.2.3
	MIL-S-58095A	PARA: 3.4
	MIL-R-81729	

9. ARMOUR PROTECTION

105 9.	MIL-STD-1288	PARA: 5.3.3

10. COCKPIT STRUCTURE

105 10.	MIL-STD-1333B	PARA: 4.1
	MIL-STD-1472C	PARA: 4.4, 5.14
	MIL-D-23222A	PARA: 3.12.12
	MIL-H-46855B	PARA: 3.2.2.3

11. CONTROLS

105 11.	MIL-STD-250D	PARA: 4.4, 4.5, 4.6
	MIL-STD-1333B	PARA: 4.7.1, 4.7.2, 5.1
	MIL-STD-1472C	PARA: 4.2, 5.1, 5.4, 5.5, 5.14
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	MIL-I-18373A	PARA: 3.15.2
	MIL-D-23222A	PARA: 3.12.12
	MIL-K-25049	
	MIL-H-46855B	PARA: 3.2.2.3
	MIL-C-81774	PARA: 3, 3.8, 3.10

12. WARNING CAUTIONARY AND ADVISORY SIGNALS

105 12.	MIL-STD-411D	PARA: 5.1, 5.2
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	MIL-D-23222A	PARA: 3.12.12
	MIL-S-38039	PARA: 3
	MIL-H-46855B	PARA: 3.2.2.3

13. INDICATORS

105 13.	MIL-STD-250D	PARA: 4.5
	MIL-STD-411D	PARA: 5.1, 5.3
	MIL-STD-1472C	PARA: 5.2.3, 5.2.6.5, 5.5, 5.14
	MIL-M-18012B	PARA: 3.2.1
	MIL-L-5057F	PARA: 3.4
	MIL-L-18276	PARA: 3.1.4
	MIL-I-18373A	PARA: 3.5.1
	MIL-D-23222A	PARA: 3.9, 3.12.12, 3.14.1.12
	MIL-C-25050	PARA: 3
	MIL-S-38039	PARA: 3.5.3
	MIL-H-46855B	PARA: 3.2.2.3
	MIL-C-81774	PARA: 3.5, 3.6.3.1, 3.7

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105 14.	MIL-STD-250D	PARA: 4.5
	MIL-STD-411D	PARA: 5.1
	MIL-STD-1472C	PARA: 5.14
	MIL-D-23222A	PARA: 3.12.12, 3.14.2
	MIL-C-25050	PARA: 3
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15. LIGHTING

105 15.	MIL-STD-1472C	PARA: 4.4, 5.1.1.5, 5.2.1, 5.2.2, 5.2.6, 5.8.2
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	MIL-L-18276	PARA: 3.5, 3.6
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	MIL-H-46855B	PARA: 3.2.2.3
	MIL-C-81774	PARA: 3.6

16. BLACKOUT CURTAINS

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17. FIRST AID KITS

105 17.	MIL-STD-1472C	PARA: 4.4
	MIL-STD-2161	PARA: 5.2.2.10.6.11

18. PERSONAL SURVIVAL PACKS

105 18.	MIL-STD-1472C	PARA: 4.4, 5.14
	MIL-D-23222A	PARA: 3.12.10

19. AIRCREW OVERNIGHT KIT STOWAGES

105 19.	MIL-H-46855B	PARA: 3.2.2.3

20. HAND FIRE EXTINGUISHER

105 20.	MIL-STD-2161	PARA: 5.2.2.10.6.9
	MIL-E-5627	PARA: 3.4

21. AXES AND ASBESTOS GLOVES

105 21.	MIL-STD-2161	PARA: 5.2.2.10.6.8, 5.2.2.10.6.10

22. SANITATION

105 22.	MIL-L-38779	
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CONTROLLED DISTRIBUTION:

	MIL-STD-1776	AIRCREW STATIONS AND PASSENGER ACCOMMODATIONS
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	MIL-STD-1789	SOUND PRESSURE LEVELS IN AIRCRAFT
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110 REF	MIL-L-6503H	PARA: 6.6
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	MIL- H -46855	HUMAN ENGINEERING REQUIREMENTS FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
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	MIL-HDBK-336-2	SURVIVABILITY, AIRCRAFT, NON-NUCLEAR, AIRFRAME, VOLUME 2
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	MIL-STD-1288	AIRCREW PROTECTION REQUIREMENTS, NON-NUCLEAR WEAPONS THREAT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NON-NUCLEAR SURVIVABILITY PROGRAM
	MIL-STD-2072	SURVIVABILITY, AIRCRAFT, ESTABLISHMENT AND CONDUCT OF PROGRAMS
	MIL-STD-2089	AIRCRAFT, NON-NUCLEAR SURVIVABILITY TERMS

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113	MIL-STD-1288	AIRCREW PROTECTION REQUIREMENTS, NON-NUCLEAR WEAPONS THREAT
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4. REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

113 4.

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PART 2

STRUCTURAL STRENGTH AND DESIGN FOR FLIGHT

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*In preparation

CHAPTER 200

STATIC STRENGTH AND DEFORMATION

1 DEFINITIONS

1.1 This chapter states the requirements for static strength and freedom from deformation which might adversely affect airworthiness, handling or maintenance.

1.2 The requirements apply to airframes and mechanical components, including undercarriages and systems.

1.3 Critical design cases, corresponding to the most severe authorised use of the rotorcraft, must be identified and the loads associated with each case must be traced through the structure.

1.4 All Grade A details must be identified and for each of them an allowable stress or strength must be determined. This would normally be the appropriate allowable ultimate stress or strength determined on a 'B' value basis. Refer to Chapter 400, para 2.2 for definition of Grade A details.

1.5 The allowable ultimate stresses or loads must not be exceeded at the design ultimate load, normally 1.5 times the maximum load in the design case (the design limit load).

1.6 The structure as a whole must be tested to the design ultimate load and supplementary evidence must be provided that 'B' values are not exceeded. It is, of course, for the designer to decide what material allowables should be associated with his own design procedure in order that this requirement is satisfied.

1.7 In addition, it must be shown that no detrimental deformations occur at loads of at least 75% of the design ultimate load.

1.8 It must be shown that these proof and ultimate requirements can be satisfied under the most adverse loading and environmental conditions. For composites these conditions apply with the structure at the most adverse moisture level that is expected to occur during the life.

1.9 Flight load measurements must be made at an early stage to confirm design assumptions.

1.10 In service a programme of operational loads measurement must be maintained to ensure that any new critical loading actions, are identified so that appropriate action can be taken.

1.11 All compliance procedures and judgements are subject to the approval of the Rotorcraft Project Director (in conjunction with Airworthiness Division, RAE).

1.12 Definitions are given in para 7.

1.13 Advice on compliance with these requirements is given in Leaflets 200/1 to 200/4.

2 DESIGN CASES

2.1 For the purpose of design and of demonstration of compliance with the design requirements, a limited number of discrete design cases are prescribed for which the loads and other conditions are specified. They are defined in the appropriate chapters of this publication.

2.2 The loads associated with each case must be traced through the structure far enough to ensure that the requirements are satisfied everywhere. Many of the cases overlap, and the whole structure need not be analysed for every one. When it can be shown that a particular case is covered by other more stringent requirements, it need not be considered any further.

2.3 There may be cases that are more demanding of some designs than those defined in this publication. When such a case is identified it must be defined and then compliance with the requirements must be demonstrated in that case also.

3 THE ULTIMATE STRENGTH AND PROOF REQUIREMENTS

3.1 In general the ultimate factor shall be 1.5 and the proof factor shall be at least 1.125. In some particular design cases different values are required and they are given in the appropriate chapters of this publication.

3.2 Until the design proof load is reached, no Grade A items shall sustain deformation detrimental to safety, and moving parts essential to safety shall function satisfactorily. After removal of the design proof load no effects of loading shall remain that might reasonably cause the rotorcraft to be deemed unserviceable.

3.3 Until the design ultimate load is exceeded, no Grade A item shall collapse, and the stress, load or strain at each detail shall not exceed the static allowable value.

4 SUBSTANTIATION OF THE STATIC ALLOWABLE STRESS FOR GRADE A DETAILS

4.1 All static allowable values must be based on nominal dimensions unless adverse tolerances significantly affect the strength.

4.2 The allowable must be determined under the most adverse loading and environmental conditions arising in the critical design case for the detail in question.

4.3 The derivation must be based upon a strength no greater than the 'B' value for the critical failure mode of the material from which the detail is made, using an acceptable method of structural analysis.

4.4 In general the material 'B' value for each mode of failure should be derived in accordance with DEF STAN 00-932. Should this be impractical the value shall be based upon at least 15 coupons. These must be tested under the most adverse environmental conditions if these conditions significantly affect the strength.

4.5 When only 15 material coupons are used these must comprise at least 5 samples from each source of supply and at least 3 batches of material from each source. Similar representative samples must be used if larger numbers of coupons are tested.

4.6 Exceptionally, it may be acceptable to allocate allowable values to individual batches of material: in circumstances where the mean strength of the material varies from batch to batch, but the coefficient of variation (cv) (based on individual batches) remains reasonably constant, the 'B' value for a detail in a given batch of material must be reduced if the mean of the batch in question is lower than that of the batch on which the reference 'B' value is based in accordance with (paras 4.3 and 4.8). No increase in the reference 'B' value is permissible unless it can be shown that other failure modes do not thereby become significant.

4.7 METHOD OF STRUCTURAL ANALYSIS

4.7.1 The method of structural analysis must be substantiated by test for each class of structural detail for which the method has not already been verified.

4.7.2 To verify the analysis at least one specimen representing the detail in question must be tested under the most adverse loading and environmental conditions specified for design. The value of strength so obtained must be reduced in order to obtain an acceptable estimate of the 'B' value for the test specimen (Table 1), and where appropriate a further allowance must be made for differences between the test specimen and the structural detail itself. The size of the reduction factor of Table 1 depends upon the failure mode that is observed, the variability associated with this failure mode and the number of coupons used in establishing that variability (paras 4.4 and 4.5). In general, the larger the sample of coupons upon which the variability is based the smaller the reduction factor.

4.7.3 It is possible that the overall airframe static test adequately exercises some of the Grade 'A' details and a separate detail test may not be necessary. Indeed it may be difficult to apply the correct boundary conditions to the detail test.

5 DEMONSTRATION OF COMPLIANCE WITH THE ULTIMATE STRENGTH AND PROOF REQUIREMENTS FOR COMPLETE STRUCTURES OR COMPONENTS

5.1 Usually compliance must be demonstrated by testing a complete structure or component. Exceptionally such a test may be deemed unnecessary. In these circumstances an appropriate allowance must be made for the added uncertainties that are involved.

5.2 The test programme and the results of the tests must be discussed with the Rotorcraft Project Director (in conjunction with Airworthiness Division, RAE).

5.3 To allow for the variability in strength between nominally identical items, a test factor appropriate to each test shall be determined from Table 2. The factors given in Table 2 correspond to a 'B'level of assurance normalised to the variability test factor of 1.0 that is customary for metal structures.

6 MEASUREMENT OF LOADS ON ROTORCRAFT STRUCTURES

6.1 Measurements of the loads and temperatures on the rotorcraft structure during manoeuvres in flight and on the ground, to an extent to be agreed with the Rotorcraft Project Director (in conjunction with Airworthiness Division, RAE), shall be made on one or more early development rotorcraft and on Service rotorcraft as required by the appropriate chapters of this publication.

7 DEFINITIONS

7.1 COUPON The simplest form of test specimen suitable for obtaining the properties of a material in a particular mode of failure.

7.2 STRUCTURAL DETAIL A structural detail is part of a component such as a joint, panel or structural or mechanical assembly.

7.3 THE DESIGN LIMIT LOAD is the greatest load that is expected to occur during the specified life in any design case.

7.4 THE DESIGN PROOF LOAD is the product of the design limit load and the proof factor.

7.5 THE DESIGN ULTIMATE LOAD is the product of the design limit load and the ultimate factor.

7.6 THE STATIC TEST FACTOR is the ratio between the load that an item is required to withstand under test and the design ultimate load. The factor also applies to the proof load.

7.7 THE STATIC ALLOWABLE VALUE OF STRESS is the 'B' value of stress for a structural detail under the adverse loading and environmental conditions arising in the critical design case for the detail. For composites these conditions apply with the detail at the most adverse moisture level that is expected to occur during the life. (The 'B' value is that below which not more than 1 in 10 items will fail with 95% confidence).

8 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE (Chapter 112)

8.1 The rotorcraft structure should be designed to be tolerant to Battle Damage and to be easily repairable.

8.2 Drain holes and drip fences should be provided wherever needed.

TABLE 1
FACTORS BY WHICH THE MEAN STRENGTH OF DETAILS OR ELEMENTS MUST BE REDUCED TO OBTAIN A 'B' ALLOWABLE VALUE:
WHERE APPROPRIATE ENVIRONMENTAL DEGRADATION MUST BE INCLUDED IN THE TESTS:
THESE FACTORS APPLY TO ALL GRADE A DETAILS AND TO ALL MATERIALS

	Estimated population cv from n coupon tests i.e $cv = \frac{\sqrt{\frac{\sum (x - \bar{x})^2}{(n - 1)}}}{\bar{x}}$	Sample size of coupon tests n, used to estimate population cv in observed failure mode														
		15					30					100 or more				
		Number of element tests N														
		1*	2	3	5	10	1*	2	3	5	10	1*	2	3	5	10
Characteristic cv in observed failure mode	3% or less	1.14	1.11	1.10	1.09	1.08	1.12	1.10	1.09	1.08	1.07	1.10	1.09	1.08	1.07	1.06
	5%	1.26	1.20	1.18	1.16	1.14	1.20	1.17	1.16	1.14	1.13	1.18	1.15	1.14	1.12	1.11
	7.5%	1.37	1.31	1.28	1.26	1.23	1.32	1.27	1.24	1.22	1.20	1.28	1.23	1.21	1.19	1.17
	10%	1.53	1.44	1.40	1.36	1.32	1.45	1.37	1.34	1.31	1.27	1.39	1.32	1.29	1.27	1.24
	15%	1.89	1.74	1.68	1.61	1.55	1.75	1.62	1.57	1.51	1.46	1.63	1.53	1.48	1.44	1.39

* Normally at least 2 tests must be done so that any gross inconsistency is likely to be revealed

TABLE 2

TEST FACTORS BY WHICH THE MEASURED STRENGTH OF COMPLETE STRUCTURES FOR COMPONENTS MUST EXCEED THE DESIGN ULTIMATE LOAD IN ORDER TO ALLOW FOR THE INFLUENCE OF VARIABILITY: WHERE APPROPRIATE A SEPARATE ALLOWANCE MUST BE MADE FOR ENVIRONMENTAL DEGRADATION

Estimated cv for type of complete item under test*	Number of tests upon which the mean value is estimated	
	1†	2
5% or less	1.00	1.00
8%	1.10	1.05
10%	1.15	1.10
15%	1.33	1.25

* To be agreed with the Aeroplane Project Director (in conjunction with Airworthiness Division RAE). It is customary to use a figure of 5% for conventional metal structures.

† Wherever practical at least 2 tests must be done so that any gross inconsistency is likely to be revealed.

LEAFLET 200/1

STATIC STRENGTH AND DEFORMATION

PRINCIPLES UNDERLYING THE REQUIREMENTS

1 INTRODUCTION

1.1 The purpose of this leaflet is to outline the basic principles underlying strength and deformation requirements in this publication and, in particular to show why these requirements are sometimes expressed in terms of proof loads, sometimes ultimate loads, and sometimes both.

2 THE ULTIMATE STRENGTH AND PROOF REQUIREMENTS

2.1 A rotorcraft in service is subject to an enormous variety of combinations of loads and other conditions, and it is obviously impracticable and unnecessary to consider them all. The most adverse of these are identified in the form of design cases, which must be used in demonstrating compliance with the requirements.

2.2 In the majority of design cases, the severity of the loads that may arise in service is not limited naturally, and limiting conditions have to be chosen for design purposes. A classic example is provided by the symmetric flight case. There are usually no means to prevent the exceedance of either the speeds or the normal accelerations that are permissible for a rotorcraft in symmetrical flight. In choosing the limiting conditions for design, use is made of past experience of the speeds and accelerations needed for the operational duty of the type, of the predicted flight characteristics and performance of the rotorcraft, and of the way structure mass is likely to depend on the severity of the chosen conditions.

2.3 Every rotorcraft in service is, of course, subject to flying limitations that prohibit its use beyond the limiting conditions chosen for design.

2.4 There are several causes of uncertainty that must be taken into account in designing the rotorcraft to withstand the chosen limiting conditions. First, the methods used in the evaluation of the aerodynamic loads and in the stress analysis are inexact. Second, every rotorcraft is expected to encounter the chosen limiting conditions in each design case only once, or at most a few times, during its life. In other words the rotorcraft is designed to sustain the maximum load in an average spectrum, of loads. Some rotorcraft will inevitably experience loads greater than the average and will, therefore, experience a maximum load that is greater than the limit load. Finally, when the rotorcraft enters service, the most severe conditions under which it is actually used may differ from the design cases. For all these reasons the rotorcraft must be designed to have, in each design case, an ultimate strength that is more than would be needed just to sustain the limit load.

2.5 The uncertainty caused by variability in the strength of materials and structural features is taken into account separately, in 3 ways.

2.5.1 First, the designer is encouraged to use 'B' allowable values of stress in design (see Chapters 400 and 401 and DEF STAN 00-932) because they are founded on a standardised test procedure and give a uniform statistical assurance of the strength of materials. No such uniformity is associated with the minimum specification or S values although these typically were somewhat lower than 'B' values and closer to 'A' values. A problem in accepting a move from 'S' to 'B' values was a natural reluctance to move from an allowable value that would be reached or exceeded by 99 out of 100 items to a higher value that would be reached or exceeded by 9 out of 10 items. however, it was noted that the allowable value was associated with the design ultimate load which itself had an extremely low probability of occurrence. It was agreed that the probability of structural failure in service was influenced predominantly by considerations such as the identification of design cases, choice of design limit load and the value of the static factor of safety.

2.5.2 Second, any test of a structural feature or test element is subject to a test factor. The test factors required by Table 1 of Chapter 200 are related to the measured variability in the strength of the materials from which the features or elements are made; relevant data on variability will usually be found in DEF STAN 00-932. The test factors are calculated, by means of statistical theory, to give assurance, with 95% confidence, that 90% of similar items will equal or exceed the required strength. It is emphasised that separate allowance must be made for any uncertainties arising from differences between elements and the structural features that they represent.

2.5.3 Finally the designer must make separate allowance for environmental degradation and demonstrate that the allowable values (or S values) of ultimate stress are not exceeded under the design ultimate load. The most appropriate allowance for degradation can be obtained by conducting tests on elements under degraded conditions.

2.6 Though the design ultimate strength defines the margin of strength that must be provided, it is not sufficient on its own to ensure that there is also a margin of airworthiness. This margin of airworthiness is defined by the design proof requirement.

2.7 The proof requirement defines the minimum conditions up to which the airworthiness and serviceability of the rotorcraft can be depended upon. The designer should ensure, by design and tests as appropriate, that loads up to the design proof load will neither impair safety nor necessitate servicing other than to predetermined and defined schedules acceptable to the operator. Thus under the design proof load and after its removal there should be no loosening or pulling of rivets, structural deformation which might cause unacceptable redistribution of aerodynamic loads, jamming or undue slackness of controls, or any other sign of structural distress which might cause a service engineer in the field to have doubts about the serviceability of the rotorcraft. In this matter, tests are the only trustworthy guide. Moreover a proof test should not be stopped merely because the design proof load has been reached; it should be continued until the real proof load has been found, beyond which the proof requirement cannot be satisfied. Two or three tests at increasing loads may be necessary (see Leaflet 200/2).

2.8 Since the real proof strength defines the conditions up to which the structural airworthiness can be depended upon, it is usually a more meaningful measure of the capability of a design than the ultimate strength that is achieved. Beyond the proof strength, catastrophe could result from structurally induced causes such as jamming of controls or unstable aerodynamic loadings. For these reasons, greater emphasis will be placed on establishing the real proof load.

3 SPECIAL CASES

3.1 In most design cases both strength and deformation are regarded as important. It seldom happens that both requirements can be satisfied exactly; usually a margin above the standard required by one requirement must be provided in order to comply with the other. Nevertheless when both requirements are imposed, the designer is expected to demonstrate compliance with them both.

3.2 In some exceptional design cases, deformation is all that matters and a design ultimate requirement is superfluous.

3.3 In a few other cases, strength is critical but deformation is not important, and a design ultimate requirement is imposed without a design proof requirement. An example is that of seats and harnesses at crash stations, and their supporting structures. In a crash landing that would do serious damage to the rotorcraft, some distortion of the seat and harness attachments would be trivial, (and may be beneficial in absorbing the energy of impact), but the crew must remain safely restrained within reasonable physiological limits. Accordingly an ultimate factor of 1.0 is required in the crash landing case but there is no proof requirement (see Chapter 307).

3.4 There are a few other special cases in which the respective values of the proof and ultimate factors are lower than the usual 1.125 and 1.5. These cases represent rare events such as malfunctions of systems and operation of systems under exceptional conditions. Provision of the full strength and stiffness that would be necessary to achieve the usual factors in these cases would add mass to the rotorcraft that would penalise it in normal operations. In emergencies that are infrequent and of short duration, and in which the rotorcraft may be at additional risk for other reasons, the higher risks to the rotorcraft and its crew that are associated with the lower factors are regarded as acceptable. Examples are malfunctions of automatic flight control systems.

3.5 Finally there are several cases in which both the proof and ultimate factors are higher than the usual values. They represent the normal operations of systems and installations in which the loads are particularly difficult to estimate. The difficulties are caused principally by transient pressures in hydraulic, pneumatic and fuel systems, by variability in the performance of pressure relief devices, and by variability in the loading on undercarriages. Since these cases represent events that happen frequently, the greater risks associated with the greater uncertainty in estimation of the loads must be compensated by higher factors. Examples are undercarriage retraction and lowering (Chapter 306) and refuelling and defuelling systems (Chapter 701).

4 COMMENTS

4.1 The foregoing paras indicate the diverse nature of the strength requirements and the reasons why emphasis is sometimes placed on a proof requirement, sometimes on an ultimate strength requirement, and why often both are specified. It is clear that the same factors, and the same ratio of proof factor to ultimate factor, cannot be used in all cases without serious loss in structural integrity. This is a consideration that far outweighs the convenience that uniform factors might bring.

REFERENCE

National Agency for Finite Element Methods & Standards (NAFEMS) - Guidelines to Finite Element Practice

LEAFLET 200/2

STATIC STRENGTH AND DEFORMATION
STATIC STRUCTURAL STRENGTH TEST LOADING SEQUENCE

1 INTRODUCTION

1.1 Experience has proved the value of testing representative examples of rotorcraft structures. This Leaflet sets out the aims which should be kept in view when the programme of static tests is being planned and gives recommendations on some points of test procedure which will help to ensure that these aims are achieved.

2 TEST AIMS

2.1 The test procedure adopted should give the maximum practicable amount of information on the structure. In particular, the tests should be conducted with the following aims:

(i) to verify the structural analysis by means of strain, measurements under all test conditions and thereby to demonstrate, for each Grade A structural detail, that the allowable values of stress or strain are not exceeded at the design ultimate load,

(ii) to establish, for selected design cases, the highest load at which the proof requirement is satisfied,

(iii) to demonstrate, in the critical design case established by earlier testing, a strength that is sufficient to show, as far as practicable, that all Grade A structural details have been identified.

3 RECOMMENDED TEST PROCEDURE

3.1 The test item must be fitted with such control runs and systems as are necessary to demonstrate compliance with the proof requirement.

3.2 The test procedure should be phased in with any flight load measurement programme (see Leaflet 1015/1). Any environmental conditioning should meet with the approval of the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

3.3 The following procedure is recommended:

(i) Load in increments up to the design proof load in each critical loading case. Exceptionally, measured flight loads may be available. At the design proof load, check that moving parts essential to safety, function satisfactorily and that there is no structural deformation which might cause unacceptable redistribution of aerodynamic loads. After removal of the load, examine and report on the condition of the specimen noting, with photographs, all structural distress which, if it occurred on a rotorcraft in service, might cause it to be declared unserviceable or not airworthy, (e.g., skin distortion, loose rivets, serious permanent deformation, etc).

Note: The adjustment of the applied loads and temperatures to conform to the prescribed values may not be practicable with some loading systems, if a change in load distribution or in surface temperature is involved. In such cases an estimate of the effect of the difference between the prescribed values and the test values should be made and appropriate corrections applied to the results.

(ii) Continue the loading above the specified design proof load in suitable increments, each time checking that essential moving parts function satisfactorily and there is no unacceptable deformation, before removing all load and examining the condition of the specimen, until the real proof load for that particular loading condition has been established.

(iii) Demonstrate an appropriate test factor from Table 2 of Chapter 200 on the design ultimate load for static tests and 80% of the design ultimate load for residual strength tests. If during these tests, serious damage to the specimen occurs unexpectedly or appears imminent, a decision should be made whether to repair the specimen or proceed to (iv).

(iv) Repeat tests (iii) for each of the other selected loading cases.

4 NOTES

4.1 The aim of the proposed sequence is to secure as much factual information as possible from each structural specimen tested; especially information needed for determining Service clearances on permissible speed and normal acceleration.

4.2 The proposals in this Leaflet are not intended to cover any part of the fatigue test programme (for fatigue testing, see Leaflet 201/1).

4.3 The meaning of structural distress is admittedly vague. Opinions differ between inspectors, Service engineer officers and the test engineer. Therefore the aim must be to record facts so that, as experience accumulates, it will be possible to make the definition more precise.

LEAFLET 200/3

STATIC STRENGTH AND DEFORMATION GENERAL PROTOTYPE TESTS

1 INTRODUCTION

1.1 The requirements of DEF STAN 05-123, Chapter 238 specify that laboratory, ground and flight tests must, be carried out on every prototype rotorcraft (or its components as appropriate) to establish that its vibration characteristics, functioning, and fatigue strength are satisfactory over the whole range of operating conditions.

1.2 Particular tests are specified in the following Chapters:

(i) Chapter 201 and Leaflet 201/1 - Fatigue.

(ii) Chapter 500 - Aero-elasticity.

(iii) Chapter 705 and Leaflet 705/1 - Transmission Systems.

Engine Type Test requirements are given in D Eng, R D 2060 for piston engines, and in D Eng R D 2100 for turbine engines.

2 SEQUENCE OF TESTS

2.1 The various tests are interdependent and some are best run concurrently. The following tests need to be performed:

(i) Rig tests of components and assemblies to prove functioning and reliability, and preliminary fatigue tests.

(ii) Resonance tests of the rotorcraft including stability tests of all powered flying controls (see Chapter 500) (rotor stopped).

(iii) Preliminary investigation of vibration of the whole rotorcraft during ground running, with particular attention to torsional vibrations and whirling of transmission shafting. A preliminary investigation into the vibratory stresses in the rotor system and controls should also be made (rotor running). In addition, ground resonance tests need to be made.

(iv) Ground test, to an agreed schedule, of the rotorcraft in a condition for initial flight.

(v) Preliminary flight tests within a defined envelope of manoeuvres to verify control characteristics.

(vi) Further preliminary flight tests within a defined envelope of manoeuvres to measure vibratory stresses in the rotor system, transmission and controls.

(vii) Additional flight tests within an increasing envelope of manoeuvres up to the complete range of operating conditions during which comprehensive flight strain gauge measurements shall be made.

(viii) Further fatigue tests of components subject to vibratory stresses.

(ix) Ground and flight testing to an agreed schedule to demonstrate satisfactory functional and mechanical reliability of the engine, transmission and rotor systems.

(x) Measurement of in-flight accelerations and rotor head moments.

(xi) Flight tests to establish satisfactory functioning of the automatic flight control system (AFCS).

LEAFLET 200/4

STATIC STRENGTH AND DEFORMATION

STRENGTH OF STRUCTURES UNDER CONDITIONS OF HEATING AND COOLING

1 INTRODUCTION

1.1 When a structure is heated or cooled, the following effects need to be taken into account:

(i) Stresses and strains caused by differential expansion within the structure.

(ii) A temporary loss in strength in the material due to temperature rise.

(iii) Permanent deterioration in material properties due to prolonged exposure to high temperature.

(iv) Creep due to prolonged exposure to temperature and stress.

1.2 This Leaflet contains recommendations on the methods to be used in the design of structures and in the application of safety factors to ensure that an adequate standard of safety is maintained when thermal effects are significant.

1.3 It will not generally be necessary to allow for temperature changes with altitude on rotorcraft, but the effect of temperature changes on transparencies and of high temperatures in structural parts arising from their proximity to the engines should be considered.

2 THERMAL STRESS AND STRAIN

2.1 When a structure* is subjected to a change in temperature each element will tend to change its dimensions by $\lambda\ \alpha\ T$ (where λ is the length in the direction being considered, α is the coefficient of thermal expansion and T is the change in temperature). However if either α or T has different values in different parts of the structure, the change in length of each element may differ from λ or T because of mutual constraints between the various elements. The final length of each element will be such that the conditions of geometrical compatibility and equilibrium of internal forces are satisfied. Thermal stresses and strains are thus induced. The amount of thermal strain in each element is equal to the difference between the actual change in length and $\lambda\ \alpha\ T$, divided by the length.

3 CALCULATIONS

3.1 The process of calculating the magnitude of the thermal stresses and strains and their effect on the ability of the structure to withstand the specified proof and ultimate loads involves a knowledge of:

(i) the flight plan in terms of ambient conditions and of speeds and speed changes,

* For the purpose of this definition, a structure can be either a single member or a number of members joined to make a complete structure.

(ii) the flow pattern over the surfaces of the rotorcraft and the type of flow, laminar or turbulent,

(iii) the heat transfer to the surface allowing for emission and the convective flow from the boundary layer, and

(iv) the temperature distribution in the structure taking account of the heat flow through complex paths and joints and the possible presence of other heat sources and sinks.

From the temperature distribution, the thermal stresses and strains as defined above are obtained.

3.2 Allowance should be made for the variability of thermal properties of materials and structural elements, and it is preferable that design values and their variability be determined by test wherever possible. Otherwise, reasonably conservative values of all properties should be assumed in the calculations.

3.3 The conductivity of joints may depend on the temperature, the pressure between the surface and the surface finish. The variability can be very large, but this does not necessarily result in large variations in stress. In those cases where local stress is sensitive to the conductivity of a particular joint, the variability should be investigated and then taken into account when assessing the results of subsequent strength tests.

3.4 When external loads are present, there are additional stresses and strains which combined with the thermal stresses and strains must satisfy the overall conditions of loading equilibrium and geometrical compatibility. If the total stress is below the elastic limit, the stresses and strains due to the two effects are independent, and can therefore be calculated separately and superimposed. However, if the total stress exceeds the elastic limit, the stresses and strains due to the two effects are interdependent and should therefore be calculated simultaneously.

3.5 The inaccuracies which arise in each stage of the calculation of thermal stresses and strains are likely to be as great as those occurring when estimating the magnitude and distribution of stress due to the external loads. Hence an equal safety factor on thermal effects is necessary and it is recommended that factored conditions should be calculated as described in para 4.

4 FACTORED CONDITIONS

4.1 When the stresses due to thermal effects and to the applied loads are of the same sign, the combined factored conditions for an individual structural member may be obtained by the method of either para 4.2 or 4.3. The case when stresses are opposite in sign is discussed in para 4.5. When a designer wishes to use a different method he should consult the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

4.2 In the first method, the factored thermal strain for a particular member is obtained by calculating the unfactored thermal strain assuming no applied loads and multiplying it by the same proof and ultimate factors as are used for the loads with which the thermal effects are to be combined. This factored thermal strain is then added to the strain corresponding to the factored loads, calculated assuming no thermal strain. This gives the total factored strain, and the corresponding, factored stress is obtained from the stress-strain curve for the particular member. Strains, not stresses, are added because thermal strain cannot produce more stress than that corresponding to the plastic deformation if the elastic limit is exceeded. It is permissible to neglect the effect on adjacent members of factoring the loads and strains, but the combined effect obtained in this way should be treated with reserve as it may result in lower stresses than those obtained if the total forces are brought into equilibrium under factored conditions by the method of para 4.3.

4.3 In the second method, the loading equilibrium and geometrical compatibility of the structure are recalculated in the presence of the factored loads when the thermal expansion coefficient(s) of the material(s) concerned is multiplied by the same proof and ultimate factors as applied to the loads. This calculation will give the total factored strains due to the factored loads and factored thermal effects which the member has to withstand. This method is preferable to that stated in para 4.2 because it is a closer approximation to what would happen in flight or in a laboratory test, but it is recognised that, for complex structure, it may not be practicable.

4.4 In both methods, the thermal strain, rather than the thermal stress, is factored, and this means that the ratio of stress at limit conditions to that at ultimate conditions is usually higher for thermal effects, and therefore also for combined loading and thermal effects, than the usual values associated with loading only. This choice was made on the grounds that an ultimate factor on thermal stress is not justified as far as static strength is concerned, because thermal stresses are relieved when elastic deformation occurs. As a consequence, a thorough investigation is necessary of all fatigue and creep aspects, including, creep buckling and progressive distortion under cyclic temperature and loading conditions.

4.5 Cases where the load and thermal effects are opposite in sign may not be critical because the most severe cases may well be when either one or other of these affects is absent. If there is a critical case when both are present and the load effect is the greater, it may be desirable to disregard any alleviating thermal effects. When the thermal effect is the greater, it appears reasonable to combine it with the load effect using, the appropriate factors. In either case, care should be taken to ensure that an adequate net load is used for design purposes, particularly when the opposing effects are comparable in magnitude.

5 DESIGN CASES

5.1 The chapters of Part 2 define the manoeuvres to be considered in design, specifying rotorcraft speeds in each case, but they do not define those conditions giving rise to thermal stress or strain (i.e., rate of change of speed and time spent at any given speed). For each type of rotorcraft, therefore, it is necessary to examine all flight conditions in which the combination of applied loads and thermal strains are likely to produce critical structural loads. The following conditions should therefore be considered:

(i) Sustained steady flight at all speeds up to V_K *, in each case at the most adverse temperature and altitude conditions of Chapter 101, following the appropriate manoeuvres or gusts of Part 2, taking account of any change in temperature due to speed changes during the manoeuvre or gust.

(ii) The most adverse practicable speed and altitude change between any two sets of conditions within the speed range up to V_D, and within the temperature and altitude conditions of Chapter 101, immediately followed by, or combined with, the appropriate manoeuvres or gusts of Part 2, including excursions from any point within the envelope to V_D and back.

Notes 1 When considering speed changes to or from V_D, it may be necessary in some cases to consider a longer period of time at V_D than that implied by a transient excursion. This will depend on the length of time the rotorcraft is capable of sustaining V_D.

2 The above stressing cases call for the combination of full manoeuvre loads with the most adverse thermal effects. Where such cases are thought to be improbable, appropriate alternative cases should be formulated and these should be discussed with the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

6 THERMAL EFFECTS ON MATERIALS

6.1 Structural materials may suffer a temporary loss in strength while at high temperature, and also permanent deterioration in properties due to prolonged exposure to high temperature (see AvP932). The material properties assumed in calculations should, therefore, take account of the temperature that could occur in the case considered, the time for which it is maintained, and also the temperature history. Both upper and lower limits of the values of the various properties should be considered in cases where the thermal strains may either aggravate or relieve the stress due to the applied loads. In any estimate of structure life under conditions of thermal degradation and/or creep, an appropriate allowance should be made for the variability of material properties, the variability, between individual rotorcraft, of structural temperatures, and the variability of operational conditions and also to cover other unknowns such as thermal cycling effects and the interaction between these and fatigue.

* V_K is the maximum permitted speed for continuous cruising, and may be based on handling qualities, structural strength or engine limitations. For the purpose of these stressing cases, it should be chosen for the particular rotorcraft and need not be greater than the expected Service limiting speed for the type.

6.2 This Leaflet does not call for the provision of a safety margin by an arbitrary increase in the structural temperature, either directly by factoring the temperature or indirectly by factoring the design speed. Thus flight in excess of the design speed could lead to a temperature higher than that considered in design. Although cases of exceeding, the design speed are rare, it is recommended that, as far as practicable, materials should be used which do not suffer a substantial reduction in properties at a temperature slightly higher than that corresponding to the design speed.

7 GROUND TESTS

7.1 When strength tests including thermal effects are necessary to demonstrate compliance with the requirements, the factored loads and, if practicable, the factored thermal strains should be applied. The Method of simulating the factors on thermal strain should be discussed with the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

7.2 It should be noted that, in order to obtain realistic conditions in any structural member, it will casually be necessary to test it as part of the complete structure so that all balancing loads can be developed. When this is not done, results of the tests should be interpreted bearing, in mind any differences between the actual and the applied balancing loads.

8 INSTABILITY FAILURES

8.1 For structures in which failure by instability is possible, thermal strain may cause instability which could be followed by structural collapse under low external loads, whereas neither the thermal strain nor even the design ultimate load would cause collapse when applied separately. It is important, therefore, that this effect be taken into account in all design cases when considering the combination of thermal strain and applied load.

CHAPTER 201

FATIGUE

1 DEFINITIONS (see also Chapter 200)

1.1 THE FATIGUE TEST FACTOR is the ratio between the life or load amplitude that a structure, system or feature sustains under test and the life or load amplitude that can be considered to have been demonstrated for certification purposes: combinations of life and load factors may be acceptable in some circumstances.

1.2 THE RESIDUAL STRENGTH TEST FACTOR is always 1.0 on 80% of the design ultimate load following a suitably factored fatigue test.

1.3 THE ALLOWABLE S-N CURVE FOR A FEATURE is the curve obtained when the estimated mean S-N curve is reduced by factors on life and stress that make allowance for scatter and other uncertainties arising in the estimation of the shape and location of the mean curve.

1.4 THE SAFE LIFE of a structure, system or structural feature, is the period of operation during which the general standard of strength, deformation and stiffness is not appreciably lowered by fatigue.

1.5 AN INSPECTION-DEPENDENT FEATURE is a feature that is capable of maintaining satisfactory strength, deformation and stiffness while containing insidious damage such as may be caused by fatigue, handling, corrosion or manufacture until this is detected by scheduled inspection.

2 SAFE LIVES

2.1 A feature that has a safe life less than the specified life shall be accepted into service only if it can be economically replaced, repaired or treated as "inspection-dependent", and if its use will allow a reasonable standard of serviceability to be maintained.

2.2 Every feature that has a safe life that is known to be less than the specified life shall be declared by the contractor. Its use shall be subject to the approval of the Rotorcraft Project Director, who will be advised by the appropriate Service branches and the Airworthiness Division RAE.

2.3 In establishing the safe life of an element, account shall be taken of the environment experienced in service. In particular, the effects of mechanical loading, temperature and humidity shall be fully accounted for.

3 DEMONSTRATION OF COMPLIANCE

3.1 Usually compliance with the requirements of para 2 shall be demonstrated by test, using the methods described in Leaflet 201/1. Exceptionally, calculations alone may be sufficient, subject to the approval of the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

LEAFLET 201/1

FATIGUE

AN ACCEPTABLE PROCEDURE FOR ESTIMATING FATIGUE LIFE

1 INTRODUCTION

1.1 This Leaflet describes an acceptable procedure for estimating the fatigue life of an item as required by Chapter 201 and shows how compliance with the specified life can be established.

1.2 Compliance will normally be demonstrated by tests on one or more examples of a complete item supported by calculations. Exceptionally, when calculations show a large margin on the required life, a test may be deemed unnecessary.

2 LOAD SPECTRA

2.1 During its specified life, any rotorcraft structure or system is subjected to a history of load fluctuations which occur during the various ground and flight operations. It is usual to divide this loadings history into its separate phases e.g:

(i) loads due to turbulence,

(ii) loads arising from manoeuvres,

(iii) loads arising from taxying, take-off and landing,

(iv) loads due to pressurisation.

2.2 The loading history for each phase of operation is then reduced into individual cycles. If it is expressed as a cumulative frequency spectrum, this is done by division into a number of intervals of load. Care should be taken that the number of load intervals is sufficient to avoid introducing inaccuracies into the estimate of the fatigue damage produced by the spectrum. It is then usual to assume that the loads within each interval all have a range given by the difference between the maximum and minimum loads in that interval. Alternatively the number of maximum and minimum cycles may be deduced from a range mean pairs analysis of a suitable sample of service load measurements, or from direct knowledge of the periodic loading sequence on the item concerned. Hence the number of occurrences of each load amplitude and associated mean stress in the flying time represented by the sequence, can be obtained. It is emphasised that the interpretation of the available loading data and the sequence of loadings chosen for analysis and test can have a very significant influence on the validity of the safe life that is established. In cases of doubt the loading spectrum should be agreed to the satisfaction of the Rotorcraft Project Director (in conjunction with Airworthiness Division RAE).

3 ESTIMATION OF SAFE LIFE BY CALCULATION

3.1 Normally the safe life of all features in the main load paths must be estimated by calculation and the lives so obtained should be recorded by the Design Authority. The process requires that the calculated stresses in the various features are related to existing data on fatigue. Data on the fatigue performance of joints and components similar to those in the structure or system under consideration should be used as far as possible. From these data, an estimated mean S-N curve for each critical feature of the structure or system should be drawn.

3.2 There is considerable variability in the fatigue performance of nominally identical items when they are subjected to the same loading conditions. To maintain safety standards, the weakest item should have a safe life that is not less than the specified life.

3.3 Evidence from many tests indicates that, for most types of structural feature made of aluminium alloys and subjected to spectrum loading, an average value of standard deviation in $\log_{10}$ endurance of 0.12 can be assumed to apply over the steep part of the S-N curve. At higher endurances, where the S-N curve is much flatter, a coefficient of variation in alternating stress is a more appropriate measure of variability in fatigue performance. For aluminium alloys, a coefficient of variation of about 10% is appropriate; for medium strength steels a figure of nearer 7% can be assumed.

3.4 It is usual to assume that the weakest rotorcraft in the fleet will have a fatigue performance which is about 3 standard deviations below the mean. For a normal distribution, this is equivalent to specifying a probability of about 1 in 1000 that the damage that occurs at the mean life will actually occur in service.

3.5 The factor that must be applied to the estimate of the mean S-N curve to obtain the allowable S-N curve upon which the safe life is based must account for uncertainty in the estimate of the mean endurance as well as for the variability in fatigue performance.

3.6 It is customary to reduce the estimated mean curve by a factor on life where the curve is steep, and by a factor on stress where the curve is more or less flat; the two parts should be faired conservatively together to give the allowable curve upon which the calculated life is based. Life factors of the order of 10 and load factors of the order of 2 have usually been used, but each case should be assessed on its merits.

3.7 The fatigue life of a feature depends on its ability to withstand the cumulative effects of the relevant stress spectrum. In order to express this effect in quantitative terms, some form of cumulative damage relationship must be used. Experience has shown that normally an acceptable estimate is provided by Miner's rule. This states hypothetically that the fatigue damage caused by n cycles of a particular alternating stress is proportional to n/N, where N is the number of cycles of the stress that cause failure, and the damage caused by a number of different alternating stresses is proportional to n/N. In estimating the safe life, values of N should be taken from the allowable S-N curve. Failure is assumed to occur when $\sum n/N = 1$. An exception occurs in the case of moderately notched or unnotched features where, in the absence of beneficial residual stress effects, a value of $\sum n/N = 0.2$ should be used unless evidence is presented to support the use of a higher value.

3.8 The factors given in para 3.6 allow for both scatter and uncertainty in calculations. The allowances for uncertainty in calculations are equivalent to about 3 on life where the curve is steep and about 1.25 on strength over the flatter part of the curve. In some cases, therefore, a feature for which the calculated life is inadequate may subsequently be shown to be satisfactory by test.

4 ESTIMATION OF SAFE LIFE BY TEST

4.1 Complete items, representative of production rotorcraft, should be tested. Loading sequences should be applied to represent the significant loads that occur during the operation of the rotorcraft in flight and on the ground.

4.2 The magnitude, distribution and time history of the loads applied should correspond as nearly as possible to service conditions, and for this reason flight-by-flight loading should normally be used. However, the purpose is to demonstrate that the test specimen, which is likely to be of a typical strength, has a sufficient margin of life over the weakest specimen to allow for variability in fatigue performance and this may mean that the shape of the test loading spectrum differs significantly from that for the service condition.

4.3 Sometimes, particularly when the predominant loads are small, block-programmed loading may be acceptable. In programme loading the testing is done by repeating a loading sequence in which the loading is applied at a number of different levels to represent the whole spectrum. Each sequence represents a period of flying time which should be chosen so that failure of a test specimen would be expected to occur on average in no less than 30 of them.

4.4 Acceptable fatigue test factors that allow for variability in fatigue performance and for the number of items tested are given in Table 1 (a). The factors on life are based on the established practice of dividing the mean life of the two halves of a complete wing by 3.33 and equating this to an overall probability of failure of 1 in 1000. The factors on strength are based on the practice of dividing the mean strength obtained from 6 tests on aluminium alloy rotorcraft rotor blades by 1.6; again equating this to an overall probability of failure (typically arising from 4 blades per rotorcraft) of 1 in 1000. The factors for other numbers of tests have been obtained by interpreting these datum conditions in terms of the variabilities stated above in para 3.3 using a level of confidence of 95%.

4.5 In determining the loading needed to prove the safe life of each particular feature in the structure or system, the partial test factor curves should be faired together, as in para 3.6. In general, life factors should be used where this does not lead to an unrealistic testing time. Where load factors are necessary, these must not cause the test loads to exceed the maximum loads in service in either the positive or negative sense.

4.6 Fatigue test factors for transmissions and gearboxes are given in Table 1 (b)

4.7 Close control of loading is necessary in fatigue tests because small changes in loads can cause large changes in fatigue life. In tests on complete structures or systems, or on large components, the applied loads should be accurate to within ±3% of the maximum demanded load and all applied loads should be measured periodically during the test to check repeatability which should be within ±2% of the particular applied load.

5 INSPECTION - DEPENDENT STRUCTURE

5.1 GENERAL

5.1.1 In an inspection - dependent (fail safe) structure, the structural and mechanical design, the inspection techniques and the intervals between inspection must be such that any damage that could eventually become catastrophic will be found with a high probability before it can endanger the rotorcraft. Crack growth times and inspection intervals must be related, so that a crack that has already started but has just failed to be detected at one inspection will have a low probability of reducing the strength, deformation or stiffness of the rotorcraft below a required minimum before the next inspection is made.

5.1.2 The initial crack size should be that which will be found on 9 occasions out of 10 (with 50% confidence) by typical operators using such equipment and techniques that are specified for the location concerned.

5.1.3 Inspection-dependent parts should have safe lives that are not less than half the specified life. Beyond its safe life, each part will be subject to regular inspections for fatigue cracking; it is expected that the whole airframe will receive regular routine inspections. Normally, inspection intervals will be derived by test in accordance with para 5.6.

5.1.4 Parts vulnerable to accidental damage or corrosion, should be subject to regular inspections from the time that the parts enter service.

5.2 DETAIL DESIGN

5.2.1 Inspection-dependent characteristics in a structure or system can be achieved by design of panels, stringers and frames so that any damage is confined within safe boundaries for a given period of operating time that is related to the interval between inspections: multiple-load-path structure can sometimes be equally effective. Alternatively, but preferably in addition, the rate of damage propagation can be kept to a minimum by careful choice of materials and by design to low stress levels.

5.2.2 It should be noted that multiple-load-path structure in which the paths are nominally identical, and subject to the same loading, seldom has acceptable inspection-dependent characteristics.

5.3 EXTENT OF DAMAGE DURING INTERVAL BETWEEN INSPECTIONS

5.3.1 For designs which depend on multiple-load-paths in the structure and crack barriers in the skin, it should be assumed for the purpose of a strength criterion that a crack once started spreads fairly rapidly to its first barrier. Hence the rotorcraft may possibly be operated for almost the full interval between inspections with one load path failed or with a crack at the full length between the crack barriers. In designs which depend on a slow rate of crack propagation, it should be assumed that the crack grows throughout the whole of the inspection interval, and due allowance should be made for the length of crack that remains undetected at the last inspection.

5.4 REQUIRED RESIDUAL STRENGTH OF DAMAGED STRUCTURE

5.4.1 The residual strength of a damaged complete item should never be allowed to fall below 80% of its design ultimate strength. This value of 80% is regarded as the minimum strength that will allow the pilot, who may be unaware of any weakness, to fly the rotorcraft to its full operational standard with reasonable safety during the limited time that the weakness is undetected. The effects of accelerated fatigue in other parts due to load re-distribution should also be considered.

5.5 DEFORMATION AND STIFFNESS OF DAMAGED STRUCTURE

5.5.1 Consideration should be given to the possibility of fatigue damage causing a significant change in deformation or reduction in stiffness of the structure. The criterion for deformation should be that the proof requirement should always be satisfied and for stiffness, that the Service Limiting Speed should always remain safe with an adequate margin.

5.6 TESTING OF INSPECTION-DEPENDENT ITEMS

5.6.1 Testing of inspection-dependent items is usually essential. In some instances, a test of a complete structure or system may be necessary. The most satisfactory test is a fatigue test in which service loading conditions are simulated because it is likely to indicate the site of damage in service and how it will propagate. It will also provide valuable information that will guide inspection techniques and periodicity.

5.6.2 When a crack has started under test conditions, it should be allowed to grow for 3 times the required inspection interval or until the residual strength requirement is only just satisfied. The effectiveness of crack barriers should be established. Complete structures and systems should be repaired at this stage, to avoid unrealistic stress distributions, and testing continued until more cracks appear. At the end of the test, each repair should be removed in turn and the specimen should be subject to a static strength test to demonstrate adequate residual strength and compliance with the proof requirement in the damaged condition. The duration of the test will depend on the performance of the specimen, but it should be sufficient to demonstrate the requirement of para 5.1.3 and establish that the factors of Table 2 are realised. These factors are based on the established practice of dividing the crack growth period by 3 to obtain the inspection interval. The factors for larger numbers of tests have been derived by interpreting this datum condition in terms of the variabilities stated above (para 3.3) using a level of confidence of 95%.

5.6.3 Tests in which the structure is artificially damaged to demonstrate its inspection-dependent properties are not wholly satisfactory because it is difficult to anticipate where a crack will start in service and to adequately simulate conditions at the crack tip if this is produced artificially. If this is done, however, any artificial damage should be as realistic as possible and should include complete fracture of at least one skin reinforcing member, as experience has shown that skin cracks are likely to begin in this way. Some crack propagation should be induced by cycling to give more representative conditions at the crack tip.

6 FACTORS TO ALLOW FOR VARIATION OF OPERATIONAL LOADING CONDITIONS

6.1 As discussed earlier, the estimated safe life or inspection period is usually based on an average load spectrum that it derived from generalised data, or from measurements made on a small number of rotorcraft. Since there is considerable scatter in the severity, or amplitude, of loading that occurs in any one role and since damage is, to first order, proportional to amplitude raised to a power of at least 3, it follows that using an average severity of loading admits a disproportionate risk of underestimating the actual damage to some rotorcraft. For example, if for any one rotorcraft, the actual severity of loading was 20% lower than average, the damage would be at least 50% lower than estimated. If the severity were 20% higher, however, the damage would be at least 75% higher than estimated. To avoid imbalances of this type, direct evidence of the loads encountered by each individual rotorcraft is obtained whenever practicable. In the absence of this evidence, some compensation for the greater damage caused by the higher severities of loading is made in one of two ways. When the predominant damage occurs on the steep part of the S-N curve, the loads in the average loading spectrum are increased by 1.5 in number in estimating the life. When the predominant damage is associated with the fairly flat part of the S-N curve, the loads in the average spectrum are increased by 1.2 in amplitude. It is customary to waive these factors for those components on which the service loads are monitored, directly or indirectly by an RAE Fatigue Load Meter. Nevertheless, they are included in the calculation of the life consumed in service by each individual rotorcraft during any period in which the loads are temporarily unmonitored.

6.2 In those cases where loads are not monitored and a 1.2 load factor would normally be required, it may be acceptable to make a large number of representative flight load measurements of the loads that occur in each fatigue-damaging condition and associate these maximum measured loads with the duration of each condition in service.

TABLE 1

FATIGUE TEST FACTORS FOR SAFE-LIFE COMPLIANCE

Test factors to allow for variability in the fatigue performance of nominally identical items tested under the same spectrum loading.

TABLE 1 (a) STRUCTURAL FEATURES AND COMPLETE ITEMS

Number of items tested	Factor on geometric mean endurance over the steep part of the S-N curve	Factor on arithmetic mean strength at high endurances	
		Aluminium Alloys	Medium strength steels
1	3.75	1.74	1.49
2	3.33	1.67	1.45
3	3.16	1.64	1.43
4	3.06	1.62	1.41
6 or more	3.00	1.60	1.40

TABLE 1 (b) TRANSMISSIONS AND GEARBOXES

Number of gear wheels tested	Factor on arithmetic mean strength
1 to 3	1.4
4 or more	1.3

TABLE 2

FATIGUE TEST FACTORS FOR INSPECTION-DEPENDENT COMPLIANCE

Test factors to allow for variability in the crack propagation period if nominally identical items are tested under the same spectrum loading.

Number of items tested	Factor on geometric mean propagation period
1	3.00
2	2.66
3	2.52
4	2.44
6 or more	2.35

CHAPTER 202

MANOEUVRES

1 INTRODUCTION

1.1 The requirements of this Chapter apply to all types of rotorcraft and to the rotorcraft as a whole in each case.

1.2 Refer to Leaflet 202/1 for definitions.

2 FACTORS

2.1 All components of the rotorcraft which are subjected to loads resulting from manoeuvres shall have proof and ultimate factors as specified in Chapter 200 para 3, unless otherwise stated in the Rotorcraft Specification. The factors shall be applied to the external loads and inertia loads which arise in each of the following cases unless their application to the resulting internal stresses is more conservative.

3 MANOEUVRES TO BE CONSIDERED

3.1 When the flight control system of the rotorcraft is of conventional design such that the movement of the control motivators is directly related to the movement of the pilot's inceptors the cases stated in paragraphs 4 to 7 of this Chapter shall apply.

3.2 When the flight control system of the rotorcraft incorporates load limited devices the appropriate combinations of manoeuvres to be considered shall be stated in the Rotorcraft Specification. The cases stated in paragraphs 4 to 7 of this Chapter shall be considered in so far as they are applicable. Reference shall be made to Chapter 207.

4 FLIGHT LOADS

4.1 The rotorcraft shall be designed to withstand the loads which result from appropriate combination of control motivator movement as required to meet the requirements stated in the Rotorcraft Specification. All conditions of operating altitude shall be considered.

5 LIMIT MANOEUVRING LOAD FACTOR

5.1 The positive and negative values of the limit manoeuvring load factor shall be stated in the Rotorcraft Specification and shall be applied at all forward speeds from zero to the design speed unless otherwise stated in the Rotorcraft Specification.

5.2 LOAD APPLICATION

5.2.1 The flight load factor shall be assumed to act perpendicularly to the longitudinal axis of the rotorcraft and be of equal magnitude and opposite in direction to the rotorcraft normal acceleration load factor at the centre of gravity.

5.2.2 The loads resulting from the application of limit manoeuvring load factors shall be assumed to act at the centre of each rotor hub and at each auxiliary lifting surface and to act in directions and with distributions of load among the rotors and auxiliary lifting surfaces, so as to represent each critical manoeuvring condition.

5.2.3 In addition to the loads resulting from the requirement of paragraph 5.1 the rotorcraft shall withstand the maximum loads which arise from the most severe movements of the controls including the effect of aft movement of the pitch inceptor, which it is anticipated will occur during operational flight including the emergency conditions consequent upon an engine failure. Allowance may be made for control limitations imposed to avoid rotor/airframe strikes. The most adverse combinations of flight speed, rotor rotational speed and control movements shall be considered.

6 YAWING FLIGHT

6.1 DESIGN CONDITIONS

6.1.1 The manoeuvre shall normally be considered with the rotorcraft in initially unaccelerated flight at zero yaw at:

(i) All obtainable airspeeds up to the rotor never exceed speed.

(ii) All altitudes up to the maximum operating altitude.

(iii) With maximum rotor rotational speed (power on).

6.2 DESIGN CASES

6.2.1 The input from the pilots directional control inceptor shall be such that in the absence of response of the rotorcraft it results in an effectively instantaneous application of the yaw motivator(s). Unless otherwise stated in the Rotorcraft Specification the deflection of the yaw motivator(s) shall be determined by the following considerations whichever yields the critical case for the particular rotorcraft:

(i) That corresponding to the maximum output permitted by the flight control system for the power unit of each yaw control motivator.

(ii) That corresponding to the maximum deflection of the yaw motivator(s).

(iii) That corresponding to maximum pilot effort.

6.2.2 The following specific phases of the manoeuvre shall be considered:

(i) The loads consequent upon the rotorcraft yawing to the resulting sideslip angle.

(ii) With the rotorcraft yawed to the resulting sideslip angle the loads which arise when the motivator deflection(s) is/are instantaneously returned to the initial condition of paragraph 6.1.

7 SIDEWAYS FLIGHT CONDITIONS

7.1 DESIGN CONDITIONS

7.1.1 The manoeuvre shall be considered with the rotorcraft in a trimmed condition in initially unaccelerated flight at:

(i) All obtainable airspeeds up to the rotor never exceed speed in level flight.

(ii) Both engine power-on and power-off conditions.

7.2 DESIGN CASES

7.2.1 The loads arising when a sideways velocity is superimposed of magnitude defined by the lower of:

(i) The maximum sideways velocity attainable by the rotorcraft in either direction.

(ii) A sideways velocity of 15.2 m/s in either direction.

8 AUXILIARY ROTORS AND CONTROL/STABILISER SURFACES

8.1 Each auxiliary rotor and each fixed or movable control or stabilising surface shall be designed to meet the loads arising from:

(i) All relevant conditions specified in paragraphs 4 to 7 of the Chapter.

(ii) The requirements specified in Chapter 203, Control Systems - Mechanical Components.

9 POWERPLANT AND TRANSMISSION SYSTEM

9.1 Consideration shall be given to the torque loads which arise in the powerplant and transmission as a direct consequence of rotorcraft manoeuvres such as spot turns and rapid recovery from sideways flight.

10 SUPPLEMENTARY CONDITIONS AND ASSUMPTIONS

10.1 Unless otherwise stated the following conditions shall be considered for each loading case.

10.2 ENGINE POWER

10.2.1 All engine conditions (including power-off) likely to occur in association with the particular case under investigation.

10.3 ROTOR ROTATIONAL SPEED

10.3.1 The range of main rotor rotational speed for the power-on and power-off conditions together with the corresponding maximum attainable speeds in forward flight.

10.4 LANDING GEAR

10.4.1 All positions of the landing gear likely to occur in association with the particular case under investigation shall be considered.

10.5 MASS AND MASS DISTRIBUTION

10.5.1 Consideration shall be given to all masses between:

(i) The minimum flying mass (that is the take off mass less bombs, ammunition and other items readily dropped or expended except for sufficient fuel for a normal descent and 15 minutes cruise at sea level at the engine conditions appropriate to maximum endurance).

(ii) The maximum mass at which the rotorcraft can reach the altitude considered.

10.5.2 The design centre of gravity positions at each mass and aerodynamic condition shall include an appropriate tolerance beyond the actual maximum forward and actual maximum aft positions which are attainable in that configuration.

LEAFLET 202/1

MANOEUVRES

DEFINITIONS

1 INTRODUCTION

1.1 The definitions given in this Appendix and in the British Standard Glossary of Aeronautical Terms (B.S. Specification No. 185) are recommended for general use. To avoid confusion, it is particularly important that alternative and undefined terms should be avoided in type records, strength calculations and test reports submitted for official approval.

2 ACCELERATION

2.1 NORMAL ACCELERATION COEFFICIENT (n)

2.1.1 For structural design purposes the normal acceleration coefficient "n" is the resultant of the total external force acting perpendicularly to the fore and aft datum of the rotorcraft, divided by the total weight.

2.2 MAXIMUM NORMAL ACCELERATION COEFFICIENT (n_1)

2.2.1 The maximum normal acceleration coefficient is the value of "n" which is accepted at the design stage as being the highest positive value for which a particular rotorcraft need be designed to cover flight manoeuvres in its plane of symmetry. The value is chosen on the basis of the intended use of the type, modified (if appropriate) by the characteristics of the individual design. It is denoted by the symbol n_1.

3 POWERED FLYING CONTROLS

3.1 A powered flying control system is one in which the whole or part of the power required to move the main flying control motivator(s) concerned is supplied by an electric, hydraulic, or other non-human source.

4 SPEED

4.1 DESIGN DIVING SPEED V_D

4.1.1 The design diving speed is the speed accepted at the design stage as being the highest Equivalent Air Speed for which the particular rotorcraft need be designed, the value being chosen on the basis of the intended use of the type, modified (if appropriate) by the characteristics of the individual design. It is denoted by the symbol V_D.

CHAPTER 203

CONTROL SYSTEMS - MECHANICAL COMPONENTS

1 INTRODUCTION

1.1 Unless otherwise specified the requirements of this chapter apply to the mechanical components of all control systems, including flight and engine controls, of all rotorcraft.

1.2 The requirements shall be met over a range of temperatures (as given in Chapter 101) from that at ground level to the minimum value at the service ceiling of the rotorcraft when operating at the lightest practicable weight, and for all design speed and acceleration conditions.

1.3 The requirements shall be considered in conjunction with other relevant chapters including:

Chapter 200 Static Strength and Deformation
Chapter 201 Fatigue Damage Tolerance
Chapter 207 Active Control Systems (JAC Paper No. 1234)
Chapter 500 Aero-elasticity

2 STRENGTH

2.1 GENERAL

2.1.1 The strength requirements of this paragraph apply to conventional shaft driven single rotor rotorcraft with manual control systems in which each control is mechanically independent of the others. The same requirements are also applicable in principle, to rotorcraft of different design with control systems having more than one degree of freedom. For such control systems however, pilot effort alone may not define stressing loads for the whole system, or the application of simple criteria based on pilot effort may lead to ambiguities. In these circumstances, the procedure to be adopted shall be stated in the relevant Rotorcraft Specification.

2.1.2 Each circuit shall have proof and ultimate factors not less than 1.125 and 1.5 respectively under the given load conditions, unless otherwise stated. (See Chapter 200, para 3.1).

2.1.3 The ultimate factor for cables shall be 2.0 and for chains shall be 3.0.

2.1.4 The loads shall be applied in each case at the point at which the pilot would normally apply them and shall be reacted as appropriate:

(i) by the control system with the control surface deflected to its stops, or

(ii) by the attachment of the control system to the control surface horn (with the control in the critical positions for the affected parts of the system within the limits of its motion), or

(iii) by an intermediate component specifically provided for the purpose of reacting the input loads; such as an irreversible mechanism locked in the most adverse position, or local system stops used in conjunction with a load limited device in the circuit. When this is the case a rational analysis shall be used to determine the strength requirements for the remainder of the circuit.

(iv) by the attachment of the control system to the rotor blade pitch control horn only (with the control in the critical positions for the affected parts of the system within the limited of its motion).

2.2 PRIMARY CIRCUIT DESIGN LOADS

2.2.1 Conventional Controls

(i) Cyclic Hand Inceptor fore and aft motion

(a)	Stick control column	600 N (two hands together on a single hand grip)

(ii) Cyclic Hand Inceptor lateral motion

(a)	Stick type control column	356 N

(iii) Collective Hand Inceptor vertical motion stick type control column — 600N

(iv) Foot inceptor (see also Chapter 310)

(a)	Rudder bar/pedals	756 N (on one side only)
(b)	Rudder bar/pedals	445 N (simultaneously on each side in the same direction) except where overridden by brake loads - see Leaflet 310/1, para 7

(v) Twist grip inceptor — 17 Nm torque

2.2.2 Side Stick Inceptor — 665 N (in any direction in a horizontal plane)

2.3 DUAL CONTROL

2.3.1 Dual control systems shall be designed to the following conditions:-

(i) Pilots acting together.
Each pilot applying simultaneously loads equal to 75% of those specified in para 2.2.1 for each inceptor. For dual control rotorcraft which are derived from a single control rotorcraft and are intended for operational conversion training, the loading may be reduced, to each pilot applying simultaneously loads equal to 50% of the loads specified in para 2.2.1.

(ii) Pilots acting in opposition.
Each pilot applying simultaneously the loads specified in para 2.2.1 for each inceptor.
Note: This case provides a design condition for the inter-connecting circuit between the two sets of controls.

2.4 GROUND GUST CONDITIONS

2.4.1 The primary control circuits must be designed to cater for control surface loads which arise from gusts when the rotorcraft is on the ground, with the rotors stationary.

(i) When gust locks are not fitted that part of the control circuit between the cockpit controls and the control stops nearest to the motivator shall be subjected to the loads arising from a limit load, P, defined in para 2.4.2 below but not exceeding those of paras 2.2.1 or 2.3.1 above, as appropriate.

(ii) When gust locks are fitted the gust locks and the stops nearest to the motivator and any parts of the control circuit between these locks and stops and the control motivator final operating lever shall be subjected to the loads arising from a limit load, P, para 2.4.2.

2.4.2 The limit load, P, shall be that appropriate to the case when the rotorcraft is subjected to a gust of 41m/s unless otherwise stated in Rotorcraft specification.

2.5 DUPLICATE SYSTEMS

2.5.1 When duplicate systems are employed, the requirements of para 2.2 will apply to each system separately.

2.6 SECONDARY CIRCUIT DESIGN LOADS

2.6.1

(i) Crank, wheel or lever.

$$\frac{220(1+40R)\ N}{3}$$

where R is the radius at which the load is applied in metres (inches) but not less than 220 N, nor more than 660 N.

(ii) Twist 15 Nm

(iii) Push-pull controls. There are no specific loads.

3 CONTROL CIRCUITS - OVERALL DESIGN

3.1 EFFICIENCY AND MOVEMENT

3.1.1 Every effort shall be made to keep the efficiency of the mechanical control circuits as high as possible. Movement shall be smooth and positive as appropriate to the function.

3.1.2 Within the limitations of the airframe design and the overall control system requirements the routing of the circuits shall be as simple and direct as possible. Except where designed to do so, eg by the inclusion of items such as series actuators the length of circuits shall not change due to:-

(i) Operation of the control

(ii) Structural deformation, flight acceleration or differential thermal expansion/contraction

3.1.3 In the event that flight accelerations result in control system motion due to out of balance forces, whether these be unintentional or the result of the deliberate incorporation of devices in the system, the uncommanded control input shall act as to reduce the acceleration causing the motion.

3.1.4 Control systems for essential services shall be so designed that when a movement to one position has been selected, a different position can be selected without waiting for the completion of the initially selected movement, and the system shall arrive at the finally selected position without further attention. The movements which follow and the time taken by the system to allow the required sequence of selection shall not be such as to adversely affect the airworthiness of the rotorcraft.

3.1.5 Where practical the sense of motion involved in the operation of the controls shall correspond with the sense of response either of the rotorcraft or, where more relevant, of the inceptor operated.

3.1.6 Where appropriate the design shall be such that each circuit will retain any given setting and will not tend to creep under control loading or vibration.

3.1.7 Each circuit shall possess adequate stiffness to react to the operating loads without excess deflection.

3.2 INCORRECT ASSEMBLY

3.2.1 Each element of each control system shall be designed so as to minimise the probability of incorrect assembly that could result in malfunctioning of the system.

3.2.2 For control systems which, if incorrectly assembled would hazard the rotorcraft, the design shall be such that at all reasonable break-down points it is mechanically impossible to assemble elements of the system to give out of phase action, reverse sense of the control or unintended interconnection between two systems.

3.3 FAILURE IMMUNITY, SAFETY

3.3.1 A failure mode and effects analysis shall be carried on the control system, supported where necessary by tests. All combinations of potential failures and jamming shall be considered. The associated probabilities of occurrence shall be assessed in relation to the continued safe flight, and landing of the rotorcraft to levels stated in the relevant specification for the Rotorcraft.

3.3.2 The design and location of the controls shall be such as to minimise the risk of inadvertent operation either by personnel entering or leaving the rotorcraft or by the flight crew during normal movement in the cockpit.

3.3.3 Protection shall be provided to prevent any control circuit components from being used as a step or handhold and to prevent interference, jamming or chafing from cargo, passengers or loose objects. There shall be means, especially in the cockpit area, to prevent foreign objects entering locations where they would jam the system.

3.4 DURABILITY, WEAR AND BACKLASH

3.4.1 The fatigue life of the control systems shall be agreed with the Rotorcraft Project Director.

3.4.2 The design of the control system shall be such as to minimise backlash, see also Chapter 500 paras 3.1 (ii) and 3.7.

3.5 BREAKOUT FORCE

3.5.1 In the case of primary flight control circuits the force required on the inceptor to overcome the static friction in the relevant circuit, shall ensure that static friction has no material adverse effect on the stability and control characteristics of the rotorcraft. Where necessary controls shall have an adequate degree of self-centring on release; any breakout forces shall be as small as possible.

3.6 ENVIRONMENTAL CONDITIONS, TEMPERATURE, MOISTURE, AND ICE

3.6.1 The control circuits shall be designed to operate over and whole temperature range specified in para 1.2 and without any deleterious effects due to tightening or slackening resulting from differential expansion.

3.6.2 Sufficient clearance shall be provided to ensure the efficient operation of all detail fittings, such as jacks, bearings, guides, fairleads, etc., over the temperature range of para 1.2. It shall be demonstrated in hot and cold temperature tests that the clearances are sufficient.

3.6.3 It shall be demonstrated that the clearances provided are sufficient to ensure efficient operation of the controls under all conditions of humidity. Wherever possible materials which expand appreciably with moisture shall be avoided for such parts as fairleads and washers. Component design shall avoid pockets, traps, wells, etc., into which water, condensed moisture or other liquids would collect; or adequate drain provision shall be made.

3.6.4 Clearances shall be sufficient to avoid the possibility of jamming due to the accretion of ice.

3.7 COMBAT VULNERABILITY, PRIMARY, TRIM, SERVO AND ENGINE CONTROLS

3.7.1 On all rotorcraft except trainers, consideration shall be given to the reduction of the vulnerability of all flight and engine controls vital to the safety of the rotorcraft. The aim shall be that the failure of any single part of the flying control system shall not leave the pilot without adequate control for a return flight and safe landing.

3.7.2 Control circuits shall be run in those areas of the airframe where combat damage is less probable with respect to threat and the role of the rotorcraft.

3.7.3 Duplicated parts of control circuits shall be located as far apart as possible.

4 CONTROL CIRCUIT - DETAILS

4.1 COMPONENTS (see also Leaflet 203/5)

4.1.1 The components used in control circuits shall be compatible with one another and their design and installation shall be such as to ensure compliance with the requirements of para 2 and 3.

4.2 STOPS

4.2.1 The range of control motivator movement, including blade pitch angle and of pilot's inceptor movement and the associated trimming devices, shall be limited by resilient stops which are robust and positive.

4.2.2 Each stop shall be located so that wear, slackness, or take-up adjustments will not adversely affect the control characteristics of the rotorcraft as a result of a change in the range of surface travel.

4.3 GUST LOCKS

4.3.1 Where a device is used for locking a control motivator whilst the rotorcraft is stationary, the locking device shall:

(i) Automatically disengage when the primary flight controls are operated in the normal manner, or

(ii) Limit that operation of the rotorcraft so that the pilot receives unmistakable warning at the start of take-off.

4.3.2 Means shall be provided to preclude the possibility of the locks becoming engaged in flight.

4.4 PEDAL INTERCONNECT

4.4.1 Pedals shall be interconnected to ensure positive movement of each pedal in both directions.

4.5 UNCONVENTIONAL CONTROLS

4.5.1 When other than conventional pilot controls are used, prior agreement shall be obtained from the Rotorcraft Project Director.

5 SERVO AIDS

5.1 If any means are employed to obtain lightness of the main flying controls (eg, by spring tabs or servo) the design shall be such that it will not be necessary for the pilot to exercise any unusual degree of restraint in order to ensure freedom from structural instabilities. The use of reliable safety devices limiting the control operation under appropriate conditions is an acceptable means of meeting this requirement.

6 POWER CONTROL UNITS AND SYSTEMS (see also Chapter 207 and Leaflet 203/1)

6.1 Power operated control systems shall comply with the requirements of this chapter in so far as they are applicable.

6.1.2 Where appropriate the strength requirements shall be met with the system both operative and inoperative.

6.1.3 The proof and ultimate factors shall be 1.125 and 1.5 respectively under the maximum load which can be developed in the system under all practicable operating conditions.

6.2 TESTS ON POWERED FLYING CONTROL UNITS (see also Leaflets 203/2, 203/3 and 203/4)

6.2.1 The tests required by para 6.2.2 and 6.2.3 shall be carried out in a ground test rig.

6.2.2 Flight Clearance Test - Any new type of powered flying control unit shall be submitted to a flight clearance test before the first flight test of the unit.

6.2.3 Unit Design Clearance Test - Any prototype of any new design of powered flight control unit shall be submitted to a ground test for design clearance.

6.3 TESTS ON POWERED FLYING CONTROL INSTALLATIONS (see also Leaflets 203/2, 203/3 and 203/4)

6.3.1 The tests required by paras 6.3.2 and 6.3.3 below may be made partly in a ground rig but shall include adequate ground testing in the rotorcraft.

6.3.2 Installation Flight Clearance Test - Any new installation of powered control unit shall be submitted to a ground test for flight clearance before the first flight of the installation.

6.3.3 Installation Design Clearance - A prototype of any new installation of a powered flying control shall be submitted to a ground test for design clearance, see also Leaflets 203/2 and 203/3.

6.4 FATIGUE TESTS (see also Chapter 201 para 3.1)

6.4.1 Each powered flying control system shall have a safe life determined in accordance with Chapter 201 at least equal to the specified life of the rotorcraft, unless otherwise agreed. This shall be demonstrated by a fatigue test made in accordance with Leaflet 203/4 using the standard duty cycle in Table 1 with the additional requirement that the rotorcraft designer shall specify in additional oscillatory loading arising from the rotorheads for a given rotorcraft.

7 AUTOMATIC FLIGHT CONTROL SYSTEMS (AFCS) OR AUTOMATIC STABILISATION EQUIPMENT (ASE)

7.1 When an automatic input is incorporated into a flight control circuit, the mounting of the actuator or servo motor to the structure of the rotorcraft, and their attachments to the control system itself, shall have a proof and ultimate factor not less than 1.125 and 1.5 respectively under the more severe of the following two conditions:

(i) the greatest force which the actuator or servo motor can apply, and

(ii) the force exerted by the pilot from the cockpit (see para 2.2) carried through to the actuator or servo motor stops.

8 MAINTENANCE (see also Part 8)

8.1 ACCESSIBILITY

8.1.1 Components shall be designed, installed, located and provided with access so that inspection, rigging, repair, replacement and lubrication can be easily accomplished, including visual inspection of fairleads, pulleys, terminals and turnbuckles.

8.2 SAFETY

8.2.1 Systems and components shall be designed and installed to preclude injury of personnel during maintenance and testing. Where positive protection cannot be provided adequate warning and instructions shall be given.

8.3 RIGGING INFORMATION

8.3.1 The limiting breakout force established for the design in accordance with para 3.5.1, as appropriate to the type, shall be stated on the drawings for inspection purposes together with the conditions of its measurements.

8.3.2 In the case of cable-operated systems, the initial tensions at which, at normal hangar temperatures, each primary control circuit should be rigged shall be stated on the appropriate drawings for inspection purposes. The requirements of para 3.5.1 for static friction shall be met at the correct rigging tension.

8.3.3. The information on correct rigging dimensions, rigging tensions and allowable static friction limits shall be included in the relevant Publications.

LEAFLET 203/0

CONTROL SYSTEMS - MECHANICAL COMPONENTS

REFERENCE PAGE

RAE Reports

Aero 2232	..	..	Papers and proceedings of a meeting to discuss powered lying controls
Mech.Eng. 12	..	..	Transitional friction efforts in powered controls with particular reference to hydraulic jacks

RAE Technical Reports

66304	..	..	An investigation into the problem of valve stiction associated with the use of high temperature hydraulic fluids

RAE Technical Notes

Aero 1173	..	..	Note on friction in control circuits
SME86	..	..	Alternative bearings for control circuits
SME247	..	..	High thermal expansion steel in aircraft control circuits
SME367	..	..	The vulnerability of flying control systems in military craft
Mech.Eng. 2	..	..	The presence of air in oil and its probable effect on the irreversibility of hydraulic transmission systems
Mech.Eng. 10	..	..	Tests on servodyne hydraulic control assister
Mech.Eng. 77	..	..	An electro-hydraulic-powered flying control system with automatic safeguards and variable control characteristics
Mech.Eng. 98	..	..	A contribution to the theory of servo mechanisms: dynamic rigidity of valve-controlled hydraulic servos
Mech.Eng. 100	..	..	Free oscillations of valve-controlled hydraulic servos
Mech.Eng. 108	..	..	Impedance of idealised hydraulic valve-controlled servos
Mech.Eng. 110	..	..	Portable hinge moment simulator for powered control circuits
Mech.Eng. 129	..	..	Comparative rigidities of certain valve-controlled hydraulic servos
Mech.Eng. 132	..	..	Valve proportionality effects in aircraft hydraulic flying control systems
Mech.Eng. 144	..	..	Some noted on irreversible machines
Structures 58	..	..	Aircraft with automatic pilots and powered controls; safety aspects

A & AEE Reports

Tech. 105	..	..	Control system requirements for military aircraft in relation to the task of the human operator
Tech. 194 Iss.2	..	..	A further resume of pilots' experience with various forms of feel and power control systems

RAE Specifications

AD115	..	..	Control column force indication

Scientific and Technical Memoranda

13/51	..	..	Some studies in hydraulic servo flying controls
15/51	..	..	An investigation of methods of duplicating hydraulically-operated irreversible flying controls
8/53	..	..	On the stability of powered flying controls
2/54	..	..	Some problems of powered flying controls

RAeS Journal

March, 1949	..	..	Some thoughts on the use of powered flying controls in aircraft

LEAFLET 203/1

CONTROL SYSTEMS - MECHANICAL COMPONENTS

POWERED FLYING CONTROLS

1 INTRODUCTION

1.1 Basic safety requirements for powered flying controls are specified in Chapter 604. This leaflet deals with the strength of the powered flying control circuit, mounting the control unit into the rotorcraft and the control characteristics when the controls are operated in conjunction with an automatic stabilisation device.

2 CONTROL CIRCUIT STRENGTH

2.1 The design load for a mechanical input circuit of a powered flying control system is that applied by the pilot in the presence of a feed-back or artificial feel mechanism: the powered control unit valve(s) input may, therefore, be loaded by the pilot's maximum effort, less whatever is absorbed by the feel mechanism. Hence there is no significant difference between a mechanical input circuit of a powered control and a manual control circuit. This part of the circuit should meet the strength requirements of Chapter 203 which call for an ultimate factor of at least 1.5 on the maximum effort which a pilot is capable of applying, with a jam of the powered control unit input. The pilot's normal effort, however, is only a small fraction of his attainable maximum and so this part of the circuit will usually meet the fatigue strength requirements of Chapter 201 (see para 2.4 below), when designed to the strength requirements of Chapter 203. Reaction of the maximum effort in both directions should be assumed at all possible parts of the circuit including an extreme position at the power unit. As this loading implies a failure of the powered control unit or its input valve to respond to normal effort, its application must be considered in both directions.

2.2 Account should be taken of loads that may be induced in the parked condition when components may deflect downwards under weight or upwards and downwards due to wind loads. Circuit loads will depend on the location of input circuit stops, possibly artificial feel forces, and damping devices if fitted.

2.3 The design load for the output circuit should be the maximum load which the power unit is capable of applying increased by the pilot's effort through the feed-back or artificial feel mechanism where this is possible. The contribution of the powered control unit is, however, much larger than that due to the pilot and in general the working load in this circuit is much closer to the maximum attainable load than is the case in the input circuit. Consideration should also be given to conditions arising in the circuit during and after change-over to any emergency system of control. High transient peak loads may occur when the system is brought to rest rapidly, eg by sudden closure of a valve after movement at maximum velocity, owing to the kinetic energy to be dissipated in a small displacement. It is important therefore in hydraulic systems to stress for the loads then arising as these may be greater than those due to the maximum working pressure.

Care should also be taken to avoid stress concentrations; testing to an approval schedule will be necessary. The maximum load may be greater than that needed for rotorcraft control: for instance, each part of a duplicated power system may be capable of three quarters of all of the necessary force.

2.4 The fatigue strength of mechanical powered flying control circuits should be established in accordance with the principle outlined in Chapter 201. (Account should be taken of the loading actions referred to above).

3 POWERED FLYING CONTROLS IN CONJUNCTION WITH AN AUTOMATIC FLIGHT CONTROL OR WITH AN AUTOMATIC STABILISATION DEVICE

3.1 When the powered flying controls are to be operated by an automatic flight control or in conjunction with an auto-stabiliser the control characteristics under powered flying controls should be satisfactory both with and without the automatic flight control and/or the auto-stabiliser in operation (see Chapter 604).

LEAFLET 203/2

CONTROL SYSTEMS - MECHANICAL COMPONENTS

TESTING OF POWERED FLYING CONTROLS

1 INTRODUCTION

1.1 In this Leaflet, information and explanation are given regarding tests of powered flying controls called for in Chapter 203, para 6.

1.2 The tests are divided into two stages:

(i) the powered control unit by itself - part of the Type Approval of the Unit, and

(ii) the powered control installation for a particular rotorcraft covering the complete system from pilot's input to the control surface.

2 GROUND TEST RIGS

2.1 The test rig for the powered control unit should be capable of applying loads to the output member of the unit varying at the required relationship to the displacement and should operate the input member through the required duty cycle at any required load, speed, frequency, and for any desired endurance period. Means should be provided for measuring all characteristics of the control such as loadings, friction, rates of operation, elasticity, mass, inertia, resonance frequencies, stability, and temperatures and pressures, where applicable, at critical points in the control. The need to apply assisting loading and not merely resistive, when this may occur naturally in flight, should not be overlooked. A chamber should be available for checking the operation of the unit at high and low temperatures as required by Chapter 101.

2.2 In the case of the complete control system the test rig should represent, as closely as possible, the conditions of the rotorcraft in which the powered control is to be flight tested. All loads should be applied to the actual control motivator or its test rig equivalent and operation of the input member should be through the pilot's input. A local rig structure to represent rotorcraft stiffness for stability and resonance tests is normally essential. For check testing of the complete system when installed in the rotorcraft, some form of control motivator loading rig should be available.

3 TESTS OF THE POWERED CONTROL UNIT

3.1 PROTOTYPE CONTROL UNIT - FLIGHT CLEARANCE TEST

3.1.1 During the development stages of a new design of control unit some testing of the prototype unit will be necessary before the unit reaches the test stage of para 3.2. In order to ensure a reasonable standard of safety in flight, before any flight testing is done the prototype control unit should be submitted to a Flight Clearance Test in a ground rig as required by Chapter 203. This test may be either separate from, or included in, the Flight Clearance Test of the installation in which the unit is to be flight tested, described in para 4.1 below. An endurance test, followed by an inspection and performance check, should be carried out before flight.

3.2 THE CONTROL UNIT - UNIT DESIGN CLEARANCE TEST

3.2.1 The development of a new design of control unit to the stage when it is ready for production may involve much rig and flight testing. Before production is started, however, approval of the type should be obtained and for this purpose a prototype/typical production unit should be subjected to the Unit Design Clearance Test called for in Chapter 203. This should be an extended test covering all aspects of performance, endurance, stability and functioning of safety devices under all reasonably foreseeable types of simulated failures. During the test the unit should demonstrate that it gives its declared performance, that it functions within its declared temperature conditions and also when subjected to flight acceleration conditions.

3.2.2 For endurance testing, the input member should be operated through a 'duty cycle' repeating it continuously for an adequate total running time. The composition of this duty cycle should be decided on before the start of the test to suit the particular control and the type of rotorcraft for which the control is intended, in accordance with Leaflet 203/3.

3.2.3 The aim of the complete test should be to establish that the performance of the unit is maintained over the specified flying hours between scheduled removals. Should unit design clearance testing be included in Installation Design Clearance testing only sufficient testing to give confidence for development flying need be done.

3.2.4 No modifications or major adjustments should be made during the test and on completion there should be a complete strip and examination of the unit. The unit will not be deemed to have passed the test unless the strip and examination is satisfactory, the standard to be that its condition is such that the unit would still be satisfactory for safe use in the rotorcraft.

3.2.5 If it is necessary to modify an approved control unit to suit a particular rotorcraft type, a full test of the modified version is unlikely to be necessary but a test to cover the revised design requirements and modifications to the unit should be made.

4 TESTS OF THE POWERED CONTROL INSTALLATION

4.1 THE POWERED CONTROL INSTALLATION - FLIGHT CLEARANCE

4.1.1 Before any powered control unit is flight tested either in an established type of rotorcraft or in a new prototype, the complete powered control system, including control runs and/or circuitry and control motivator actuators is required by Chapter 203 to undergo a Flight Clearance Test, partly in a ground test rig and partly when installed in the rotorcraft in which flight tests are to be made.

4.1.2 The rig test should be carried out on a prototype of the complete system to a test schedule which should be representative of the test flying intended. The test schedule should call for an endurance period but this need not be extensive if the unit is already cleared for flight since the rotorcraft controls themselves do not usually require long endurance testing. If the control unit is not already cleared for flight by either of the tests at paras 3.1 or 3.2 adequate endurance testing should be included in the schedule. All other characteristics of the system such as performance, stability, resonance and functioning of safety devices under all required conditions should be thoroughly investigated. (For resonance testing, see Chapter 500)

4.1.3 Tests should be made to demonstrate that the safety requirements of Chapter 604, section 3 are met and that the operation or malfunctioning of other powered systems in the rotorcraft has no adverse effects on the powered control system under both normal and emergency conditions.

4.1.4 The rotorcraft ground test may be made on the actual installation which is to be flight tested. This installation should be identical with that tested in the rig. If both rig and rotorcraft ground tests are satisfactory and meet the standards declared in the pre-test declarations, flight testing may proceed.

4.1.5 It is possible that the rig test may reveal that flying time should be restricted until further ground tests have proved the reliability of the system.

4.2 INSTALLATION - DESIGN CLEARANCE

4.2.1 Further development of the system involving modifications may prove necessary as a result of ground and flight tests, and when the complete control system has reached the stage when it is considered satisfactory for production it should be submitted to its Final Installation Design Clearance Test as required by Chapter 203. This test should be carried out on the final version of the prototype installation. It should be similar to the Flight Clearance Test except that the endurance test should be sufficient to establish the satisfactory continued functioning of all working parts not already tested and approved. It should be followed by a strip and examination of all working parts, unless agreed by the Rotorcraft Project Director that an extended endurance test be carried out to establish a satisfactory Functioning life without strip examination. This should be considered if it is required to leave the test rig in a life consumed condition for further reliability or life extension testing.

4.2.2 In cases where the control unit employed in the installation has not already been tested and approved separately, the Installation Design Clearance Test may be extended to cover the Control Unit Design Clearance Test of para 3.2.

LEAFLET 203/3

CONTROL SYSTEMS - MECHANICAL COMPONENTS

TEST SCHEDULES FOR POWERED FLYING CONTROLS

1 INTRODUCTION

1.1 In this Leaflet, specimen test schedules for the testing of powered flying controls, as called for in Chapter 203 and as described in Leaflet 203/2, are given in detail. These schedules are intended to show the general form that the testing procedure should take and should be modified where necessary to suit any particular design of control.

2 PROTOTYPE CONTROL UNIT - FLIGHT CLEARANCE TEST (see Leaflet 203/2, para 3.1)

2.1 PRE-TEST DECLARATIONS

2.1.1 The following, each at the appropriate power supply conditions, need to be decided before commencement of the test:

(i) the composition of the "duty cycle" appropriate to each portion of the test,

(ii) the maximum and minimum rates of operation with the corresponding maximum loads at the control unit output member, and the maximum rate at zero load,

(iii) the maximum loads at the control unit output with the corresponding maximum rate, and

(iv) temperature and pressure limitations where applicable.

2.2 ENDURANCE TEST

2.2.1 With the power supply, if engine driven, at conditions corresponding to continuous cruising, operate the input member of the unit, repeating the appropriate duty cycle for a total running time of 25 hours.

2.2.2 Every 5 hours approximately, carry out full cycles of the input member (full cycle comprises movement from the neutral to one extreme position, return through neutral to opposite extreme and return to neutral) as follows:

(i) 10 full cycles at the declared maximum rate and corresponding maximum load,

(ii) 10 full cycles at intermediate load and rate,

(ii) 10 full cycles at declared maximum load and corresponding maximum rate and

(iv) operate safety device and allow 10 full cycles in the emergency condition. Where a means of change-over from emergency control back to normal is provided, check that this operates satisfactorily.

Note: In special cases where the change-over from the duty cycle to the full cycle involves undue time and labour, the total of 200 full cycles may all be carried out at the end of the endurance test.

2.2.3 The temperature rise at critical points in the control unit, (eg electric motor cooling air, fluid in hydraulic pumps, motors and jacks, etc), should be recorded during the test.

2.2.4 At some suitable time during the test, check under low temperature conditions sufficient to clear the unit for the experimental flying intended and under rotorcraft acceleration conditions sufficient to clear the unit for operation.

2.3 STRIP EXAMINATION

2.3.1 Strip and examine working parts.

Note: Performance testing is included in the Flight Clearance Test of the installation at para 4.

3 CONTROL UNIT DESIGN CLEARANCE TEST (Leaflet 203/2 para 3.2)

3.1 PRE-TEST DECLARATIONS

3.1.1 The following, each at the appropriate power supply conditions, need to be decided before commencement of the test:

(i) the composition of the "duty cycle" appropriate to each portion of the test,

(ii) the maximum and minimum rates of operation with the corresponding maximum loads at the control unit output member, and the maximum rate at zero load,

(iii) the maximum loads at the control unit output with the corresponding maximum rate, and

(iv) temperature and pressure limitations where applicable.

3.2 PRELIMINARY FUNCTIONING TEST

3.2.1 This test, comprising about 20 cycles varying at loads and rates, should precede all performance and endurance testing. Checks on the operation of the safety device should be made by simulating a failure in various positions of the control output member allowing a number of cycles in the emergency condition in each case. This should be repeated for all agreed possible types of failure including runaway, freeing or seizure of the control. Where a means of change-over from emergency control back to normal is provided, this should be checked in each case. The time to change over to the emergency condition, and back again, when appropriate, should be measured in each case.

3.3 PRELIMINARY PERFORMANCE CHECK

3.3.1 This test should precede the endurance test in order to establish the performance of the unit in the new condition. It should include the following:

(i) sensitivity tests, including backlash and overshoot measurements,

(ii) measurement of input friction,

(iii) measurement of maximum load developed by the output member,

(iv) measurement of maximum rate of operation at various loads from zero to maximum, applied to the output member, and

(v) frequency response tests to establish stability criteria.

3.3.2 The above should be repeated over a range of power supply conditions from the minimum emergency conditions to the maximum.

3.4 ENDURANCE TEST

3.4.1 With the power supply, if engine driven, at conditions corresponding to continuous cruising, operate the input member of the unit, repeating the appropriate endurance duty cycle, for a total running time of approximately 200 hours (see Leaflet 203/2 para 3.2.4).

3.4.2 Every 10 hours approximately, carry out full cycles of the input member (full cycle comprises movement from neutral to one extreme position, return through neutral to the opposite extreme and return to neutral) as follows:

(i) 10 full cycles at declared maximum rate and corresponding maximum load,

(ii) 10 full cycles at intermediate load and rate,

(iii) 10 full cycles at declared maximum load and corresponding maximum rate, and

(iv) operate safety device as at para 3.2 allowing 10 cycles in the emergency condition.

Note: In special cases where the change-over from the duty cycle to the full cycle involves undue time and labour, the 10 hour interval between the sets of full cycles may be increased to not more than 50 hours with the number of cycles at each interval adjusted to make the same total of 800.

3.4.3 The running times should be arranged to include at least one continuous endurance period equivalent to the endurance of the particular rotorcraft for which the unit is designed, or in the case of a unit not associated with a particular rotorcraft, a period of 12 hours.

3.4.4 The temperature rise at the critical points in the control unit, (eg electric motor cooling air, fluid in hydraulic jacks, etc), should be recorded during the test.

3.4.5 At some suitable time during the test, check under the temperature conditions of Chapter 101, as appropriate to the particular case under consideration. Record time taken to reach satisfactory functioning condition from starting. Check safety device.

3.5 FINAL PERFORMANCE CHECK

3.5.1 Repeat preliminary performance check at para 3.3 and note changes.

3.6 STRIP EXAMINATION

3.6.1 Strip and examine working parts.

4 INSTALLATION FLIGHT CLEARANCE TEST (see Leaflet 203/2, para 4.1)

4.1 PRE-TEST DECLARATION

4.1.1 The following, each at the appropriate power supply conditions, need to be decided before commencement of the tests:

(i) the composition of the duty cycle appropriate to each portion of the test,

(ii) the maximum and minimum rates of operation with the corresponding maximum loads at the control motivator, and the maximum rate at zero load,

(iii) the maximum loads at the control motivator and the corresponding maximum rate, and

(iv) temperature and pressure limitations, where applicable.

4.2 PRELIMINARY FUNCTIONING TEST

4.2.1 The complete powered control system should be installed in the ground test rig under conditions similar to the installation in the test rotorcraft. About 20 duty cycles at varying loads and rates should be made. Checks on the operation of the safety device should be made by simulating a failure in various positions of the control motivator allowing a number of cycles in the emergency condition of each case. This should be repeated for all agreed possible types of failure including runaway, freezing or seizure of the control. Where a means of change-over from emergency control back to normal is provided, this should be checked in each case. The time to change over to the emergency condition and back again, when appropriate, should be measured in each case.

4.3 PRELIMINARY PERFORMANCE CHECK

4.3.1 This test should be made with the complete control system in the test rig and should precede the endurance test in order to establish the performance of the system in the new condition. It should include:

(i) sensitivity tests, including backlash and overshoot measurements,

(ii) measurement of input friction,

(iii) measurement of maximum torque developed at control motivator,

(iv) measurement of maximum rate of operation at various loads from zero to maximum, applied to the control motivator, and

(v) resonance tests applied to the test rig.

4.3.2 The above should be repeated over a range of power supply conditions from the minimum emergency conditions to the maximum, including that appropriate to the all engine failure case. The tests using the emergency power supply appropriate to the all engine failure case should check that the performance is adequate and that the system will function for a sufficient length of time to meet the requirements of Chapter 100, para 9.2 appropriate to the particular rotorcraft concerned.

4.3.3 Tests should be made to demonstrate that the safety requirements are met and that operation of all other powered systems supplied from the same power source, does not materially affect the performance of the powered control.

4.4 ENDURANCE TEST

4.4.1 With the system in the test rig and with the power supply, if engine driven, at cruising conditions, operate the pilot's control through the declared endurance duty cycle for a total running time of:

(i) 5 hours, if the control unit is already cleared for flight, or

(ii) 25 hours with the full cycles of para 2.2 if the control unit is not already cleared for flight.

4.4.2 Carry out 600 full cycles of the pilot's control (full cycle defined at para 3.4.2) as follows:

(i) 200 full cycles at declared maximum rate and corresponding maximum load,

(ii) 200 full cycles at intermediate loads and rates, and

(iii) 200 full cycles at declared maximum load and corresponding maximum rate.

4.4.3 After every 50 cycles and at same load and speed, operate safety device as in para 4.2.1 and allow 5 cycles in the emergency conditions in each case.

4.4.4 Temperature and pressure measurement should be taken throughout as applicable.

4.5 FINAL PERFORMANCE CHECK IN TEST RIG

4.5.1 Repeat preliminary performance check at para 4.3 and note changes.

4.6 STRIP EXAMINATION

4.6.1 Strip and examine all parts not already tested as part of the control unit.

4.7 ROTORCRAFT GROUND TEST

4.7.1 Install a complete powered control system identical to that tested as above, in the test rotorcraft, and carry out the performance tests of para 4.3 as far as practicable on the ground using a control motivator loading device.

4.7.2 Carry out the resonance tests.

5 INSTALLATION DESIGN CLEARANCE TEST (see Leaflet 203/2, para 4.2)

5.1 PRE-TEST DECLARATION

5.1.1 As in Flight Clearance Test at para 4.1.

5.2 PRELIMINARY FUNCTIONING TEST

5.2.1 As in Flight Clearance Test at para 4.2.

5.3 PRELIMINARY FUNCTIONING TEST

5.3.1 As in Flight Clearance Test at para 4.3.

5.4 ENDURANCE TEST

5.4.1 With the system in the test rig and with power supply, if engine driven, at cruising conditions, operate the pilot's control through the declared endurance duty cycle for a total running time of:

(i) 50 hours if the control unit employed is a type tested and approved unit, or

(ii) 200 hours if the control unit employed is not a type tested and approved unit.

5.4.2 Every 10 hours carry out full cycles (full cycle defined at para 3.4.2) making a total of 3000 cycles, as follows:

(i) in a 50 hour test:

(a) 200 full cycles at declared maximum rate and corresponding maximum load,

(b) 200 full cycles at declared maximum loads and rates, and

(c) 200 full cycles at declared maximum load and corresponding maximum rate, and

(ii) in a 200 hour test, the number of full cycles at (a), (b) and (c) above should be reduced to 50 so that the same total of 3000 cycles is achieved in the complete test. Alternatively, the time interval between the sets of full cycles may be increased from 10 to a maximum of 50 hours with the number of full cycles in each set adjusted accordingly.

5.4.3 After every 50 full cycles and at same load and rate, operate safety device as at para 4.2 and allow 5 cycles in the emergency condition in each case. When a means of change-over from emergency control back to normal is provided, this should be operated in each case.

5.4.4 Temperature and pressure measurements should be taken throughout as applicable.

5.5 FINAL PERFORMANCE CHECK

5.5.1 Repeat preliminary performance check and note changes.

5.6 STRIP EXAMINATION

5.6.1 Strip and examine all working parts.

5.7 ROTORCRAFT GROUND TEST

5.7.1 Install the same prototype powered control system, or an identical one, in either a prototype or a production version of the rotorcraft for which the system is intended, and carry out the performance tests of para 4.3 up to about 30% maximum load, using a control motivator loading device, and check results with the rig test results.

5.7.2 Carry out resonance tests.

LEAFLET 203/4

CONTROL SYSTEM - MECHANICAL COMPONENTS

FATIGUE TESTING OF HYDRAULIC POWERED FLYING CONTROLS

1 INTRODUCTION

1.1 This leaflet contains details of the fatigue testing of hydraulic powered flying controls required by Chapter 203. The tests aim to ensure that hydraulic powered flying controls have adequate fatigue life under conditions of varying pressure stress to which they will be subjected in service. Included in the tests are screw jacks driven by hydraulic motors and, where applicable, automatic flight control system actuators and artificial feel simulators.

2 STANDARD DUTY CYCLE FOR CONTROL MOTIVATORS

2.1 The duty cycle for the control motivators should be specified by the airframe designer who should state clearly whether any additional factor has been taken into account. In the absence of such a specification the standard duty cycle in Table 1 should be used. Tests should then be applied in accordance with paras 3 and 4 below.

2.2 Due to the wide variation between the performance characteristics of various types of hydraulic powered flying control, standardisation of test conditions is based on flight duty cycles as applied to the control surfaces rather than the fluid pressures in the hydraulic powered control circuits. In the case of small amplitude high frequency control motivator movements, the internal jack pressures in a throttled valve servo will be entirely different from the pressures in a variable delivery servo pump. Similarly, for identical output conditions, the internal pressure will depend upon the precise nature of valve overlaps or underlaps, and in duplicated jacks, upon the inter-jack mechanical stiffness.

2.3 In addition to the standard duty cycle, which is based on motivator loads, fatigue loading on a powered control component may arise from other sources, and in many cases these may constitute the heaviest fatigue loading. These sources include ground testing cases, operation of other surfaces both on the ground and in flight, pump ripple and parking locks. The conditions controlling the above factors must be given full consideration in addition to the effects of the standard duty cycles.

3 CONVERSION OF DUTY CYCLE INTO TEST PRESSURE

3.1 Although the external loads applied to hydraulic powered flying control jacks will be broadly similar to those applied to the local rotorcraft structure, the internal jack pressures will be determined by other considerations. In each case, therefore, it will be necessary to determine experimentally the internal pressure fluctuations at critical points throughout the powered flying control system resulting from the duty cycles. The test should be carried out on a test rig representing as closely as possible the rotorcraft installation and special instrumentation may be necessary at this stage to ensure that local peak pressures are recorded.

3.2 Factors influencing the pressure variations are given below, but it is emphasised that this list does not necessarily include all the parameters significant to the performance of a powered flying control:

(i) inertia of the system,

(ii) stiffness of the system,

(iii) aerodynamic damping,

(iv) control valve damping characteristics,

(v) phasing of control valve ports (the smallest variation in port position can cause large pressure differences particularly in duplicated jacks, thus the magnitude of pressures applied in fatigue testing should be adjusted to include the most adverse conditions),

(vi) peak pressures imposed on the system by resonant tendencies at a control motivator, and

(vii) actual control velocities and frequencies experienced in flight.

4 METHOD OF TEST

4.1 The tests should be carried out under any one or a combination of the following conditions, or by any other approved method:

(i) By simple pressure pulse applications to units with output members locked or unlocked according to whether or not the load is transferred through these members during the particular duty cycle case. The jacks should be stationary, externally restrained and the output member locked in the position given in Table 1, and the relevant fluid pressures applied to the various pressure chambers. This method offers the advantages of allowing greater acceleration of the test cycle and reduction of test time (this can be lengthy when, for instance, clearance for 1000 hours is being sought and a scatter factor of 5 is to be applied).

(ii) By stroking the units in the normal manner (ie by an extension of the normal endurance tests called for in Leaflet 203/2) with the jacks moving against externally applied loads. In this condition the normal supply pressure should be provided at the powered control inlet and the internal jack pressure will be developed automatically. When using this method, any reduction in the test time by acceleration of the flight duty cycle is liable to modify the internal jack pressures and due allowance should be made for this.

(iii) With the jack and fittings uncoupled, in particular cases where the simulated internal pressure conditions could impose loads on the external attachments of the jacks in excess of those experienced in flight. In this condition, pressures on opposite sides of the jack piston should be suitably balanced.

5 TEST FACTORS

5.1 Fatigue test factors should be derived in accordance with Chapter 201.

TABLE 1

STANDARD FLIGHT DUTY CYCLE

Loading condition	Load (% of maximum)	Displacement (% of maximum)	Number of cycles per hour
Extreme Manoeuvre	100	100 approx. Just clear of jack stops)	20
Normal Manoeuvre	30	30	200
Continuous residual displacement (straight and level)	10	2 (see Note)	3600 (see para 3.2 9 (vi))

Note: For control systems without autostabilisation the jack stroke may be insignificant in the standard duty cycle at 10% hinge moment.

LEAFLET 203/5

CONTROL SYSTEM - MECHANICAL COMPONENTS

DESIGN RECOMMENDATIONS

1 INTRODUCTION

1.1 The leaflet presents supplementary information with respect to the design of mechanical components in control systems. The aim is to amplify the requirements of Chapter 203 for particular detail design considerations. In many cases the adoption of the recommendations of this leaflet will be found to be necessary to meet the specified requirements.

2 DETAIL DESIGN CONSIDERATIONS

2.1 GENERAL

2.1.1 Chapter 203, para 3 requires that a control system shall be designed to tolerate the possibility of jamming, chafing and interference from personnel, cargo, loose objects and freezing of moisture. These overall considerations should be applied to each detail component.

2.2 PREVENTION OF JAMMING; CLEARANCES

2.2.1 Each control run should be arranged with adequate local clearances to minimise the possibility of jamming by loose objects. Control runs should not be too close to horizontal surfaces, or have levers designed to operate in local pockets. Multiple levers on common spindles should not have lightening holes, or inward facing flanges. Chains with sprockets having a horizontal pivot are undesirable and pulley guards should be sufficiently close to prevent the ingress of small parts.

2.2.2 As a guide the minimum clearances should be:

(i) 3 mm between elements which move in relation to one another but which are guided or connected to the same component,

(ii) 6 mm between elements which move in relation to one another and which are guided or connected to separate components,

(iii) 12 mm between elements and airframe structure or equipment to which they are not attached, unless structural flexibility requires a greater clearance to be provided. For example, 25mm between elements and hydraulic pipes and 40mm between control cables and dynamic components.

2.3 JOINTS

2.3.1 All adjoining parts should be secured in a manner that will prevent loosening when subjected to internal, external and vibration loads. All pin joints should use self-retaining bolts, and be secured using double locking nuts, ie slotted stiffnut and split pin. Clevis pins retained only by split pins are not acceptable.

2.3.2 Control system joints in push-pull systems that are subjected to angular motion, except those in ball and roller bearing systems, should have an ultimate factor of safety of not less than 3.33 with respect to the ultimate bearing strength of the softest material used as a bearing. The approved ratings of ball or roller bearings may not be exceeded.

2.4 PRECAUTIONS AGAINST FAILURE OR DISCONNECTION

2.4.1 If it is assumed that any failure or disconnection is extremely remote then the design should be such that no failure or disconnection could be foreseen under practical circumstances, including errors of operation, assembly or maintenance, for such an assumption to be acceptable. The following considerations should be given:

(i) Choice of materials to avoid undue notch sensitivity or stress corrosion cracking.

(ii) Adequate robustness to cater for all conditions arising, including those due to errors in operation, assembly or maintenance.

(iii) Avoidance of design features which tend to give rise to fatigue effects, including sensitivity to vibration.

2.5 CABLES AND CHAINS

2.5.1 The choice of pulley diameter for use with cables should be such as to preclude the possibility of fatigue arising from bending of the cable and local rubbing together of individual wires in the cable.

2.5.2 Pulleys should not be arranged so that a reverse bend in the cable arises as it traverses from one pulley to another, and the pulleys should be in the plane of the cables so that the cable does not rub against the pulley flange. Cable run out angle should be less than 2.5 degrees.

2.5.3 The wrap angle of the cable round the pulley should be sufficient to ensure that the pulley rotates with movement of the cable. Pulleys and sprocket should be guarded to prevent any cable or chain coming off when slack and jamming or re-engaging incorrectly after slack has been partially taken up.

2.5.4 Every kind and size of pulley should correspond to the cable with which it is used. The pulley diameter should be equal to, or larger than, the minimum recommended bend radius for the cable.

2.5.5 Fairleads should be installed so that they do not cause a major change in cable direction. Minor changes in cable direction can be beneficial, as drumming of cables against the fairlead is eliminated thus assisting in prolonging cable life. Change in direction must be kept to a minimum in order to minimise operating loads and not negate wear benefits.

2.5.6 Where turnbuckles are attached to parts having angular motion they should be arranged so that fouling is positively prevented throughout the range of travel.

2.6 ANTI-FRICTION BEARINGS

2.6.1 Bearing installations should be arranged in such a manner that failure of the rollers or balls will not result in a complete separation of the control. Where direct axial control force application cannot be avoided a failsafe feature should be provided.

2.7 PUSH-PULL AND TORQUE TUBES

2.7.1 The tubes used in push-pull and torque control systems should be adequately vented and drained, or completely sealed, to prevent the accumulation of condensed vapours.

2.7.2 Joints incorporated in torque tube control systems to enable axial movement should have sufficient engagement to ensure that disengagement will not occur and positive drive will be retained under the most adverse set of manufacturing and installation tolerances and structural flexibility. Minimum engagement should not be less than one diameter. A means for inspection of the amount of engagement should be provided.

2.7.3 All torque tubes should be mounted on anti-friction bearings with supported couplers spaced at close enough intervals with sufficient misalignment capability to prevent undesirable bending or whipping of the tubes.

2.7.4 Each torque tube in a link system should be removable and re-installable in the rotorcraft without having to disturb the support, component, or other interfacing system elements at either end of the tube.

2.7.5 The rated operating speed of a torque tube system should be no greater than 75% of the critical speed unless a supercritical design has been agreed with the Rotorcraft Project Director.

2.7.6 Where a broken tube could cause damage to other components an adequate guard should be provided.

2.8 SENSORS

2.8.1 Control system sensors, such as vanes, probes and other mechanical devices should be installed at locations which minimise exposure to conditions which could result in inaccurate output signals or failures.

CHAPTER 204

STRENGTH CONSIDERATIONS FOR AUTOMATIC CONTROL SYSTEMS

1 INTRODUCTION

1.1 The requirements of this chapter are applicable to rotorcraft fitted with any automatic control of the motivators, or automatic control capable of loading the aircraft structure, such as, for example:

(i) Stability Augmentation,

(ii) Autopilot,

(iii) Full Authority Digital Engine Control (FADEC),

(iv) Active Control of Structural Response (ACSR) & etc.

1.2 Such systems may impose additional loads on the Rotorcraft. The effects of these loads shall be considered.

Note: This chapter is only concerned with this particular aspect of automatic control system behaviour. The requirements pertaining to other aspects of their behaviour are specified elsewhere in this standard.

1.3 The correct operation of all systems shall be examined to ensure that the loads imposed on the Rotorcraft or its component parts by the operation of the actuators and, or, motivators are fully understood and that the Rotorcraft structure can be adequately designed to withstand them.

1.4 Multiple systems may produce additive loads in some parts of the structure. Where two or more systems effect a common load path, it will be necessary to consider that the loads in the common load path arise from the addition of the loads from the separate systems. These additive loads combined in the most adverse manner possible shall be considered in the initial design.

1.5 Loads arising from incorrect functioning of automatic control systems shall also be considered.

1.6 Each automatic control system shall be analysed and tested to determine the effect on the Rotorcraft of any failure mode of that system.

2 BASIC REQUIREMENTS

2.1 There shall be no risk of the Rotorcraft structure being overstressed by loads arising from normal use of the automatic control systems.

2.2 The Rotorcraft structure shall have an acceptable fatigue life when subjected to loads imposed by normal operation of the automatic control systems.

2.3 The effects of incorrect functioning in any automatic control system shall be considered.

3 CORRECT FUNCTIONING

3.1 When automatic control systems are in operation, the Rotorcraft shall have proof and ultimate factors in accordance with the requirements of Chapter 200 under the conditions arising from any input, or combination of inputs, of those systems. All operating conditions shall be considered.

3.2 When automatic control systems can be disengaged in flight, the Rotorcraft shall have proof and ultimate factors in accordance with the requirements of Chapter 200 under the most adverse conditions arising after disengagement takes place.

3.3 When automatic control systems are in operation, the Rotorcraft shall have an acceptable fatigue life in accordance with the requirements of Chapter 201 under the conditions arising from any input, or combination of inputs, of those systems. The fatigue spectrum used for the derivation of an acceptable fatigue life for the Rotorcraft structure shall take account of all operating conditions and loads imposed by normal operation of the automatic control systems. Where the specifications of an automatic control system allows for variation in performance of the system then the loads from normal operation shall be the most adverse loads that the system may generate whilst still meeting its specification.

3.4 When automatic control systems can be disengaged with rotors turning either on the ground or in flight, the fatigue spectrum used for the derivation of an acceptable fatigue life for the Rotorcraft structure in accordance with the requirements of Chapter 201, shall contain a reasonable number of occurrences which represent the most adverse conditions arising after disengagement takes place.

4 INCORRECT FUNCTIONING

4.1 The effect on the Rotorcraft of a single failure in any automatic control system shall be considered. All conditions arising from any combination of inputs from the failed system and any other systems shall be determined.

4.2 A safety assessment of each system shall be conducted to identify those failures which could produce inputs resulting in overstressing of, or unacceptable fatigue damage to, the Rotorcraft structure. All combinations of inputs from the failed system and any other systems shall be considered.

4.3 If the conditions arising from a single failure in any system could either reduce the proof and ultimate factors of the Rotorcraft below 1.0 and 1.33 respectively, or subject the Rotorcraft to unacceptable fatigue damage, then either:

(i) The failed system shall be capable of being rendered inoperative or replaced by an alternative system, which is functioning correctly, before overstressing of the Rotorcraft can occur. The disengagement/replacement can be performed automatically or manually. All operating conditions shall be considered. The Rotorcraft shall not be overstressed under the most adverse conditions arising either after disengagement of the failed system or during transition of control to a correctly functioning system.

or

(ii) The likelihood of failure within the system must be predicted with the assurance that it is compatible with maintaining the overall airworthiness standard of the Rotorcraft. This requirement shall be applicable to all automatic control systems which cannot conform to (i) above, eg, systems in which failure is undetectable, systems in which failure could result in overstressing of the aircraft before they could be disengaged/replaced, systems which cannot be disengaged/ replaced, etc.

5 INTEGRITY

5.1 Systems covered by this chapter which are digital shall be classified by the Rotorcraft Design Authority as safety critical or otherwise and constructed in accordance with Chapters 728 and 729.

5.2 In all cases considered, where the systems can be acknowledged to occupy a common load path, then the outputs must be tested to ensure that they are not cumulative. Testing here need not be any more than a theoretical exercise.

6 TESTING - GROUND AND FLIGHT

6.1 The object of testing is to verify the effects of the automatic control systems on the Rotorcraft with respect to strength considerations.

6.2 Ground testing shall be carried out to ensure that the types of failures anticipated have the failure patterns predicted and no other. It will also establish whether or not any summation effects have been adequately compensated for. Ground testing shall be as extensive as possible in order to minimise risk in flight testing.

6.3 Flight testing shall determine if the circumstances in flight are as expected including, where practicable, the complete range of expected failures.

LEAFLET 204/1

AL/5

FLIGHT UNDER AUTOMATIC STABILIZER AND AUTOMATIC PILOT

GENERAL REQUIREMENTS

1 INTRODUCTION

1.1 In this Leaflet the basic requirements given in Chapter 204 are amplified.

2 THE MODES OF OPERATION

2.1 The first stage in the development of automatic controls for rotorcraft was the provision of artificial stabilization to counter the lack of stability of the rotorcraft; the development of the auto-pilot function came later. The two modes of operation of the automatic control system are quite distinct and different airworthiness considerations may apply depending on the mode in use.

3 CORRECT FUNCTIONING

3.1 In the stabilizer mode, no additional safety requirement is specified since, neglecting the small incremental movements needed for stabilization, the controls are applied according to the demands of the pilot, and the controls have similar characteristics to manual controls. In the auto-pilot mode, the basic requirement is similar to that for aeroplanes.

4 INCORRECT FUNCTIONING

4.1 In order that the pilot may quickly regain control in the event of failure in the stabilizer mode, the manual cut-out switch should be located so that the pilot can operate it without altering his grip from the normal flying position.

4.2 In the auto-pilot mode, it has been found that suitable maximum values of servo displacement and rate, can be chosen so that in the event of a runaway, the rotorcraft displacements are not excessive. Furthermore, it can be arranged that the forces needed to overpower a runaway automatic pilot are well within the capabilities of a human pilot.

4.3 The time allowed to elapse before the pilot resumes control after a failure is defined in Chapter 604.

CHAPTER 205

CARRIAGE OF UNDERSLUNG LOADS

1 INTRODUCTION

1.1 The requirements of this Chapter apply to all rotorcraft provided with equipment for carriage of specified underslung loads, whether single-point, two-point or multi-point suspension is used.

1.2 For some single-point installations, where stated in the Rotorcraft Specification or subsequently agreed with the Rotorcraft Project Director in consultation with Airworthiness Division, RAE, the requirements of para 7 may be used in lieu of the requirements of paras 2 and 4 below.

1.3 If an auxiliary cargo hook structure is provided and this is detachable, the requirements apply to the auxiliary structure and its suspension from the rotorcraft.

1.4 For definitions of terms used throughout this Chapter, see Leaflet 205/2.

2 NORMAL CASES

2.1 For each specified underslung load and in consideration of the constraints of the movement of the cg of the load normally exerted by its suspension systems, a design envelope of the form shown in Fig.1 shall be produced. The values to be assigned to each of the parameters which define the design envelope for each specified load will be stated in the Rotorcraft Specification or shall be agreed with the Rotorcraft Project Director.

2.2 When carrying the underslung load, the rotorcraft shall have adequate strength for all conditions of flight arising from consideration of the design envelope.

2.3 The forces arising in manoeuvring between any two points on the design envelope shall also be considered.

2.4 The effect of aerodynamic forces acting on each of the underslung items, including rotor downwash, the position of the centre of gravity and its inertia due to centrifugal and coriolis forces present during rotorcraft manoeuvres, shall be taken into account where relevant.

2.5 Unless otherwise stated in the Rotorcraft Specification, the degree of turbulence to be assumed shall be as defined in Leaflet 205/1 para 5. In calculating the forces acting on the rotorcraft, allowance shall be made for the effects of the specified degree of turbulence acting on:

(i) the rotorcraft,

(ii) the underslung load,

(iii) both together.

2.6 Where relevant, account shall be taken of the dynamic magnification of the forces on the rotorcraft in respect of each of the underslung loads specified in the Rotorcraft Specification, and of any vibration of the suspension system, at any point on the design envelope, which may be caused by:

(i) the forced response of the underslung load to aerodynamic disturbances and transmitted airframe vibration,

(ii) disturbance of the underslung load,

(iii) a disturbance of the rotorcraft.

2.7 It shall be possible to jettison the load during flight in any condition within the design envelope.

2.8 The suspension system shall be capable of discharging to earth any static electricity generated in flight without damage to the equipment.

3 FAILURE CASES

3.1 The effects of the failure of any single element in any of the following shall be considered:

(i) the attachments on the underslung load,

(ii) the suspension system(s) between the underslung load(s) and the rotorcraft,

(iii) the normal release system,

(iv) the emergency release system,

(v) any related rotorcraft system.

3.2 A list of failures to be investigated in depth shall be agreed with the Rotorcraft Project Director.

3.3 The investigation shall take account of the dynamic and static forces in the remaining elements of the system and the effect of these forces on the strength, stability, controllability and handling of the rotorcraft. Where the failure does not inevitably result in dropping the load, it shall be possible after a delay of 5 seconds (between the failure and correct action) for the pilot to recover from the effects of the failure within the appropriate design or flight envelope (see Leaflet 205/1, para 6).

3.4 The maximum immunity from pilot-induced oscillation shall be given by suitable design of the collective control system.

4 STRENGTH FACTORS

4.1 NORMAL CASES

4.1.1 The rotorcraft as a whole and the underslung load suspension system shall have proof and ultimate factors of not less than 1.5 and 2.0 on the loads derived from each of the cases arising from consideration of the requirements of para 2 (see Leaflet 205/1 para 8).

4.2 FAILURE CASES

4.2.1 The rotorcraft as a whole and the underslung load suspension system shall have an ultimate factor of not less than 1.0 on the cases arising from consideration of the requirements of para 3.

5 FATIGUE STRENGTH

5.1 The fatigue life of those parts of the rotorcraft structure subjected only to the forces arising from the carriage of underslung loads and of the removable auxiliary structure, if provided, shall be established in accordance with the requirements of Chapter 201, to a load spectrum to be agreed with Airworthiness Division, RAE. The fatigue life of the fixed parts shall be at least equal to the specified fatigue life of other parts of the rotorcraft structure. The fatigue life of the removable auxiliary structure, if provided, shall be agreed with the Rotorcraft Project Director (in conjunction with Airworthiness Division, RAE).

5.2 When formulating fatigue spectra for other parts of the rotorcraft, the effects of carriage of underslung loads shall be taken into account to the extent agreed in discussion with Airworthiness Division, RAE.

6 MODEL AND FLIGHT TESTING

6.1 Consideration shall be given to the need for model tests and/or flight tests . Appropriate recommendations shall be agreed with A & AEE and submitted to the Rotorcraft Project Director whenever:

(i) the calculated safe flight envelope for a particular underslung load is less than the design envelope,

(ii) insufficient information is available to enable service limitations to be defined.

7 SINGLE-POINT SUSPENSION

7.1 STRENGTH FACTORS

7.1.1 The rotorcraft as a whole and that part of the underslung load suspension system associated with the rotorcraft shall have a design ultimate load which is the lesser of:

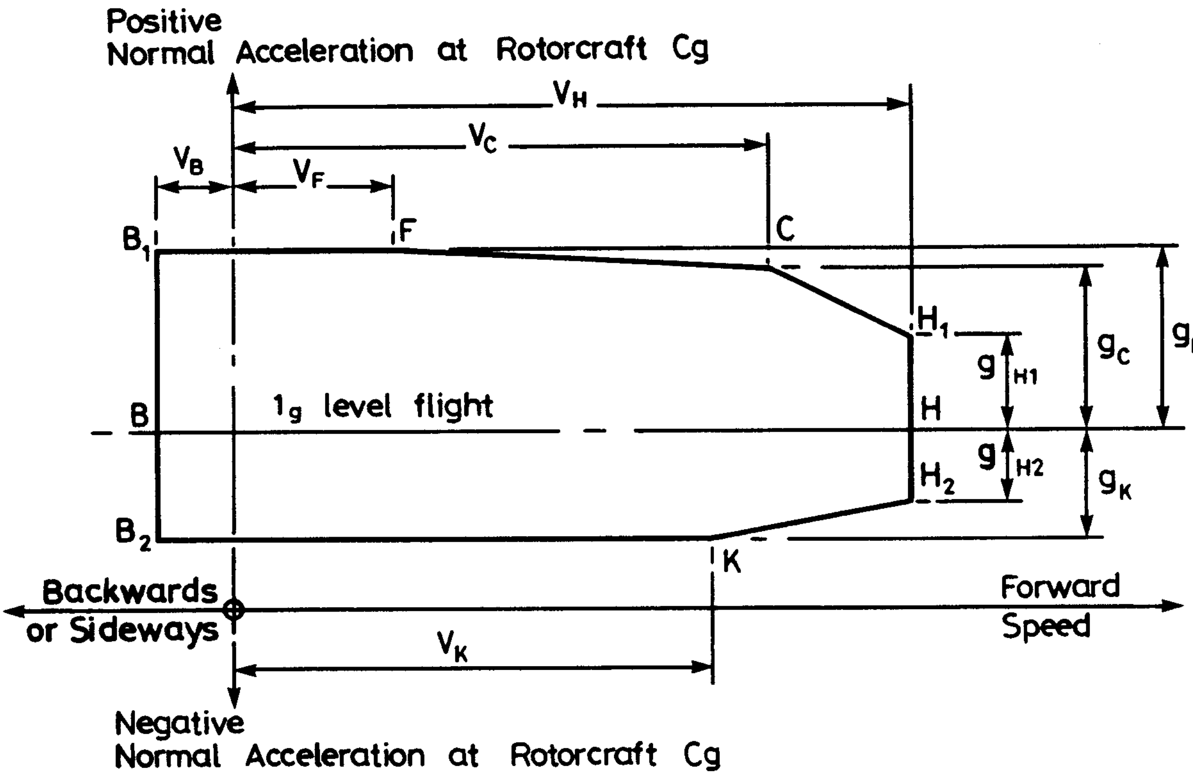
(i) the design thrust of the rotor multiplied by an ultimate factor of 1.5

(ii) the load arising from applying an ultimate factor of 2.0 to the mass of the heaviest underslung load to be carried at a value of g_F not less than 2.0. The associated proof factor shall be 1.5.

The underslung load suspension system and its local back-up structure in the rotorcraft shall be designed by proof and ultimate loads calculated as in para 7.1.1(ii), with inertia relief being allowed on the loads applied to the rotorcraft structure away from the local back-up structure.

7.2 FORCE VECTORS

7.2.1 The factored forces of para 7.1.1 shall be assumed to be applied to the rotorcraft in any direction within a cone of semi-angle 30° having its apex at the suspension point on the rotorcraft and its axis normal to the rotorcraft horizontal axis.



V_H = Maximum design airspeed for carriage of underslung loads.

V_C = Maximum cruising speed for carriage of underslung loads if not equal to V_H.

V_F = Maximum forward airspeed (including headwind) during raising.

V_B = Maximum backward airspeed (including tailwind) but see Note 2 below.

V_K = Maximum airspeed (including headwind) at which the g_K requirement is to be met.

g_C = Maximum acceleration during manoeuvres if not equal to g_F.

g_F = Maximum incremental or snatch acceleration.

g_K = Maximum negative normal g increment.

g_{H1} = Maximum positive normal g increment at V_H.

g_{H2} = Maximum negative normal g increment at V_H.

Note 1: The values of g_{H1} and g_{H2} are discussed in Leaflet 205/1 para 5.1. The values of other parameters are discussed in Leaflet 205/1 para 2.2.

Note 2: Normally V_B will also be the maximum sideways airspeed (including crosswind) unless otherwise specified.

FIG.1 DESIGN ENVELOPE

LEAFLET 205/1

CARRIAGE OF UNDERSLUNG LOADS

GENERAL REQUIREMENTS

1 GENERAL

1.1 This Leaflet gives background information in relation to the requirements of Chapter 205.

1.2 Carriage of underslung loads has been developed over a number of years and the systems considered range from a single-point attachment of the underslung load with a single-point suspension from the rotorcraft, to multi-point attachment of the underslung load with four-point suspension from the rotorcraft. In some systems, multi-point attachment may be associated with single-point suspension and vice versa. In others, an intermediate beam is interposed between the underslung load and the rotorcraft and attached to the rotorcraft at a number of points. Any such intermediate structure creates two interfaces which should be considered separately from the point of view of determining the airworthiness of the system as a whole. In most cases, some of the elements will have been designed separately according to their own applicable requirements, but it will be necessary to ensure that all are strong enough for carriage throughout the flight envelope of the specified underslung loads.

1.3 The essential elements of the system are a strong point or points or an auxiliary structure on the rotorcraft; cables, strops or slings attached to this strong point by hooks, eyes, swivels, shackles or links; and a similar system of attachment of them to the underslung loads. In some cases a winch or hoist may be provided.

1.4 It is not the purpose of DEF STAN 00-970 to state detailed requirements for the strength of the underslung load itself, or of any container in which service equipment may be carried, though clearly both must be suspended in such a manner that they do not break nor deflect unacceptably under the forces imposed on them by the rotorcraft during the period in which these loads are likely to occur. Although this is the responsibility of the designer of the underslung load, the rotorcraft designer must be supplied with sufficient information to enable him to assess its effect on the rotorcraft and to establish whether, for example, in certain cases special fins or drogues may be necessary for the safe carriage of the load. Failure of part of the underslung load itself will have to be considered if this is likely but in this case the failure will be defined in the Rotorcraft Specification. Information on requirements for slinging of items of equipment which may become underslung loads on rotorcraft is given in Ref. 4.

1.5 Carriage of underslung loads involves the following operational manoeuvres:

(i) Hook-on.

(ii) Raising.

(iii) Forward acceleration and climb-out.

(iv) Transit with flight manoeuvres.

(v) Deceleration and descent.

(vi) Emplacement.

(vii) Rapid manoeuvres.

(viii) Jettison in an operational emergency or failure state.

2 DESIGN ENVELOPE

2.1 The requirements presume that any new installation for the carriage of underslung loads will be designed with a small number of specific, probably new, underslung loads in mind which will be listed in the Rotorcraft Specification. For each load a different set of service conditions may apply. For this reason it may not be possible to work to one design envelope and each load may have to have a separate envelope. However, it may be possible for the strength of the installation to be based on one of these or on an amalgam of all of them. In any case these design envelopes should not be confused with the flight envelopes to which the individual loads may be restricted when, after all tests are complete, they are released for service use.

2.2 Each point on each design envelope defines a flight condition in which the forces acting on the rotorcraft must be considered. However, the forces arising in a manoeuvre from one point to another can be significant. In particular large structural forces may occur during raising. Even with elastic strops snatch loads of more than twice the weight of the underslung load can easily arise and with stiffer materials they could be several times greater. It is therefore mandatory that consideration shall be given to the maximum forces which could arise in this manoeuvre. Note that V_F will be greater when operating from ships than it is when operating on land. It is also necessary to consider the forces in a flight manoeuvre or gust arising from a change from rapid descent represented by a point on B_2K to rapid ascent as represented by a point on B_1F in the design envelope. The largest structural forces in manoeuvres other than these are most likely to arise from consideration of a banked turn or pull-up manoeuvre at the points C and H. At both these points it may be necessary to determine special limitations for service use. All the information required to define each design envelope numerically will be stated in the Rotorcraft Specification. For some projects this may be simplified by defining a rectangular envelope in which $V_C = V_K = V_H$ and/or $g_F = g_C = g_{H1}$ and/or $g_K = g_{H2}$. On the other hand, for some projects, it may be necessary to do gust analysis in some detail before the accelerations g_{H1} and g_{H2} can be specified.

3 DYNAMIC EFFECTS

3.1 When the rotorcraft accelerates or decelerates, a fore-and-aft pendulum oscillation of the underslung load may be initiated. Similarly lateral oscillations may be initiated. With most underslung loads and particularly with long loads the yawing mode of the suspension system may be excited by aerodynamic disturbances. In either case the resultant motion can either become rapidly divergent or stabilize at an angle. Equally a yawing oscillation may be excited by a disturbance during a turn. In addition to the inertia forces there will be aerodynamic forces derived from the lift, drag, and pitching moment of the underslung load, and these will not only modify the forces in the rotorcraft structure but will also effect its stability and performance. Sudden instability can occur in the case of flat loads and especially with loads possessing aerodynamic properties such as wing sections. The reaction to the sudden relaxation of forces, caused by jettison at any point in the flight envelope, should also be considered. If simultaneous action of the releases in a two-point suspension system cannot be guaranteed in the event of a single failure at any point in the system, consideration must be given to the behaviour of the rotorcraft and of the underslung load during the delay before the emergency system is operated. An oscillation in any mode of flight may also be initiated by one of many effects of turbulence. Examples are: passing a large building (even in light winds); wake effects of ships; entering a ravine. As the variety of specified underslung loads requiring consideration may be large, detailed calculations for each configuration may be impossible. Nevertheless some consideration of these dynamic conditions must be given to each underslung load and detailed consideration should be given to those where there is any reason to suspect that a divergent oscillation could be initiated.

4 RESONANCE

4.1 Another dynamic problem concerned with the normal carriage of underslung loads which should also be investigated concerns the possibility that the underslung load may start to resonate on its strops under the influence of forcing frequencies derived from the rotorcraft. All engine/transmission/blade modes should be considered in relation to the estimated natural frequency of the suspension system, the flight envelope, and the conditions in which the underslung load is required to be carried. The underslung load and suspension system will have more than one natural frequency. The addition of the underslung load may alter the vibration modes of the rotorcraft and hence the level of the forcing frequencies applied to the underslung load. If resonance is expected to occur, then the problem may be overcome by changing the material or the length of one or more of the elements of the suspension system or by restricting the flight conditions within which the Service are permitted to carry the underslung load. In this connection the designer should consider existing Service equipment first and use it where possible. Special strops should only the recommended if this is unavoidable. Where this is necessary, the development and testing of these strops should be discussed with the Rotorcraft Project Director.

5 TURBULENCE

5.1 In the absence of better data, the following model of atmospheric turbulence from Ref.1 may be taken as representative of the level specified:

Vertical Component:	RMS σ_W in ft/sec	3.2	6	8.2	12
	Proportion of time P in σ_W	50%	13%	6%	0.6%

Lateral Component: $\sigma_V = \sqrt{1.4} \times \sigma_w$ with respect to mean wind

Longitudinal Component: $\sigma_n = \sigma_W$

Where a discrete gust seems more appropriate, a 20 ft/sec sharp-edged gust is suggested. The incremental accelerations of the flight envelope g_{H1} and g_{H2} should be assumed to be 0.4 g unless more relevant data are available. In this case the Rotorcraft Specification will state appropriate values. It is important that the values chosen for these incremental accelerations allow for both gust and manoeuvrability margins.

6 FAILURE CASES

6.1 Whether single-, twin- or multi-point suspension or attachment is proposed the physical failure of any one element and its effect on the subsequent behaviour of the rotorcraft and the underslung load (including both longitudinal and lateral stability) must be considered. It should be a design aim to contain as many failures as possible without having to resort to jettison of the load. This is particularly applicable to failures in the auto-pilot system but should be applied equally to the suspension system as far as possible. If the failure of a single element of the suspension system results in the automatic dropping of the load (as in the case of failure of certain elements in a single point suspension system), then recovery should be possible within the normal flight envelope for the rotorcraft. If the failure of a single element of the suspension system merely transfers more load to other elements (as in the case of failure of a single strop in a four-strop system), the dynamic and structural consequences of the failure must be considered and recovery from any consequent disturbance must be possible within the design envelope for carriage of underslung loads. In either case it may take the pilot up to 5 seconds to appreciate the situation and act correctly to reduce speed or to jettison the load. In both cases consideration must be given to all relevant normal conditions of the stability augmentation system if one is fitted. It is not the intention that the requirements should involve consideration of improbable failures particularly in related systems. However, all possible failures should be considered in drafting the list of those failures which will be examined in depth. This includes the release units which also present special problems discussed in para 7 below.

7 FAILURE OF RELEASE UNITS

7.1 Where two or more release units are provided to carry the underslung load and all must operate simultaneously to jettison the underslung load in emergency, then a serious hazard exists if simultaneous action between the units is not achieved because of sticking or inadvertent operation of one unit. In these cases it is essential that a full investigation of the effect of a time delay between the operation of the releases must be undertaken. The effect of this delay on the stability and performance of the rotorcraft during the delay period must be considered. It will be necessary to show that the rotorcraft can be controlled within this period, that it will not be hazarded by the behaviour of the load, and that after the load is finally released the response can also be contained. The effect of any corrective action needed by the pilot will also need to be taken into account. Evidence to date (Ref.3) suggests that for some loads, the maximum time delay which can be accepted may be half a second. However the time which elapses before the rotorcraft is seriously hazarded by out-of-trim forces or excessive structural loading can be altered considerably by careful selection of strop length and material.

8 STRENGTH FACTORS

8.1 The normal factors specified are 1.5 and 2.0 because, even with the full dynamic investigations required, there will remain some uncertainty in the stressing cases which will be derived from these requirements. However, a reduction of these factors may be justified if appropriate instruments are provided which improve the pilot's ability to keep within the flight envelope and other limitations, or if the requirements create a new design case for the rotorhead or other parts of the primary load path. In either case the designer should discuss the question with Airworthiness Division, RAE.

9 FATIGUE REQUIREMENTS

9.1 The requirements aim to ensure that all fixed parts of the suspension system and the back-up structure have a life as long as that of the rest of the rotorcraft structure. If a suspension system is designed to be easily removable and only fitted for a proportion of the life of the rotorcraft, these parts may have a lower life, subject to the agreement of the Rotorcraft Project Director (in conjunction with Airworthiness Division, RAE). In this case, as the parts could be transferred to other rotorcraft, consideration will have to be given to logging their utilisation separately.

9.2 The movement and attitude of underslung loads beneath the rotorcraft may impose additional stresses on airframe, transmission and rotor systems which, in the case of a steady condition with a stable load are primarily a result of a revised mass distribution and trim requirement determined by the position of the suspension systems and the steady aerodynamic forces on the loads. It is the Rotorcraft Designer's responsibility, in consultation with the Rotorcraft Project Director, Airworthiness Division, RAE and A and AEE, to determine the effect of underslung load carriage, including manoeuvres and transient/oscillatory conditions on component fatigue lives and define any flight limitations or additional instrumentation necessary to limit fatigue life penalties.

9.3 The extent of testing required to supplement and confirm calculations will be determined in accordance with para 10.

10 MODEL AND FULL-SCALE FLIGHT TESTING

10.1 The permutations of the variety of the underslung loads and associated slinging equipment listed in the Rotorcraft Specification which the Service may require to carry in addition to those for which the installation is basically designed, of the normal and emergency stressing cases, and of the handling techniques to be used in each case, will usually be too numerous to permit exhaustive testing of every combination. The designer must therefore satisfy himself by calculations that the associated flight limitations and method of suspension are adequately defined in each case and the appropriate limitations stated for each of the specified loads. Wind-tunnel model tests can be used to supplement calculations and predict performance and load behaviour. Where these methods are inadequate, or insufficiently accurate, consideration will need to be given to full-scale flight testing. The extent of this flight testing will be agreed between the Contractor, the Rotorcraft Project Director and A & AEE. It is the Contractor's responsibility to advise the extent to which trials are necessary to establish design criteria and to draft a proposed flight test programme. The only positive requirement stated in mandatory terms is to this effect. There will, in any case, be some permutations where Service use within a restricted flight envelope is considered safe by analogy with other loads or the same load on another rotorcraft but for which flight tests will be essential to extend the envelope.

11 WINCH INSTALLATIONS

11.1 Where a winch or hoist is used as a strong point for carriage of an underslung load, all the requirements of Chapter 205 are applicable unless the Rotorcraft Specification states otherwise. However this may be a particular case where the relaxation of Chapter 205, para 1.2 may be useful.

12 SINGLE-POINT SUSPENSION

12.1 The arbitrary requirements stated may be inappropriate when considered in relation to some designs. If the cone angle specified is excessive or if the axis is unrealistic in the vertical acceleration implied by the requirements, then the problem should be discussed with the Rotorcraft Project Director. However, g_F represents not only a manoeuvre case during climb-out or descent, but also a snatch during raising and a dynamic condition in transit.

REFERENCES

No	Author(s)	Title, etc
1	J K Zbrozek RAE Tech Note No Aero 2682 March 1960	The relationship between the discrete gust and power spectra presentations of atmospheric turbulence with a suggested model of low altitude turbulence.
2	D A Webber RAE Tech Report No 65190	Comparison of helicopter and aeroplane vertical accelerations in turbulence.
3	D F Sheldon and J Pryor RMCS Tech Note AM33 Sept 1972	Test report on failure trials of a two-point load suspension system of a helicopter.
4	Ministry of Defence	Defence Standard 00-3 Transportability of equipment.
5		STANAG 3542 HIS Technical criteria for the transport of cargo by helicopter.

LEAFLET 205/2

CARRIAGE OF UNDERSLUNG LOADS

DEFINITIONS RELATING TO UNDERSLUNG LOADS

1 UNDERSLUNG LOAD

AL/5

1.1 Any items of equipment required to be carried beneath a rotorcraft attached by slings or strops to one or more release units or hooks.

2 RELEASE UNIT

2.1 A special hook having means of opening (release) controlled by the pilot and, if required, by the crewman.

3 SINGLE-POINT SUSPENSION

3.1 An underslung load carrying system whereby the load is suspended from a single release unit or hook.

4 TWO-POINT SUSPENSION SYSTEM

4.1 An underslung load carrying system whereby the load is suspended from two release units or hooks.

5 MULTI-POINT SUSPENSION SYSTEM

5.1 An underslung load carrying system whereby the load is suspended from three or more release units or hooks.

6 STROP

6.1 A single length of cable/rope/webbing with suitable attachments (hooks, shackles) at each end, used to suspend an underslung load from a release unit. It may be used alone or in conjunction with other strops to form a sling.

7 SLING

7.1 An arrangement of strops (legs) of cable, etc. attached at one end to a common shackle or similar fitting, each leg having a fitting (hook, etc.) at the lower end. Used to attach an underslung load to a release unit.

8 HOOK-ON

8.1 The act of attaching the load (by slings or strops) to the release unit or hook.

9 EMPLACEMENT

9.1 The act of depositing the underslung load on the ground/landing site and releasing it from the release unit.

10 NORMAL RELEASE

10.1 The normal means of actuation of the release unit(s) or hook by the pilot or crewman.

11 EMERGENCY RELEASE

11.1 A secondary means of actuating the release unit should the normal release fail to operate.

12 JETTISON

12.1 Release of the underslung load in any flight mode in an emergency using either the normal or emergency release system.

CHAPTER 206

DESIGN TO RESIST BIRDSTRIKE DAMAGE

1 INTRODUCTION

1.1 This Chapter specifies the minimum requirements for the resistance of rotorcraft airframe systems and structure to birdstrikes. Engine and weapons requirements are not included.

1.2 V_{ne}, is defined as the Never Exceed Speed. (See Chapter 900, para 7.5.2 (i)).

1.3 The speed of impact is defined as the relative speed of the bird and rotorcraft structure at impact.

2 FUNDAMENTAL REQUIREMENTS

2.1 FLYING QUALITIES

2.1.1 The primary damage, or secondary effects, of a single birdstrike shall not degrade the flying qualities of the rotorcraft below LEVEL 2 as defined in Chapter 600, para 7.

2.2 MAINTENANCE EFFECTS

2.2.1 Any damage caused by a birdstrike shall be readily repairable at unit level. Repair by replacement of complete assemblies will be acceptable provided methods of temporary repair are available, subject to agreement of the Rotorcraft Project Director.

3 THE THREAT

3.1 The threat for the purpose of these requirements is a single strike by a bird of 1.0 kg mass.

4 DETAILED REQUIREMENTS - FORWARD FLIGHT

4.1 TRANSPARENCIES AND SUPPORTING STRUCTURES

4.1.1 This para applies to all transparencies with a significant frontal aspect in forward flight, except radomes (see para 4.6).

4.1.2 Following, a birdstrike at rotorcraft speeds up to V_{ne}:

(i) There shall be no penetration by the bird.

(ii) Transparencies shall not exhibit any cracking, delamination or other failure that would reduce the residual strength below the design limit load.

(iii) Windscreens and other transparencies essential to the pilot and crew to complete the mission safely, shall retain optical properties adequate for the completion of the mission under all weather and light conditions.

4.2 FUSELAGE

4.2.1 The fuselage structure shall include the cockpit, cabin, tail and all external equipments. Following a birdstrike at rotorcraft speeds up to V_{ne}:

(i) There shall be no penetration by the bird.

(ii) There shall be no damage which would prevent the completion of the mission.

(iii) There shall be no damage to the engine intakes sufficient to result in a significant deterioration in engine performance.

4.2.2 Following a birdstrike, at rotorcraft speeds up to V_{ne} there shall be no detachment of any part of the structure, including aerials, doors or panels likely to cause secondary damage to the tailrotor (and engines).

4.3 MAIN ROTOR BLADES AND ROTOR HEAD ASSEMBLY

4.3.1 These are defined as the rotor system and associated controls which are not protected by structure defined in para 4.2.1 and are exposed to direct impact by birds.

4.3.2 At rotorcraft speeds up to V_{ne} there shall be no significant damage to the main rotor blades and rotor head assembly from birdstrikes from any direction.

4.4 TAIL ROTOR BLADES AND ROTOR HEAD ASSEMBLY

4.4.1 These are defined as the tail rotor system and associated controls which are not protected and are exposed to direct birdstrikes from any direction.

4.4.2 At rotorcraft speeds up to V_{ne} there shall be no significant damage to the tail rotor system and associated controls, from birdstrikes from any direction.

4.5 SYSTEMS

4.5.1 At impact speeds up to V_{ne} in forward flight birdstrikes shall not result in interference with flight instruments, flying controls, other systems or sensors likely to hazard the rotorcraft or prevent it from completing its mission.

4.5.2 The vibration level caused by any birdstrike damage shall not prevent the completion of the rotorcraft's mission.

4.6 RADOMES

4.6.1 Birdstrikes at rotorcraft speeds up to V_{ne} are permitted to cause delamination of composite radomes, but there shall be no penetration. If it is not possible to achieve both this requirement and a satisfactory electromagnetic transparency the Rotorcraft Project Director shall be informed.

5 DETAILED REQUIREMENTS - SIDEWAYS AND REARWARDS FLIGHT

5.1 Birdstrikes at rotorcraft speeds up to V_{max} in sideways or rearwards flight shall not result in any significant damage to any part of the rotorcraft or adversely affect any part of the structure, flying controls, rotors, systems or sensors.

6 TESTING

6.1 The methods and the extent of testing required to demonstrate compliance with these requirements shall be established by agreement between the Rotorcraft Project Director and the contractor.

CHAPTER 207

ACTIVE CONTROL SYSTEMS

1 GENERAL REQUIREMENTS (see also Leaflet 207/1)

1.1 INTRODUCTION

1.1.1 This Chapter contains the requirements relating to Rotorcraft flight with Active Controls. Reference should be made to Chapters 204 and 604 which deal with other aspects of flight control, automatic flight control, and related flying qualities. Definitions of terms used are at para 8.

1.1.2 Active Control Systems (ACS) may be broadly classified as those control systems in which commands to the control motivators are continuously computed from sensor inputs both with and without pilot inceptor inputs. The title therefore naturally embraces both Manual Flight Control and Automatic Flight Control Systems (Chapter 203). However, this chapter is only concerned with the most critical of control systems i.e. those which must operate continuously (full time) and/or those without whose correct operation safe flight cannot be maintained.

1.1.3 ACS may provide functions other than flight control. In all cases the requirements of this chapter will apply if maintenance of safe flight is involved.

1.2 APPLICATIONS (see also para 7)

1.2.1 Full-time Active Flight Control systems must provide vehicle control and, where necessary, stability augmentation. They may also provide assistance to the pilot in respecting rotorcraft flight envelope limits (Carefree Handling) and control demands to engines. These additional functions may, however, be designed as discrete systems, may not be full-time and may not have flight safety implications.

1.3 FUNCTIONAL REQUIREMENTS

1.3.1 The designer must provide performance, to the levels stated in the Rotorcraft Specification. An ACS may be required to function at the following levels of integrity:-

(a) Full-time
(b) With reversion to equivalent performance
(c) With reversion to degraded performance
(d) Fail-safe (for systems other than primary flight control)

1.3.2 For systems involving reversion, as in 1.3.1(b) and (c) above, the integrity of the reversionary system and the ability to make the essential transition to it shall be considered to be flight safety critical.

1.4 REFERENCE TO OTHER AREAS

(Categories of Status, Performance, Flying Qualities, Handling Qualities to be added later).

2 SYSTEM REQUIREMENTS (see also Leaflet 207/2)

2.1 GENERAL REQUIREMENTS

2.1.1 In considering the design of an ACS it is important to understand that effective ACS operation will depend upon the compatible interactions of pilot, airframe, ACS and the total possible environment.

2.1.2 ACS will include some or all of the following features:

- Pilot's inceptors, switches, and indicators required for operating the ACS.
- Motion sensors, e.g. rate gyros, accelerometers, airstream direction sensors.
- Transducers and sensors required for command and measurement of outputs.
- Structural strain sensors.
- Air data sensors.
- Computers and system management logic.
- Status data storage.
- System power supply conditioning.
- Actuators.
- Environmental control system.
- Electrical power supply.
- Hydraulic power supply.
- Engine functions.
- Signalling elements: mechanical, electrical, optical etc.
- Software.
- Self-Test and Maintenance facilities.

2.2 INTEGRITY

2.2.1 The system shall be designed to meet specified levels of integrity to be agreed with the Rotorcraft Project Director.

2.2.2 A detailed analysis of the System Design in respect of integrity shall be presented to show the basis upon which the Rotorcraft Specification is to be met. In particular the philosophy for treating common-mode failures shall be stated and shall be to the approval of the Rotorcraft Project Director.

2.2.3 It shall be shown that the ACS can meet the defined integrity requirements after allowing for any reasonably probable failures of other systems with which it interfaces such as electrical power and hydraulic supplies.

2.3 RELIABILITY

2.3.1 The system shall be designed to meet the specified levels of reliability, to be agreed by the Rotorcraft Project Director.

2.3.2 A full reliability development plan shall be prepared to show how the full reliability potential of the design is to be realised, to establish compliance with the requirements, and to specify how the reliability achievement will be maintained throughout production and service. (See DEF STAN 00-40 (Part 1) (NATO ARMP -1)).

2.3.3 The ACS design shall take account of maintenance requirements under all anticipated environmental conditions.

2.4 INVULNERABILITY

2.4.1 The system shall be designed to survive all threats defined in the specification both individually and in the worst combination. The susceptibility levels for all environmental influences shall be determined, and all practical limitations affecting effectiveness and survivability shall be declared to the satisfaction of the Rotorcraft Project Director as early as possible in the design and development process.

2.4.2 The systems design shall take full account of its installation in the airframe, variations in the operating environment including the effect of failures of and within other on-board systems, the potential for maintenance error, flight crew error, enemy action and flight in lightning conditions.

2.4.3 Wherever possible survival shall be achieved by providing inherent passive resistance to the influencing factors.

2.4.4 In the event of the rotorcraft sustaining damage, the probability of the ACS providing less than Level 3 flying qualities shall be to the satisfaction of the Rotorcraft Project Director.

2.4.5 Where parallel processing or control paths are provided to ensure redundancy, they should be physically separated as far as possible, so that minor accidental or battle damage does not cause failure of the active control system.

2.4.6 In designing resistance to Electromagnetic Interference (EMI) consideration shall be given, in particular, to the following:

(i) An Electromagnetic Compatibility (EMC) test and EMI avoidance philosophy must be stated by the designer.

(ii) Effects on performance, both physical and operational effects, must be considered during system design.

(iii) The designer must show that his design meets the specified threats as part of an overall system safety analysis (ref para 2.2.2).

(iv) The effect of the requirements on choice of system must be clearly indicated to the Rotorcraft Project Director early in the design stages.

2.4.7 The ACS and its installation, together with all other installed equipments shall be mutually electromagnetically compatible when installed in the rotorcraft; and demonstration of their satisfactory integration into the total rotorcraft is required.

2.4.8 Maintenance of essential screening and bonding effectiveness must be considered and means for in-service testing incorporated into the design.

2.4.9 Exceptionally, where compliance with requirements for invulnerability proves impracticable, it shall, with the Rotorcraft Project Director's approval, be permissible to adopt such strategies as shut-down and reinstatement to achieve survivability.

2.4.10 To avoid ACS failure due to maintenance error, the system shall be designed so that it is impossible to install or connect any component part improperly.

2.5 POWER SUPPLIES

2.5.1 Hydraulic, pneumatic and electrical power shall be provided to the system with characteristics, levels of integrity and reliability compatible with system requirements.

2.5.2 Power transients shall not degrade the performance of the ACS.

2.5.3 Consideration shall be given to the effects of engine failure upon power supply functioning. Sufficient non-engine driven power supplies shall be provided to meet the integrity levels agreed with the Rotorcraft Project Director (see 2.2).

2.5.4 Essential flight controls shall be given priority over non-flight phase essential controls in apportioning the use of power supplies.

2.6 COMPUTERS

2.6.1 The response of the ACS to both pilot commands and external disturbances shall be determined by signal computation effected by computers. Feedback signals from specially provided sensors shall enable the computers to take account of the motion of the rotorcraft, changes in configuration and airframe loads as necessary for correct functioning of the ACS.

2.6.2 ACS computers shall be designed to provide speed of operation and levels of discrimination fully compatible with the intended performance of the control laws, and they shall enable all management functions to be effective without incurring significant penalties arising from time delays

2.6.3 The designer shall ensure that all functions embedded in a processor or computer are identified and fully understood with respect to their possible influence upon correct functioning of the ACS, whether or not these functions are intentionally employed during normal ACS operation.

2.6.4 Each ACS computer, as an LRU and its interface, shall be uniquely identified, such that its correct function when installed in a rotorcraft system is assured.

2.6.5 The ACS computer design and its interface shall have sufficient reserve or be capable of growth, to meet later expansion requirements to be agreed with the Rotorcraft Project Director.

2.7 MOTION SENSORS

2.7.1 The locations of primary sensors in the rotorcraft shall be chosen making due allowance for the measurement of rigid and flexible body motion, and the achievement of adequate protection from bird-strike, accidental and battle damage. Any necessary compromise shall be to the satisfaction of the Rotorcraft Project Director.

2.7.2 Where use is made of sensors already provided for other systems any resulting compromise in the ACS rotorcraft performance shall be to the satisfaction of the Rotorcraft Project Director.

2.7.3 Where airspeed or airflow sensors are used, consideration shall be given to the effects of rotor downwash on sensor performance in the low speed/hover flight regime.

2.7.4 Where different sensor types are used to measure the same parameter the designer shall ensure that the sensor inputs are compatible throughout the rotorcraft operating envelope.

2.8 INCEPTORS

2.8.1 The design of inceptors shall meet the performance, reliability and integrity requirements specified for the system. For novel inceptors, operating characteristics shall be to the approval of the Rotorcraft Project Director. The operation of control switches etc., mounted on flight control inceptors shall be shown to produce minimal interference with inceptor output signals.

2.8.2 Where appropriate, upon failure of a feel system, reversion shall be made to, at worst, the least acceptable breakout force and gradient defined in the Rotorcraft Specification.

2.8.3 Dual Control Inceptors

(i) Where mechanical interconnection between inceptors is impracticable (e.g., between force sensors or where physical constraints do not permit) the designer shall advise the Rotorcraft Project Director, as early as possible of the method of interfacing inceptors with the control law computing.

(ii) The interface logic should whenever possible avoid the use of manually operated inhibiting switches.

(iii) Inhibition of primary inceptors shall be avoided, (for example the instructor's or primary pilot's).

(iv) The effects of dual command upon control law authority and stability shall not lead to over controlling or system instability.

2.8.4 The effects of pilot-coupling mass shall be shown not to degrade precision control during rapid manoeuvres and flight in turbulence.

2.8.5 The inceptors shall be designed for ease of use and ergonomic acceptability. Realistic simulation trials shall be undertaken prior to flight to gain confidence in the ergonomic acceptability of the inceptors.

2.8.6 The inceptors and their installation shall be designed so as to avoid injury to the crew during crash conditions. The definition of crash loadings and accelerations shall be agreed with the Rotorcraft Project Director.

2.8.7 The inceptors and their installation shall not unduly hinder ingress and egress from the cockpit in normal or emergency conditions. Any special measures taken to facilitate ingress/egress, such as hinged mountings, shall be designed so as to avoid inadvertent operation and the introduction of additional failure modes.

2.9 ACTUATORS

2.9.1 The actuators shall meet the requirements of the specification which shall at an early stage be amplified by an interface document to ensure compatibility of all mechanical, hydraulic, pneumatic, electrical and electronic interfaces.

2.9.2 Demonstration of actuator performance shall be carried out in a rig which is fully representative of control and installation parameters.

2.9.3 Where redundancy is used to meet integrity targets, care shall be taken that performance after failure is acceptable. This is to be demonstrated by testing of simulated failure modes.

2.9.4 Particular attention shall be given to the maintenance of performance with service life and to the accommodation of the duty cycle expected to result from the control laws in use.

2.9.5 The requirements of paras 2.11, 3.2.4 and 3.2.5 shall also be noted.

2.10 COMMUNICATION SYSTEM

2.10.1 An ACS typically makes use of non-mechanical signalling between inputs and outputs.

2.10.2 The forms of signalling adopted, e.g., electrical or optical, shall be shown to provide the required integrity and invulnerability.

2.10.3 Where electrical signalling is used, due consideration shall be given to:-

(i) The need to provide adequate protection from electromagnetic hazards.

(ii) The need for maintenance actions during the life of the rotorcraft.

(iii) The need to provide acceptable power budget margins for the life of the rotorcraft.

2.10.4 Where optical signalling is used, due consideration shall be given to:-

(i) The need to provide acceptable power budget margins for the life of the rotorcraft.

(ii) The need for maintenance actions and battle damage repair during the life of the rotorcraft.

2.11 FAILURE MANAGEMENT

2.11.1 Where redundancy is employed, special care shall be taken to eliminate sources of common-mode failure. This provision does not prohibit the use of interlane transfer of information, but it does mean that methods to achieve this shall be designed and implemented to meet identified integrity, reliability and performance levels.

2.11.2 Interlane tracking errors and voting transients shall be kept to a minimum; in particular, errors between sensors measuring the same parameter shall not compromise required maximum allowable failure transient levels.

2.11.3 Where degradation of performance due to failures is unavoidable, the designer shall aim to achieve a gradual regression in flying qualities between normal and emergency characteristics.

2.11.4 The effects of failures upon the motion of the rotorcraft shall be minimised and shall not exceed the levels stated in the Rotorcraft Specification. It is important that first failures shall incur minimal disturbance, subsequent failures may be permitted to give rise to greater transients but these shall be to the satisfaction of the Rotorcraft Project Director.

2.11.5 The means whereby failures are detected and isolated shall be designed so that the compromise between failure transient level and nuisance disconnect frequency is acceptable to the Rotorcraft Project Director.

2.11.6 The designer shall ensure that nuisance disconnects do not compromise system integrity and performance.

2.11.7 A lack of coherent physical input to sensors due to normal effects shall not cause incorrect rejection of sensors to occur (e.g., random inputs to airstream sensors at low airspeed).

2.12 INTERACTION WITH OTHER SYSTEMS

2.12.1 Interaction with other systems may form part of the overall rotorcraft avionics strategy. Any interaction with the ACS shall be fully defined and restricted in respect of any data imported by the ACS.

2.12.2 Data acquired by the ACS must be controlled by the ACS to render its use fully compatible with the ACS integrity requirements.

3 CONTROL LAWS AND SOFTWARE (see also Leaflet 207/3)

3.1 GENERAL REQUIREMENTS

3.1.1 The control laws incorporated in an ACS shall provide the levels of performance stated in the Rotorcraft Specification. In respect of flying qualities and operational performance, when operating in turbulence these shall be at least to the standard of DEF STAN 00-970.

3.1.2 The designer must treat Control Laws and common software as common-mode elements in design and proving.

3.2 CONTROL LAWS

3.2.1 The control laws shall use the minimum number of sensor derived feedbacks, in particular the most rugged and reliable sensors shall be employed for primary feedbacks essential to continued safe flight.

3.2.2 Control laws essential to initial flights of any new configuration shall be robust so that they are insensitive to any aspects of the configuration that cannot be adequately characterised prior to flight.

3.2.3 Gain and phase margins or equivalent measures relating to all modes shall be maintained at acceptable levels for all predictable variations in system operating conditions and rotorcraft configurations throughout the specified life of the system.

3.2.4 Residual oscillations and quantisation effects shall not attain levels which significantly affect the piloting task or pilot subjective acceptance or which can result in significant wear of system and airframe components.

3.2.5 The probability of loss of gain or phase margins which result in an unrecoverable condition shall be compatible with the system integrity target.

3.2.6 Mode compatibility logic shall provide flexibility of ACS operation and ease of mode selection by the pilot. Where changes of control law (mode) call occur in flight, either automatically or by pilot selection, they shall incur minimal disturbance to controlled flight. The need to manually re-trim following a mode change shall be avoided by design as far as possible.

3.2.7 The designer shall ensure that control laws permit the rotorcraft to make safe and controlled entries into and departures from conditions of full and partial constraint (e.g., undercarriage restraint).

3.2.8 Where control laws involve a trimming or integral function which can result in motivators moving progressively towards authority limits during normal operations, such conditions shall be fully identified and appropriate pilot warnings introduced.

3.2.9 The Flight Control Laws shall be defined in an unambiguous Flight Requirements Document (FRD) and Software Requirements Statement (SRS). The document shall include a definition of the Control Law Mode Logic. These documents shall be revised and corrected throughout the ACS development.

3.2.10 The data describing the mathematical model of the rotorcraft and ACS shall be defined in Master Data Sets which shall be revised and corrected with the SRS.

3.3 SOFTWARE

(The overall standards for software in Critical Systems are given in Interim DEF STAN 00-55. The requirements of this chapter are not intended as replacements for, nor do they invalidate, the above DEF STAN. However, they shall be read as enlargements which emphasise the needs of this critical software application)*

3.3.1 The FRD or Software Requirements Statement shall include definitions of system redundancy and failure management algorithms, built in test functions, preflight test, and all other functions to be written in software terms. A method by which the structure and algorithms of Flight Safety Critical Software shall be evaluated to eliminate design errors arising from Control Law Specification ambiguity or lack of definition shall be agreed with the Rotorcraft Project Director.

3.3.2 Every reasonable effort shall be made to ensure that the software is free from errors and is wholly compatible with Safety Critical function.

3.3.3 The Software shall be specified, designed, verified and validated in a rigorous and controlled process which shall be defined to the satisfaction of the Rotorcraft Project Director:

(i) The Software Design Process shall facilitate software assessment and Certification of the ACS.

(ii) All Software associated with the ACS shall be identifiable, separately deliverable, and appropriately documented.

3.3.4 The software shall be structured so that the functions and flow of control are visible, and to permit discrete changes without involving large areas of code. The structure shall provide mechanisms for the partitioning of safety - critical and non-safety - critical software where required.

3.3.5 Independent Software Audits shall be carried out on occasions specified by the Rotorcraft Project Director. Audits should be carried out in accordance with DEF STAN 00-16. Independence may be achieved by the employment of a separate and different team for this purpose from that designing the software.

3.3.6 A listing of the techniques to be employed in the software audit shall be submitted for the approval of the Rotorcraft Project Director.

3.3.7 All software anomalies or related anomalies which arise during software development and testing shall be recorded together with the details of changes arising therefrom.

*See also JAC Paper No 1231

3.3.8 The Flight Resident Software (FRS) or In-Flight Software, shall be tested on the ACS rig (Leaflet 207/6).

3.3.9 These FRS rig tests shall be carried out using the ACS flight standard hardware.

3.3.10 FRS rig tests shall cover all modes of operation and shall explore the full context of conditions to be met in the required flight envelope.

3.3.11 Where software other than the FRS is required to facilitate testing of the ACS hardware, e.g., during environmental testing, such software shall be rigorously tested to show that it is free from anomalies which could reduce the effectiveness of the hardware tests.

3.3.12 Automatic test software shall be shown to be free from anomalies which could hazard the ACS with which the Automatic Test Equipment has been designed to be used.

3.3.13 Each FRS program shall be clearly and uniquely identifiable.

3.3.14 A Software Plan or Control Statement shall be prepared and shall cover all aspects of the software development life cycle.

4 AIRFRAME ASPECTS (see also Leaflet 207/4)

4.1 GENERAL

4.1.1 Interaction will occur between the overall airframe design and ACS design. Consideration shall be given to this interaction so that performance, reliability and integrity requirements are achieved. Airframe in this context includes the rotorcraft structure, rotors and transmission system.

4.1.2 Rotorcraft fitted with ACS shall exhibit the same level of airframe integrity as a rotorcraft designed to fulfill a similar role without an ACS.

4.1.3 In the assessment of airframe integrity, consideration shall be given to all ACS modes, including those degradations and failures from which the rotorcraft can reasonably be expected to recover. The rotorcraft must be capable of withstanding the loads imposed by action taken to recover from failures.

4.1.4 In demonstrating compliance with static strength requirements, the designer shall determine the loading gradients (the variation of structural load with input function) for the major structural components of the airframe. An example of an input function would be the achieved rotorcraft response (e.g., rate of pitch) in the case of a manoeuvre or gust velocity in the case of a gust The loading gradients for a rotorcraft with an ACS shall not be more severe than those for a conventional rotorcraft without an ACS designed for a similar role.

4.1.5 When assessing designs loads, consideration shall be given to the loads due to manoeuvres, gusts and combinations of manoeuvres and gusts. It must be shown that the ACS can function adequately throughout range of manoeuvres, gusts and turbulence that the rotorcraft is likely to encounter. This range must be defined and agreed with the Rotorcraft Project Director. The loading assessment shall consider loading conditions which could exist following the occurrence of reasonable combinations of structural damage and system degradation such as may be caused by minor battle damage.

4.1.6 Where recovery of the rotorcraft following ACS failure is possible, the load spectrum used to establish the Design Limit Load shall be reviewed and limitations defined, if necessary, to establish an acceptable risk for post-failure flight.

4.1.7 Where pilot intervention is required following ACS failures in order to prevent a potentially catastrophic situation from developing, an unambiguous failure warning shall be provided for the pilot. If provided by visual means, this warning shall be within his normal field-of-view.

4.1.8 Use of an ACS on rotorcraft may allow the pilot to fly in a more spirited manner at high speed and low level. As a consequence, the pilot may carry out more medium and high "g" manoeuvres than he would in a more conventional rotorcraft. Special consideration must therefore be given to the definition to fatigue load spectra for the rotorcraft as a whole.

4.1.9 In cases where there would be a significant structural penalty, if the structure was designed to sustain the critical load arising from a malfunction of the active control system, attention shall be given to the possibility of improving the integrity of the ACS.

4.2 DESIGN CASES

4.2.1 The critical design cases for rotorcraft incorporating ACS may be more difficult to determine than those for conventional rotorcraft, consequently many more flight conditions may need to be investigated at the design stage and in flight test.

4.2.2 Critical design cases will be dependent not only on rotorcraft category and role but also on the actual application of ACS in a given rotorcraft design.

4.2.3 Non-linearity effects which arise due to system dwells, motivator rate limits and system authority limits must be accurately or conservatively represented in the design loads analysis.

4.2.4 The principles to be used in the derivation of critical design cases for rotorcraft incorporating ACS are given in Leaflet 207/4.

4.3 LOADS MEASUREMENT

4.3.1 A sufficient number of prototype, development and pre-production rotorcraft shall, as required by the Rotorcraft Project Director, be fitted with comprehensive instrumentation to enable fatigue and static loads to be measured and critical loading actions to be defined. The data obtained must be analysed to assess the validity of design loads and to determine whether any additional critical loading actions could occur if the relative phasing of manoeuvre and gust loads were altered. When critical loading actions are identified which have not been envisaged by the designer, early action (for example: an increase in structural strength, a control system re-design, a structural integrity substantiation by calculation, or the use of a static test airframe) must be taken to ensure that the structure has sufficient strength to sustain the new loads.

4.3.2 In-service rotorcraft incorporating ACS shall, as required by the Rotorcraft Project Director, be fitted with instrumentation to enable defined critical fatigue loads (as defined by actions taken to comply with para 4.3.1) to be measured and assessed, so that consumption of structural life can be quantified. In addition, a representative sample of in-service rotorcraft shall be fitted with a comprehensive load measurement system, which although not necessarily as comprehensive as that fitted to prototype and development vehicles, shall be sufficient to enable defined critical static and fatigue loads to be monitored and to allow any new critical loading actions (caused for example by differences between actual rotorcraft usage and that assumed during design and development flying) to be identified. In cases where such new critical loading actions are identified there must be a reassessment of the structural substantiation for the rotorcraft to determine whether any remedial action is required.

4.3.3 If a re-design or development of in-service ACS software is necessary, either to ensure that the structure has sufficient strength to sustain newly identified critical loading actions or to modify rotorcraft performance characteristics, then flight trials must be repeated to reassess all critical loading actions.

4.4 MODIFICATIONS TO SOFTWARE AND HARDWARE

4.4.1 Before approving any modifications to either the ACS software or hardware the design authority shall ensure that the proposed changes do not alter the characteristics of the flying control system sufficiently to invalidate the structural substantiation of the rotorcraft. In those cases where the structural substantiation would be invalidated by the proposed changes this must be stated on DEF STAN 05-123 Form 714, and the design authority must ensure that the necessary resubstantiation of the structure is completed, preferably before approving the associated modification. Until the completion of the resubstantiation, flying limitations shall be applied which shall ensure that the loads applied to the structure do not exceed 80% of the unfactored limit load.

5 OPERATIONAL AND PILOTING ASPECTS (see also Leaflet 207/5)

5.1 GENERAL CONDITIONS

5.1.1 The ACS shall be designed to function within a specified time, to include gyro run-up, Pre-flight (PF) test etc., following the application of power. This time shall be defined in the Rotorcraft Specification.

5.1.2 The system shall be designed so that the pilot is kept informed, in t timely manner, of any change of status or failure which requires pilot action to maintain performance or to safeguard airworthiness.

5.1.3 The ACS shall be designed so that normal pilot reaction to cues indicating status changes or failures will be instinctively correct.

5.1.4 The pilot shall be given clear indication of any modes which are engaged whether by manual or automatic selection.

5.1.5 All otherwise undisplayed status information shall be made available on demand during flight, wherever possible.

5.1.6 All system performance aspects must be considered in the context of normal service and battle damage deterioration of component and due allowance made for these effects. The method whereby the designer takes account of these deteriorations must be to the satisfaction of the Rotorcraft Project Director.

5.1.7 Indication of normal ACS functioning at those times where no operating limitations are implied shall not be distracting to the pilot.

5.1.8 Where the ACS provides visual warnings of approach to envelope limits, clear indications shall be given within the normal field-of-view of the pilot.

5.1.9 Where the ACS actively prevents limits being exceeded, a clear indication shall be presented to the pilot. Consideration shall be given to the need for the pilot to be able to override the limiting function in emergency conditions and appropriate recording facilities for such events.

5.2 PRE-FLIGHT TEST

5.2.1 The installed ACS shall incorporate an automatic, and where unavoidable pilot-interactive, preflight test function to ensure that all redundant elements, failure detection and signal selection algorithms, etc., are correctly functioning. Where pilot-interactive testing is necessary, clear, unambiguous instructions regarding the action to be taken shall be provided to the pilot.

5.2.2 The level of testing provided in the preflight test shall be sufficient to determine whether the ACS is flight worthy.

5.2.3 Normally it shall be mandatory for the preflight test to be satisfactorily completed before flight. The actual fault status of the system shall determine the ability to proceed to take-off.

5.2.4 The fault status for take-off under normal conditions shall be specified by the rotorcraft designer. Special operational cases where take-off will be required to be made at lower levels of integrity will be defined by the Rotorcraft Project Director.

5.2.5 System status, as detected by the preflight test and interruptive and continuously operating BIT, shall be presented to the pilot during preflight tests.

5.2.6 The ACS shall contain safety provisions to ensure that preflight testing cannot pose any safety hazards. For example, interlocks may be needed to prevent large actuator movements when the rotors are turning.

5.3 IN-FLIGHT TEST

5.3.1 Any automatic In-flight system self-tests incorporated as part of ACS failure detection and management shall not give rise to levels of disturbance which may affect the pilot's ability to perform precision tasks.

5.3.2 Essential elements of an ACS which are only required to operate in discrete parts of the flight envelope (modes) shall be monitored to ensure that they are fit for use when required.

5.3.3 Where an ACS includes a number of subsystems or modes with significantly degraded performance and handling levels, a training facility shall be provided. This shall enable the pilot to select the otherwise naturally occurring degraded modes for training purposes. This provision must not compromise the status of the failure detection and monitoring of functions involved with these modes. Selection of failed sub-systems or modes shall not be possible.

5.4 FAILURE MONITORING

5.4.1 Status data derived from the system shall be made available for fault diagnosis purposes. These data shall be accessible after flight and shall survive removal of main bus and external electrical power from the system.

5.4.2 If pilot action is required in response to a failure, then an appropriate indication shall be automatically generated in the cockpit. Information regarding system status which may affect flight management shall be accessible by the pilot.

5.4.3 Where the design requires failed signals to be automatically rejected by the system, these signals shall be identified and their rejection recorded with time and other appropriate context data for maintenance purposes, in any appropriate recorder.

5.4.4 Fault location to at least line replacement unit (LRU) level shall be automatic and to a confidence level to be agreed with the Rotorcraft Project Director.

5.5 COCKPIT INSTRUMENTS

5.5.1 All displays, indicators, selectors and switches providing essential functions for an ACS, shall be demonstrated to have the necessary levels of integrity.

5.5.2 The layout of switches, indicators etc., shall be designed to minimise the probability of the crew incorrectly operating the ACS in a way which could degrade system operation. Attention shall be given to the correct positioning and sequencing of controls and switches.

6 ROTORCRAFT AND SYSTEM PROVING (see also Leaflet 207/6)

6.1 GENERAL REQUIREMENTS

6.1.1 While the design is being developed, and on completion, every effort shall be made to identify all possible faults and to determine the consequences in order to eradicate all identifiable design errors, in order to meet the design specification. A record of this proving process shall be kept.

6.1.2 All Flight Resident Software (FRS) shall be subject to a full independent audit.

6.1.3 All flight control laws shall, during design and development, be subject to piloted simulator assessment to establish both quantitative measures of performance and qualitative acceptability.

6.1.4 An Electromagnetic Compatibility (EMC) Philosophy Statement and Test Plan shall be written to precede design and development.

6.1.5 A Reliability Demonstration Plan for System and LRUs shall be prepared before testing commences.

6.1.6 A Common-Mode Failure Analysis shall be made to justify the design.

6.1.7 All identifiable hardware faults shall be apportioned probabilities of occurrence which together with the effects upon performance shall be shown to be compatible with the requirements.

6.1.8 A total system safety statement shall be prepared by the rotorcraft designer.

6.2 RIG TESTING

6.2.1 The complete ACS shall be tested in a working rig in which all the flight hardware is installed and provided with suitable power supplies, cooling and motivator loads. The rig shall be capable of accommodating those parts of the Rotorcraft Instrumentation System that may be used in conjunction with the ACS during Development Flight Testing. The inclusion, or representation, of rotorcraft and engine dynamics and other rotorcraft systems that interact with the ACS shall be to the satisfaction of the Rotorcraft Project Director.

6.2.2 Rig testing shall include measurement of performance, both as a fault free system and for all classes of failure.

6.2.3 A schedule of tests to be performed detailing all performance and failure measurement shall be prepared and agreed with the Rotorcraft Project Director prior to the commencement of rig testing.

6.2.4 The rig tests with hardware-in-loop shall include piloted 'flight' testing. These tests shall cover all aspects of flight and ground operations.

6.2.5 A period of fault-free endurance testing of the full system in the ACS rig with pilot in the loop, shall precede first flight. Testing time shall be kept ahead of flight time during initial development. The duration of the test shall be agreed with the Rotorcraft Project Director.

6.3 ROTORCRAFT-ON-GROUND TESTING

6.3.1 The complete ACS shall be correctly installed in the rotorcraft for which it has been designed and tested to prove correct functioning and compatibility with the airframe (structural coupling) and all other systems.

6.3.2 Special environmental compatibility tests shall be performed on the installed system as may be required by the Rotorcraft Project Director.

6.4 FLIGHT TESTING

6.4.1 Flight tests shall be conducted to establish verification of the performance of the ACS.

6.4.2 Early flight tests shall be carried out to verify the aerodynamic data used for the ACS design.

6.4.3 Any special functions required for the purposes of flight test (e.g., fault simulator) shall be designed and tested in compliance with this chapter. Such functions must be identified early in the system design and their provision made as part of the overall ACS Design.

6.5 IN-SERVICE CHANGES

6.5.1 All changes which may affect an ACS shall be scrutinised and implications identified. All changes to an ACS which arise during development and in service use shall be subject to the requirement of this chapter unless specifically excepted by the Rotorcraft Project Director.

7 APPLICATIONS

7.1 INTRODUCTION

7.1.1 Active Control Technology may be applied to rotorcraft in a number of ways and for a variety of purposes. The most important applications relating to the use of Active Control for flight control purposes are:-

(i) Primary Flight Control

(ii) Carefree Handling

7.1.2 Where an ACS is used for any of these applications in such a way that system failures could lead to imminent loss of the rotorcraft, it shall be shown that the system saturation characteristics are compatible with reasonable margins of exceedence of the maximum gust or manoeuvre levels stated in the design specification.

7.1.3 Where necessary for reasons of safety, consideration shall be given to providing the facility for the pilot to override specific ACS modes (e.g., 'g' limiting) whilst remaining within structural limits to be agreed with the Rotorcraft Project Director.

7.2 PRIMARY FLIGHT CONTROL

7.2.1 ACS technology may be applied to primary flight control either in its full, closed-loop, form (i.e. with motivator commands continuously computed from sensor inputs both with and without pilot inceptor inputs) or in a reduced, open-loop, form (without sensor inputs). Fig 1 shows the fundamental features of both the full and reduced forms of application.

7.2.2 The main objectives of using full Active Control are to improve rotorcraft handling qualities and reduce pilot workload.

7.2.3 The main objective of using the limited, open-loop, direct signalling form of ACS shown in Fig 1 is to eliminate the undesirable characteristics of conventional mechanical control links which include high weight, a high level of vulnerability, wear and degradation with time and installation difficulties.

7.2.4 Relevant requirements outlined in paras 1 to 6 apply to both full and limited applications of Active Control unless otherwise agreed with the Rotorcraft Project Director.

7.3 CAREFREE HANDLING

7.3.1 The main objective of applying Carefree Handling is to allow the pilot to exploit fully the capability of the rotorcraft, for long periods if necessary, at a reasonable workload level.

7.3.2 Where Carefree Handling Systems automatically impose limits upon the pilot's control of motivator position or rate of movement, consideration shall be given to the provision of an override facility for use by the pilot in emergency situations. Use of the override facility shall be recorded.

7.3.3 Where automatic limits are applied by a Carefree Handling System, the pilot should be advised, by visual, aural or tactile feedback, when a limit has been reached unless otherwise agreed with the Rotorcraft Project Director.

7.3.4 The Carefree Handling System must exhibit sufficient integrity to ensure that system failures will not hazard rotorcraft safety by either applying inappropriate control demands to failing to limit hazardous demands from the pilot or other sources (e.g., the Primary Flight Control ACS).

7.3.5 All relevant requirements outlined in paras 1 to 6 shall apply to Carefree Handling Systems.

8 DEFINITIONS

8.1 ACTIVE CONTROL SYSTEM (ACS)

An ACS is a system in which commands to the control motivators are continuously computed from sensor input both with and without pilot inceptor inputs. A full-time ACS must operate continuously and without it, safe flight cannot be maintained.

8.2 ACTUATOR

A device, usually powered by electrics or hydraulics which amplifies the command from an ACS computer to move the motivator(s).

8.3 AUTOMATIC TEST EQUIPMENT (ATE)

Any test equipment which uses automatic methods of testing which may be controlled by computer, stored data or hard wired forms of controller.

8.4 AVAILABILITY

Availability is the probability that the item will be available for use.

8.5 BUILT-IN-TEST (BIT)

The facility integrated into a rotorcraft Equipment or LRU to measure and check out its serviceability.

8.6 CONTINUOUS BUILT-IN-TEST (CBIT)

An on-rotorcraft test feature, whereby the correct functioning of the equipment is determined by continuously monitoring the modules within the equipment or by continual tests which do not interfere with the normal operation of the equipment.

8.7 DEFECT

The non-conformance of an item to any one or more of its required parameters within their limits as established in the contractual requirements.

8.8 DEFECT DETECTION PROBABILITY

The ratio of the number of defects detected, by in-built test features, to the total number of defects occurring, expressed as a percentage taken over a statistically significant period of time.

8.9 DEFECT LOCATION PROBABILITY

The ratio of the number of defects located to LRU/module level, by in-built test features, to the total number of defects occurring, expressed as a percentage taken over a statistically significant period of time.

8.10 DEFECT, DORMANT

A defect, the effect of which is not apparent immediately it occurs, and which may remain undetected until a specific function is required, a subsequent defect occurs or a servicing procedure/functional check is carried out which identifies the defect.

8.11 DEFECT, PRIMARY

A defect which is attributable to the items and not caused by user maintenance or personnel factors, defect of related components, or foreign object damage.

8.12 FAILURE, PRIMARY

The termination of the ability of a previously acceptable item to perform its required functions on the rotorcraft, within their limits as established in the Contractual Specification, and which is attributable to the item and not caused by user maintenance or personnel factors, failure of related components or foreign object damage.

8.13 FLIGHT RESIDENT SOFTWARE (FRS) (In-Flight Software)

FRS forms the software program to be implemented within the ACS and which is to fly.

8.14 FULL-TIME

As in Full Time System, a system which must continue to operate at all times in order to ensure continuation of safe flight.

8.15 HAZARDOUS SITUATION

A situation where conditions arise which threaten the safety of the crew and/or the rotorcraft.

8.16 INCEPTORS

The means by which the pilot's primary flight control demands are input to the system, e.g., control column, rudder pedals, etc.

8.17 INTEGRITY

The probability that the system will provide a specified level of safety.

8.18 INTERRUPTIVE BUILT-IN TEST (BIT)

An on-rotorcraft test sequence, initiated by a stimulus which will interfere with the normal operation of the system.

8.19 LRU BUILT-IN TEST (LBIT)

The facility integrated into an LRU to measure and check out its serviceability to module level.

8.20 LINE REPLACEMENT UNIT (LRU)

Any readily accessible rotorcraft unit normally consisting of sub-assemblies or modules, designed for ease of replacement and capable of being handled and changed, preferably by one man, within the time laid down and as identified by the equipment specification.

8.21 MAINTAINABILITY

The economy in time, manpower, equipment and necessary materials with which potential or actual failures can be detected, diagnosed, prevented or corrected, and with which routine handling, replenishment and servicing operations can be carried out.

It may be measured as the ability of an item under stated condition of use to be retained in or restored to a specified condition, when maintenance is performed by personnel having specified skill levels under stated conditions and using prescribed procedures and resources.

8.22 MODE

A discrete and selectable control law (function). A mode may be automatically and/or manually selected and deselected within an ACS. (cf Auto-pilot modes).

8.23 MOTIVATORS

The devices which produce forces or moments which affect the rotorcraft motion, e.g., rotor blades.

8.24 NUISANCE DISCONNECT

An undesirable condition not due to component defect, which is identified by a monitoring system as if it were a defect. The condition may or may not persist.

8.25 RELIABILITY

The probability that the system will achieve a specified level of performance.

8.26 SENSORS

Detecting devices, which transduce motion etc., into signals suitable for transmission in the ACS e.g., gyros, accelerometers, windvanes, displacement pickoffs etc.

8.27 SYSTEM FAILURE

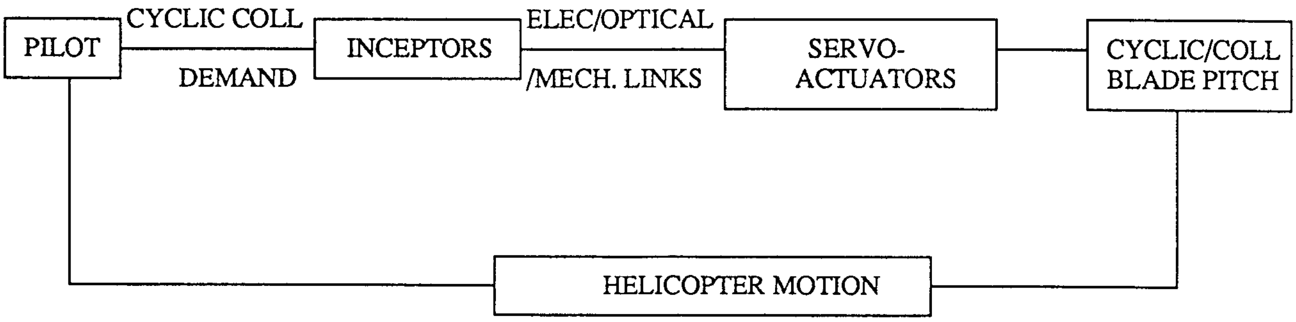
An occurrence in which essential system function is lost and in the context of full-time ACS may lead to loss of the rotorcraft or termination of mission.

8.28 TESTABILITY

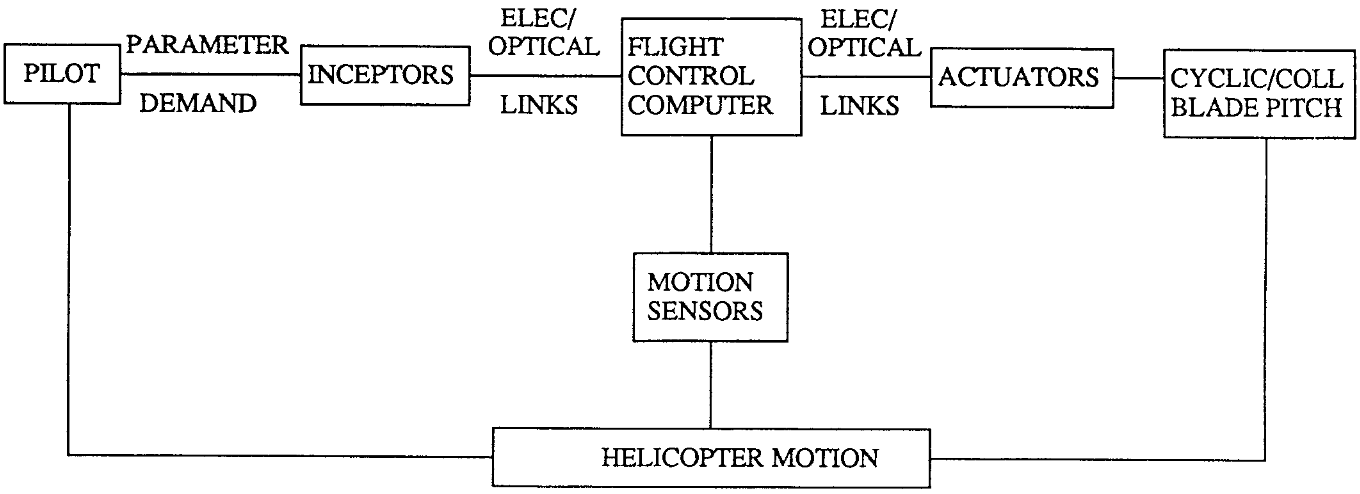
Testability is an element of both producibility and maintainability. Therefore testability refers to the ability of both equipment manufacturer and the maintainer to establish the correct performance characteristics of the system/equipment.

8.29 TRANSDUCERS

A subset of sensors which are mostly linear and rotary displacement sensors applied to actuators, inceptors and motivators.



Open-Loop ACS



Closed-Loop ACS

FIG 1. FUNDAMENTAL FEATURES OF FULL AND REDUCED FORMS OF ACS

LEAFLET 207/1

ACTIVE CONTROL SYSTEMS

GENERAL REQUIREMENTS AND INTEGRATION ASPECTS

1 INTRODUCTION

1.1 The primary purpose of using an ACS on a rotorcraft is to improve flying qualities and thus reduce pilot workload and improve mission performance. Additional benefits in the areas of, for example, reduced weight, reduced vulnerability and improved cockpit design may also be achievable. These latter benefits may also result from a reduced, open-loop application of the technology (see Chapter 207, para 7.2).

1.2 Both closed and open-loop systems will require full control authority. In all cases, the ACS must operate with a level of integrity at least as high as that inherent in previous generations of primary control systems.

1.3 An ACS offers opportunities for the integrated design of control system, airframe and avionics systems so as to optimise performance and efficiency.

2 AIRFRAME INTEGRATION

2.1 Use of an ACS will have a significant impact upon airframe design. In designing a new airframe with ACS, an interactive design approach should be adopted to ensure that airframe and ACS designs are compatible and that full advantage is taken of the capabilities offered by ACS.

2.2 Where an ACS is to be retrofitted to an existing airframe, great care should be taken to ensure that airframe/ACS designs are compatible. A review of airframe design, taking into account absolute loads and loads spectra likely to be generated by the use of the ACS in service should be carried out. The airframe aspects outlined in Chapter 207 para 4 should be given special consideration.

2.3 An integrated airframe/ACS approach offers opportunities for:-

(i) Harmonising airframe and control system limits through the adoption of "Carefree Handling" features in the ACS (see Chapter 207 para 7.3).

(ii) Introducing additional motivators (e.g., moving tailplane, wing with ailerons or additional thrust generators) without imposing unacceptably high workload upon the pilot.

(iii) Optimising airframe layout, particularly in the cockpit area, by careful design and positioning of ACS components.

3 AVIONICS SYSTEM INTEGRATION (see also Chapter 207, para 2.12)

3.1 Integration of the ACS with the rotorcraft avionics system offers potential for optimising the sensor fit, improving performance and increasing the range of facilities available. Typically, a rotorcraft ACS might be integrated in some way with the

Navigation, Health and Usage Monitoring, Flight Management and Cockpit Display sub-systems.

3.2 Integration with other sub-systems must not compromise the integrity of the ACS. Care must be taken, therefore, to ensure that malfunctions or failures in other sub-systems cannot cause the ACS to malfunction in such a way as to hazard rotorcraft safety. The detailed failure modes and effects analysis (FMEA) for the ACS must include consideration of malfunctions in other sub-systems that interact with it.

LEAFLET 207/2

ACTIVE CONTROL SYSTEMS

SYSTEM REQUIREMENTS

1 INTRODUCTION

1.1 Two of the most dominant problems in the development of an ACS are the selection of the means to achieve a very low probability of catastrophic failure and to demonstrate that this has been achieved.

1.2 Designers may be asked to aim for a risk of loss of the rotorcraft due to ACS failure in the order of 10^{-7} per flight hour. For this reason, systems employing redundancy are necessary.

1.3 Single lanes using electronic control may exhibit failure rates exceeding 1 in 1000 flying hours, it follows that a simple cross-compared redundant system of this quality may require to comprise four independent lanes of control.

1.4 Fewer lanes may be utilised if adequate methods of self-monitoring can be satisfactorily demonstrated.

1.5 Use of redundancy introduces the prospect of having to transfer data between otherwise isolated lanes.

1.6 Where such intercommunication is utilised great care should be taken to avoid all sources of common-mode failure.

1.7 Use of similar redundancy is attractive to the designer because of the ability to thereby secure good identity of data. It does, however, present another source of common-mode failure.

1.8 The natural distribution of random defects or failures amongst similar hardware components results in a low probability of simultaneous failure, but there nevertheless remains the possibility of an unidentified common susceptibility to undefined influences arising from the inherent commonality of design and manufacture. For this reason great care should be exercised in the selection of components for use in redundant systems.

2 SENSOR CONFIGURATIONS

2.1 Conventionally, in redundant systems, there are 3 choices of primary motion feedback sensor configurations, which may be summarised as follows:

(i) Orthogonal, single axis packs.

Measuring axis close to rotorcraft body axis and packaged in packs of redundant similar sensors.

(ii) Orthogonal, multi-axis sets.

Measuring axes close to body axes but with a group of, say, 3 axis sensors in each set, one set per lane.

(iii) Skewed axis, redundant sets.

This configuration permits a reduction in the total number of sensors needed to meet the integrity and reliability requirements.

2.2 The disadvantages of the skewed axis system lie in the extra computing required to resolve the skewed sensor signals into the required axis and that necessary to handle the failure management, together with the architectural complications associated with this computing and its power supplies.

2.3 A trade-off exists between the minimum number of sensors and the magnitude of failure transients and nuisance warnings.

2.4 There are inevitable compromises which arise from the co-located sensors and the fact that there may need to be at least 2 redundant sets located apart.

2.5 Each sensor and each set may sense variable mixes of structural mode responses, the eradication or resolving of which demands special attention.

2.6 The optimisation of management of a skewed axis sensor configuration can lead to this being implemented as a discrete sub-system with a different level of redundancy to that of the ACS proper. In this case the choice of method by which essential sensor data is transferred to the control computers needs special care, in order to ensure that the required levels of integrity are met

3 CONTROL AUTHORITIES

3.1 Two levels of control authority may exist in an ACS:

(i) Motivator (or actuator) authority, which normally represents the maximum usable geometric output.

(ii) Mode authority, or internal signal authority. This is a flexible parameter which may be fixed or variable, and which is usually established with an integrity of similar order to that of the system itself.

Any mode may be allocated a particular authority.

3.2 Finite motivator authorities have a profound effect upon the behaviour of a rotorcraft which relies upon an ACS to provide essential performance. The choice of authority for a given function has to be made in the light of both specified performance and malfunction requirements and the nature of circumstances expected to be met at and beyond control saturation.

3.3 Other levels of authority exist in an ACS in association with mechanical implementation and are concerned with the provision of a finite tolerance between electrical transducer operating authority and mechanical, absolute, authority. These authorities are not usually relevant to overall control performance.

LEAFLET 207/3

ACTIVE CONTROL SYSTEMS

CONTROL LAWS AND SOFTWARE

1 MATHEMATICAL MODEL

1.1 In order that the overall performance objectives may be addressed, it is necessary to acquire a thorough understanding of all relevant parts of the rotorcraft and system dynamics. From this information, a mathematical model of the total rotorcraft and system should be built up. The efficient design of control laws will depend upon the accuracy of the model.

1.2 The model should be as complete as possible. For full closed-loop ACS control law design the model should include representation of all dynamics up to a frequency of at least twice that of the bandwidth of the control system, and representation of aerodynamics including non-linear effects and interactions. The following elements should be modelled:

(i) Main rotor.

(ii) Tail rotor.

(iii) Auxiliary surfaces (e.g., tailplane).

(iv) Fuselage.

(v) Engines.

(vi) Transmission.

(vii) Control system hardware (e.g., sensors, actuators).

(viii) Undercarriage.

Some simplification of the model may be permissible for some elements of the control law design process, e.g., for the use of frequency domain methods or for real-time simulation.

1.3 The accuracy of the model and any simplified versions should be validated as soon as appropriate flight test data can be collected. The rotorcraft flight test programme should take into account the need for this validation work. Some additional system features may be necessary to facilitate validation. These may include provisions for the injection of defined control inputs (e.g., pulses, sine waves etc.). The design of these provisions should take due account of the requirements of Chapter 207.

2 COMMONALITY OF SOFTWARE

2.1 If the designer chooses to utilise one common software interpretation of control laws for use in all lanes of a redundant system, then he should recognise the implications of this in respect of common-mode failure susceptibility.

2.2 This element of commonality is difficult to assess in terms of reliability, and attention is drawn to various methods of establishing confidence in the design of such software.

2.3 Confidence is currently established by ensuring that all software, paths and functional elements are adequately exercised during rig testing.

2.4 There is however, to date, no practical method for establishing a quantitative measure of software confidence.

LEAFLET 207/4

ACTIVE CONTROL SYSTEMS

STRUCTURAL IMPLICATIONS OF ACS

1 INTRODUCTION

1.1 This leaflet provides guidance on acceptable means of compliance with the structural requirements of Chapter 207 and related requirements such as those for static strength (Chapter 200), fatigue and damage tolerance (Chapter 201) and aero-elasticity (Chapter 500).

1.2 The procedures used for static and fatigue design of rotorcraft incorporating ACS are similar to those for conventional rotorcraft. However, critical design cases may be more difficult to determine, and the design process will necessitate integration of the procedures used for structural, aerodynamic and active control system design.

2 GENERAL DESIGN CONSIDERATIONS

2.1 From a structural point of view Active Control Systems (ACS) may be grouped into two categories:

(i) Those whose prime purpose is to enhance performance characteristics such as agility and weapon platform stability and,

(ii) Those whose prime purpose is to alleviate loading actions so as to permit reductions in structural weight, size or stiffness.

2.2 ACS in the first category might include, for example, carefree manoeuvring (to prevent departure from controlled flight), relaxed stability, manoeuvre enhancement and vibration reduction.

2.3 ACS in the second category might include, for example, gust load alleviation, manoeuvre 'g' limiting and control of rotor stability.

2.4 For systems in the first category and, for those systems in the second category which as a secondary effect enhance performance, loading actions are usually of a different form from those for conventional rotorcraft.

2.5 For ACS rotorcraft it is likely that design requirements alone are insufficient to define all critical static loading cases. Therefore the intended rotorcraft usage should be probed for potentially critical or hazardous loading actions due to pilot input or particular gust patterns, separately or in combination. The effects of augmentation system disconnects, degrades and failures should be evaluated.

2.6 In particular, attention should be given to the flight and ground loads which arise when the active control system is:

(i) Fully serviceable.

(ii) Transient between the serviceable and degraded state.

(iii) Degraded either actively (producing undemanded and/or excessive control movements) or passively (producing either no control movement or control movements in the correct sense but of inadequate authority).

2.7 When identifying those loads which are likely to occur during flight in the degraded state the designer should consider whether the pilot could, if he was aware of the degradation, reduce the severity of the loads which are likely to be encountered by taking appropriate measures such as:

(i) Limitation of maximum speed.

(ii) Limitation of maximum 'g'.

(iii) Reduction of adverse stresses by jettison of stores or fuel.

If, to cope with a particular degradation of the ACS, the designer chooses to maintain the structural integrity of the rotorcraft by requiring the pilot to observe a flying limitation, then the margin between the maximum load predicted to occur following the implementation of the limitation and the limit load capability of the degraded rotorcraft should comply with the guidance given in Leaflet 900/1.

2.8 A rotorcraft structure possesses a safety margin in terms of load bearing capability because a structure is required to withstand Design Ultimate Load (DUL), which is Design Limit Load (DLL) multiplied by a factor of safety of 1.5, without collapse. In the case of the ACS rotorcraft control system, saturation and other non-linearities may mean that operating margins, such as the increment in input function required to generate an increase in load from say DLL to DUL, may be smaller than for a similar conventional rotorcraft of which there is knowledge. Thus for an ACS rotorcraft the designer should examine the variation of structural loading with input function in respect of multi-axis inputs and gusts, separately and in combination, and should establish that the loading gradients are not so severe that a small increment in input function could produce such a large increment in load that there should be a high probability, if Service Release conditions were slightly exceeded, of DUL being approached and exceeded. In general terms this means that the margins in terms of the increment in input function required to increase load from Service Release conditions to DUL should be no worse than for a rotorcraft designed to fulfil a similar role without an ACS.

2.9 ACS rotorcraft which feature either carefree manoeuvring or manoeuvre 'g' limiting may fly to the extreme of their flight envelope more frequently than conventional rotorcraft. Providing the ACS is sufficiently reliable and DLL is adequately defined, it is unlikely that an ACS rotorcraft will exceed DLL more frequently than an equivalent conventional rotorcraft in which the pilot must observe flight limitations to prevent exceedances of DLL. Nevertheless, the use of a 'g' limiting system does not guarantee that critical loads will not be exceeded as it may only limit one of the components of load in a particular structural item. Consequently, to enable the Service Release levels in respect of static loads to be assessed, the proposed rotorcraft usage should be probed to identify possible exceedance of DLL and the ease with which DLL can be approached.

2.10 Rotorcraft incorporating ACS are likely to exhibit increased control activity, in particular more high frequency small amplitude motions, and the designer should pay special attention to the fatigue design of the actuators, the motivators, the mechanical interface between the motivators and the actuators, and the associated support structure.

2.11 The ACS should be designed to avoid inertia instability of the control system occurring when the rotorcraft is undergoing system testing on the ground.

2.12 There can be no guarantee that the loads analysis can identify all critical combinations of possible loading actions. Consequently, Chapter 207 para 4.2 contains requirements for the in-service confirmation of the load levels deduced from the analysis.

2.13 Where an ACS is used to contain structural loads within limits, the system capabilities and performance derived from calculation must be substantiated by direct strain measurement during development flying.

2.14 Military rotorcraft utilise ACS to allow the pilot to fly his rotorcraft in a more spirited manner at high speed and low level. Consequently it should be anticipated that the pilot will carry out more medium and high 'g' manoeuvres than he would in a conventional rotorcraft. Therefore, special consideration should be given to the definition of fatigue load spectra for the rotorcraft as a whole.

LEAFLET 207/5

ACTIVE CONTROL SYSTEMS

OPERATIONAL AND PILOTING ASPECTS

1 INTRODUCTION

1.1 It is likely that agile military rotorcraft equipped with ACS may be subject to increased manoeuvre frequency and magnitude in service, compared with conventionally controlled rotorcraft. It is recognised that if high levels of manoeuvring are frequent then reduced airframe life will ensue, it not allowed for in the design.

1.2 Two possible options which may be considered are:

(i) Design for maximum performance and allow for airframe life consumption by investing in a more resistant airframe (at possible weight cost).

(ii) Design for two levels of performance; maximum performance which could be termed 'Wartime' Performance and a lower level, perhaps termed 'Peace-time' Performance, either of which could be selected for use by the pilot. Carefully regulated service use of these alternatives could ensure a reasonable rate of consumption of airframe life.

The latter option has system design and training implications in that a lower level performance has to be defined and appropriate mode selection provided, with possibly instinctive override available to the higher level of performance.

1.3 NUISANCE WARNINGS

1.3.1 Redundant Systems with voting incur a finite probability of nuisance failure warnings, although with careful design these may be minimised. When apparently 'nuisance' failures occur the conditions leading to the event can, if known, enable confirmation and/or rectification to be carried out to avoid repetition and investigatory loss of rotorcraft availability. It is therefore desirable for context data to be recorded for each 'failure' incident.

1.3.2 Experience has shown that incidents of this nature prove difficult to resolve if no specific information, other than aircrew reports, is available.

1.3.3 Useful context data would include:

- Failed item
- Relative errors (redundant items)
- Control system status
- Dynamic data (3 axes)
- Flight case (speed, height)

LEAFLET 207/6

ACTIVE CONTROL SYSTEMS

AIRFRAME AND SYSTEM PROVING

1 INTRODUCTION

1.1 This leaflet is concerned with the preflight testing of ACS to gain confidence in the system so that flight testing may take place.

1.2 ACS testing should be organised so that the simplest possible operational standard of ACS that will permit safe flight can be tested first. Augmentation facilities should be added only when this simple standard has been satisfactorily proven.

1.3 The testing to be carried out should be defined in a formal Acceptance Test Procedure (ATP) document and approved by the Rotorcraft Design Authority. A formal test report should be produced on completion of testing detailing results of all tests defined in the ATP.

2 RIG TESTING

2.1 In order to develop the ACS to flight standard, a rig is required in which the complete system hardware can be fully functioned. A comprehensive real-time flying simulation which permits all operational performance to be assessed should form part of the rig.

2.2 The rig simulation should also be utilised to enable test pilot assessments to be carried out with particular reference to system management including the effects of failures. These tests should also allow impromptu pilot investigations which explore off-design cases and any other points of curiosity.

2.3 This facility should provide the basis for system performance proving prior to first flight. A detailed schedule of failure cases should be assessed.

2.4 This testing should precede installed system testing in the rotorcraft and enable installed testing to be restricted to essential items.

2.5 Endurance testing on this rig may be restricted to whole system operations with a pilot in the loop or may be in part automated. Sequences of start-up, pre take-off checks, taxy, take-off, climb, through representative flight conditions and manoeuvres covering all operational roles, etc., landing, post-flight checks and shut-down should be included.

2.6 These tests are all to be completed to an agreed equivalent number of flying hours before first flight.

2.7 The required number of hours before first flight are to be conducted with full flight standard ACS equipment and are expected to exhibit defect free operation. As flight standard equipment should have undergone burn-in testing it is not unreasonable to expect 100 hours defect-free operation.

2.8 Testing should include subjecting the ACS to incorrect operating procedure sequences that could occur in service due to operator inexperience e.g., incorrect mode engagement logic.

2.9 When the testing has been completed, the equipment should be removed from the rig and placed in a bonded store until required for installation on the rotorcraft.

LEAFLET 207/7

ACTIVE CONTROL SYSTEMS

PART-TIME SYSTEMS

1 INTRODUCTION

1.1 The provisions of Chapter 207 are specifically for those control systems where correct operation is essential for continued safe flight. These are termed full-time systems. A full-time system may include those systems where there is automatic reversion to an alternative active mode in appropriate circumstances. When an active control system is employed in such a way that continued safe flight is possible after incorrect system operation by reversion to some form of direct control by the pilot it may be referred to as a "part-time" active control system.

1.2 Examples of ACS may be defined as part-time are:-

(i) Where there is provision for reversion to direct control, the reversionary mode not being part of the full-time active control system.

(ii) Where, in a dual control rotorcraft, the ACS is operated from one crew station only and a "safety pilot" at the other crew station has a means of controlling the rotorcraft which is not affected by failures of the ACS.

(iii) Where a rotorcraft employs an ACS mode to improve performance but where failure of that mode is not flight critical e.g., some carefree handling modes where failure may result only in reduced fatigue life.

2 SAFE DESIGN PHILOSOPHY FOR PART-TIME ACS

2.1 The ACS should not introduce any situation which is likely to jeopardise the safety of the rotorcraft whilst operating (either correctly or incorrectly) or when inoperative.

2.2 In the event of an incorrect operation of the ACS being detected, there should be an automatic reversion to the back-up mode and there should be an indication to the crew that reversion has taken place.

3 APPLICABILITY

3.1 In the case of a part-time ACS the application of the requirements of Chapter 207 may be limited to those components of the system which affect overall safety of the rotorcraft e.g., facilities that detect failures and initiate automatic reversion and common-mode elements (e.g., control laws and software) where failures could hazard safe flight whilst the ACS is operative.

3.2 The extent to which the provisions of Chapter 207 apply will be related to the rotorcraft flight envelope and will be defined in conjunction with the Rotorcraft Project Director. Due consideration should be given to all of the requirements of Chapter 207, however, and the requirements should only be modified where it can be clearly shown that no safety hazard will result from relaxation of the requirements.

3.3 Particular attention should be given to the transient affects that occur due to reversion. It is important that consideration is given to the effect of reversion on airframe loads. Reversion facilities should be designed so that normal pilot reaction to cues or failures will be instantly correct.

3.4 Where a part-time ACS may be engaged after take-off, due consideration shall be given to providing appropriate pre-engagement test-procedures, both on the ground and in the air, that ensure that the ACS will operate correctly when engaged.

CHAPTER 208

GUST LOADS

1 GENERAL

1.1 The requirements of this chapter are applicable to all rotorcraft and to the strength of the complete structure, when the Rotorcraft encounters gusts normal to the flight path both in the plane of symmetry (vertical gusts) and perpendicular to the plane of symmetry (lateral gusts).

1.2 The Rotorcraft as a whole shall have adequate strength for any gusts encountered from directions intermediate between the vertical and horizontal plane which provide critical loading conditions.

2 FACTORS

2.1 The structure shall have proof and ultimate factors not less than 1.125 and 1.5 respectively on the loads arising in each case specified below.

3 DESIGN CONDITIONS

3.1 The Rotorcraft is assumed to be in unaccelerated flight when gusts are encountered except for the case specified in paragraph 3.4. The Rotorcraft speed, engine operating conditions and the associated maximum velocities of gusts are specified in para 4. All altitudes up to the design maximum altitude shall be considered.

3.2 The required strength shall be achieved at all masses between the maximum mass at which the Rotorcraft can reach the altitude considered and the minimum flying mass (that is the take off mass less stores ammunition and other items readily dropped or expended except for sufficient fuel for a normal descent and cruise at sea level at the engine conditions appropriate to maximum endurance for a time to be agreed with the Rotorcraft Project Director).

3.3 The trim of the Rotorcraft shall be assumed to be unchanged by the pilot during the passage of the Rotorcraft through the gust, unless it is considered that flight control system inputs will cause an additional adverse effect on the loading, or an active alleviating system having a sufficiently low probability of failure is fitted to the Rotorcraft.

3.4 Unless otherwise agreed with the Rotorcraft Project Director the Rotorcraft shall be designed for the case when a vertical gust of 70% of that specified in paragraph 4.1 is encountered at the same time as it is performing a manoeuvre of 60% of the maximum limit load factor.

4 OVERALL GUST CASES

4.1 The Rotorcraft shall be assumed to be subjected to a gust velocity of 10.7 m/s (EAS), unless otherwise agreed with the Rotorcraft Project Director, at any speed within the flight envelope whilst flying under visual flight conditions.

4.2 Any necessary increase of the gust velocity relative to that stated in paragraph 4.1 to cover flying under instrument flight conditions shall be agreed with the Rotorcraft Project Director.

4.3 In applying the conditions of paragraphs 4.1 and 4.2 to the design of the Rotorcraft it shall be assumed that the gust is sharp edged and that loads induced on the Rotorcraft by the gust are balanced by inertia forces on the whole Rotorcraft.

5 ROTORCRAFT BLADE GUST CASE

5.1 In addition to the conditions specified in paragraph 4, each individual rotor blade shall be designed to withstand the loads which arise when a vertical gust of 15.3 m/s (EAS) is encountered.

5.2 In applying the condition of para 5.1 it shall be assumed that the gust intensity increases from zero to the design value over a specified gradient distance in accordance with the relationship:

$$U = \pm \frac{U_{de}}{2}\left(1 - \cos \frac{p\ s}{k}\right)$$

where U_{de} is the design gust velocity stated in para 5.1

s is the distance the Rotorcraft has penetrated into the gust

k is the gradient distance - 30.5m (100 ft)

5.3 The worst combination of rotor speed and forward speed shall be assumed with this case.

CHAPTER 209

RADOMES

1 INTRODUCTION

1.1 The general requirements for radomes are given in Chapter 707 'Radio and Radar Installations'. These include, in para 5.1, specific requirements relating to aerodynamic and structural considerations, including, resistance to erosion, impact damage and lightning strike.

1.2 It is accepted that good radome design will involve a trade-off between electrical and structural properties and the purpose here is to draw attention to the procedures that shall be followed in supporting any deviation from the structural standards that normally apply.

2 REQUIREMENTS

2.1 The structural requirements include those for static strength and stiffness, fatigue performance, impact from hail and resistance to bird impact and lightning strike.

2.2 In each design case where it is found to be impractical to achieve these standards, the reasons for compromise and the basis of the trade-off between structural and electrical properties shall be referred by the Design Authority, to the Rotorcraft Project Director for a concession before proceeding with the design. This supporting evidence shall include a failure analysis, containing descriptions of failure modes and consequential effects, including those on flight aerodynamics and engine(s).

2.3 For those design cases which result in a likelihood of structural failure of the radome, it shall be shown that the integrity of the rotorcraft is not impaired to an extent unacceptable to the Rotorcraft Project Director.

2.4 It is recognised that a best compromise between structural and electrical properties may sometimes be achieved by designing for replacement rather than repair. In such instances the economic and maintenance implications shall be drawn to the attention of the Rotorcraft Project Director; normally for 'disposable-type' radomes the mean-time-between failures must not be less than 500 flying hours.

LEAFLET 209/0

RADOMES

REFERENCE PAGE

MIL-A-87221 (USAF), Section 3.2.22.3	Aircraft Structures General Specification for
MIL-R-7705	General Specification for Radomes

PART 2 APPENDIX No. 1
STRUCTURAL STRENGTH AND DESIGN FOR FLIGHT
MILITARY DERIVATIVES OF CIVIL ROTORCRAFT

1 INTRODUCTION

1.1 This Appendix covers general and operational requirements for military derivatives of civil rotorcraft.

1.2 Each paragraph of the Appendix consists of a table and amplifying notes comparing military and civil requirements for individual chapters from Part 1 of DEF STAN 00-970 Volume 2. In general, only those parts of the DEF STAN in which military and civil requirements differ are included in the Appendix and where no reference is made to DEF STAN chapter/paragraph. numbers it is considered that the military and civil requirements are compatible and adequately covered by the military requirements (unless explained otherwise in the amplifying notes).

1.3 The issue or change number of the civil airworthiness requirements referred to in this Appendix are recorded in paragraph 9.5 of the main introduction to DEF STAN 00-970 Volume 2.

2 COMPLIANCE CHECKS/ASSESSMENT

2.1 Throughout this Appendix reference is made to the need for ‘Compliance Checks or Assessments’ of the existing civil rotorcraft against the military requirements of DEF STAN 00-970. A full explanation of the aims of compliance checks or assessments is given in paragraph 9 of the main Introduction to DEF STAN 00-970 Volume 2.

3 STATIC STRENGTH AND DEFORMATION

3.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 200	BCAR’S		FAR’S		JAR’S	
Para Item	Section G	29	27	29	27	29
2. Design Cases	3-1,4	301 309	301 309	301 309	301 309	301 309
3. The Ultimate Strength and Proof Requirements	3-1,4	303	303	303	303	303
4. Substantiation of the Static Allowable Stress for Grade A Details	--	--	--	--	--	--

DEF STAN 00-970 CHAPTER 200 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
5. Demonstration of Compliance With the Ultimate Strength and Proof Require-ments for Complete Structures or Components	3-1,2 3-1,4 and App 1	305 307	305 307	305 307	305 307	305 307
6. Measurement of Loads on Rotorcraft Structures	--	307	307	307	307	307
8. Reduction of Vulnerability to Battle Damage	--	--	--	-	--	--
Table 1: Factors by which The Mean Strength of Details or Elements must be reduced to obtain A 'B' Allowable Value:	--	--	--	--	--	--

3.2 REQUIREMENTS

3.2.1 DEF STAN 00-970 Chapter 200 paragraph 3.1 states that generally the Ultimate Factor shall be 1.5 and the Proof Factor shall be at least 1.125. The corresponding civil requirements in BCAR's Section G and 29 specifies an Ultimate Factor of 1.5 and a Proof Factor of 1.0. FAR's and JAR do not quote a Proof Factor, see also 3.2.2. below.

3.2.2 DEF STAN 00-970 Chapter 200 paragraph 3.2 states that until the design proof load is reached, no grade A items shall sustain deformation detrimental to safety. The civil requirements however state that up to proof (limit load x 1.0 factor) load where quoted or limit load, that deformation may not interfere with safe operation. In assessing the design of the civil aeroplane for compliance with DEF STAN 00-970 the Proof Load conditions will need to be assessed in relation to the 1.125 proof load factor.

3.3 COMPLIANCE

3.3.1 Whilst both the civil and military requirements specify an Ultimate Factor of 1.5, it is important to consider whether the proposed military usage will increase the frequency of occurrence of higher loads so that the chance of reaching or exceeding limit load is increased. In effect, such usage reduces the margin of strength implicit in the factor of 1.5 and for this reason there is less room for uncertainty in the compliance procedure. Due emphasis must also be placed on establishing the real proof strength or 'structural airworthiness limit'. Therefore, if the military derivative is to be operated very differently to its civil counterpart, or at a higher AUW, or is a new design incorporating novel features, then a detailed compliance check of the proof strength of the structure will be required to assess compliance with Chapter 200.

3.3.2 The need for a compliance check of the static strength of a military derivative of a civil rotorcraft should be discussed with the Rotorcraft Project Director.

4 FATIGUE

4.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 201	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. Safe Lives	3-1,5 and App 2	571 572	571	571	571	571
3. Demonstration of Compliance	3-1,2 3-1,5 and App 2 & 3 3-4,3	571 & ACB 572 & ACB	571	571	571 & ACJ of JAR 25	571 & ACJ

4.2 DEF STAN 00-970 Chapter 201 does not refer directly to the pressure cabin but the interaction of this if applicable with the airframe structure and vice versa, must be taken into account in the fatigue damage tolerance analysis and testing.

4.3 REQUIREMENTS

4.3.1 In general, the military and civil requirements for fatigue strength are compatible . However, it is important to recognise that DEF STAN 00-970 is primarily concerned with the compact structures whereas BCAR's, FAR's and JAR's address the large open type of structure, which is usually much easier to inspect. Military requirements specify that the residual strength of an inspection-dependent structure shall not fall below 80% of UL. Civil requirements, however specify that a fail safe structure should be expected to withstand only limit load once it has failed.

4.4 Because of the vital importance of establishing the correct fatigue life of a military derivative it will be necessary:

4.4.1 For newly built rotorcraft to effect a comparison between the rotorcraft as designed to civil requirements and the military fatigue requirements and usage and to establish safe lives, inspection periods and recording techniques appropriate to the military role(s).

4.4.2 For rotorcraft which have been used in civil operations, to carry out a procedure, similar to that in 4.1.1 to establish the equivalent military fatigue life consumed and remaining, with recording techniques for monitoring the latter.

5 MANOEUVRES

5.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 202	BCAR'S		FAR'S		JAR'S	
	Section					
Para Item	G	29	27	29	27	29
2. Factors	--	--	--	--	--	--
3. Manoeuvres to be considered	--	--	--	--	--	--
4. Flight Loads	3-2,1 3-2,3	321	321	321	321	321
5. Limit Manoeuvring Load Factor	3-2,1	337	337	337	337	337
6. Yawing Flight	3-2,3	351	351 427	351 427	351 427	351 427
7. Sideways Flight Conditions	3-2,3	352	427	427	427	427
8. Auxiliary Rotors and Control/ Stabiliser Surfaces	3-4	3-9 395	391 395	391 395	391 395	391 395

DEF STAN 00-970 CHAPTER 202 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
9. Power Plant and Transmission System	3-4	361	361	361	361	361
10. Supplementary Conditions and Assumptions		339 361 362	339 361	339 361	339 361	339 361

5.2 The military and civil requirements are very similar but manoeuvre responses may require to be much faster for the military derivative in a combat role if it is so stated in the specification. An assessment may be necessary to determine if response times are acceptable.

5.3 It should be noted that, even when the values of normal accelerations and speeds coincide, the effects of the different proof and ultimate factors (paragraph 3.2 of Appendix 1 Chapter 200) between civil and military rotorcraft will have to be considered.

6 CONTROL SYSTEMS - MECHANICAL COMPONENTS

6.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 203 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
2. Strength	3-6 4-8	391 395 397 398 399 681	391 395 397 399 681 (FAR'S 27, 29 and JAR'S 27, 29)			
3. Control Circuits-Overall Design	3-6 4-8	671 to 695 696	671 to 695 (FAR'S 27, 29 and JAR'S 27, 29)			

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<tr><th>DEF STAN 00-970 CHAPTER 203

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6.2 For comments on proof and ultimate factors see Appendix 1 Chapter 200 paragraph 3.2 but note that different factors are required for particular design cases.

7 STRENGTH CONSIDERATIONS FOR AUTOMATIC CONTROL SYSTEMS

7.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 204	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Introduction	3-6	671 672 673	671 672 673			
2. Basic Requirements	3-6 6-4	672 695	672 695			
3. Correct Functioning	6-4 and App	683 696	683			
4. Incorrect Functioning	6-4 and App	695	695			
5. Integrity	6-4 and App	391 to 399 671 673 695	391 to 399 671 673 695			
6. Testing - Ground & Flight	4-8,2 6-4 App.6	307 681 683	307 681 683			

7.2 All the requirements contained in this Chapter are referred to elsewhere in other chapters.

7.3 This Chapter highlights particular aspects of the Strength of Automatic Control Systems which may require a separate assessment of the Civil Rotorcraft Automatic Control Systems.

8 CARRIAGE OF UNDERSLUNG LOADS

8.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 205	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
Complete Chapter	4-12	252 865	865	865	865	865

8.2 Chapter 205 and BCAR Section G have more detail than BCAR 29 FAR's and JAR's, nevertheless it would be expected that the civil installation would be compatible with the military derivative specification.

8.3 Reference should also be CAA Cap 426 - Helicopter External Load Operations or any similar FAA document.

9 DESIGN TO RESIST BIRD STRIKE

9.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 206	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
Complete Chapter	4-1,10 4-2 3.2.2	631 775	775	631 775	775	631 775

10 ACTIVE CONTROL SYSTEMS

10.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 207	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
Complete Chapter	3-6 4-8	672 1309	672 1309	672 1309	672 1309	672 1309

10.2 Active Control Systems (ACS) are provided not only for flight control but also to enhance structural life by manoeuvre and gust load alleviation. It should be noted therefore that all other chapters of DEF STAN 00-970 Part 2 are structurally inter-dependent with Chapter 207 diagrammaticaly shown in its Figure 1. All strength factors, where appropriate, as specified throughout Part 2 are applicable to the requirements of Chapter 207.

10.3 Civil requirements do not have a section paragraph entitled “Active Control Systems”. BCAR 29, FAR 27, 29, JAR 27, 29 paragraph 672 reads “Stability augmentation and power operated systems”. It will be necessary to include for combat and military necessity operations when assessing the compliance of an existing system in the military derivative.

PART 2 APPENDIX No. 2
STRUCTURAL STRENGTH AND DESIGN FOR FLIGHT
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 200: STATIC STRENGTH AND DEFORMATION

200	MIL-STD-1530	AIRCRAFT STRUCTURAL INTEGRITY PROGRAM, AIRPLANE REQUIREMENTS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8861	AIRPLANE STRENGTH AND RIGIDITY, FLIGHT LOADS

CONTROLLED DISTRIBUTION:

	AFGS-87221	AIRCRAFT STRUCTURES - GENERAL SPECIFICATION FOR
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

200 1.	MIL-STD-1530A	PARA: 1.2.1, 4.1.1, 4.1.2, 5.1.3.2.(g), 5.3.1, 5.3.4, 5.5
	MIL-T-8679	PARA: 3.1.1, 3.1.4, 3.2
	MIL-S-8698	PARA: 1.1, 3.1.1, 3.1.2, 3.1.3, 3.1.10, 3.2, 3.7, 6.3

2. DESIGN CASES

200 2.	MIL-STD-1530A	PARA: 5.1.5
	MIL-T-8679	PARA: 3.1.2
	MIL-S-8698	PARA: 3.2

3. ULTIMATE STRENGTH AND PROOF REQUIREMENTS

200 3.	MIL-S-8698	PARA: 3.1.1, 3.1.2, 3.1.3.2
	MIL-S-8561	

4. SUBSTANTIATION OF THE STATIC ALLOWABLE STRESS FOR GRADE A DETAILS

200 4.	MIL-STD-1530A	PARA: 5.2.1, 5.2.5, 5.3.8, 5.4.1, 5.4.2
	MIL-S-8698	PARA: 3.1.1, 3.7.3.2

5. DEMONSTRATION OF COMPLIANCE WITH THE ULTIMATE STRENGTH AND PROOF REQUIREMENTS FOR COMPLETE STRUCTURES OR COMPONENTS

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PART 3

STRUCTURAL STRENGTH AND DESIGN FOR OPERATION ON SPECIFIED SURFACES

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CHAPTER 300

DESIGN OF UNDERCARRIAGES - OPERATIONAL REQUIREMENTS

1 INTRODUCTION

1.1 The requirements of this Chapter are applicable to the design of undercarriage units for military rotorcraft and, where relevant, to the rotorcraft as a whole (see para 2.1 below).

1.2 It is generally assumed in the requirements that each unit of a wheeled rotorcraft has one wheel and a shock absorber but the requirements apply equally to units having two or more wheels per unit except as stated in Chapter 301 para 4.

1.3 The requirements apply also to units having skids, skis, or floats as well as or instead of wheels on Appropriate Surfaces (see Leaflet 300/1).

1.4 The requirements also apply, with special variations, to structural spring undercarriages (see also Chapter 301 para 10).

1.5 Special requirements are also given where relevant for:

(i) operation from ships,

(ii) operation from other surfaces (see Table 2 and Leaflet 300/1).

2 BASIC OPERATIONAL REQUIREMENTS

2.1 The rotorcraft and its undercarriage system shall be designed to provide a suitable suspension for the airframe, having regard to crew and passenger comfort, and to:

(i) avoid ground resonance (see Chapter 301 para 11),

(ii) absorb the energy of alighting in the specified landing conditions at mass M_T (see Chapter 304),

(iii) absorb as much energy as possible before collapse in an emergency alighting (see Chapter 307),

(iv) provide a suitable capability to operate from hard rough ground (see Chapter 305),

(v) provide directional control if required and some contribution to stability on the ground (see Chapter 302),

(vi) provide for ground and deck handling using the ground equipment specified (see Chapter 308),

(vii) provide for the forces imposed by picketing and tie down systems in specified operational conditions (see Chapter 309),

(viii) be compatible with the arresting cables (see Volume 1, Chapter 311) and ground handling equipment specified,

(ix) be compatible with Rapid Securing Systems (see Chapter 309),

(x) maintain adequate clearance for externally hung stores when operating on specified surfaces,

(xi) provide for a braking system if required by the Rotorcraft Specification (see Chapter 310).

3 OPERATION FROM SPECIFIED SURFACES

3.1 Table 2 and the definitions in Leaflet 300/1 define the extent to which the basic requirements cover operations on Appropriate Surfaces for particular types of undercarriage unit.

3.2 Where operations not covered by para 3.1 are required, the Rotorcraft Specification will state a requirement - particularly in the following cases:

(i) frequent take-off and alighting with forward speed on any surface,

(ii) taxying and turning on Other Surfaces (see Leaflet 300/1),

(iii) take-off and alighting with forward speed, and taxying and turning, on damaged and/or repaired runways,

(iv) manoeuvring on, take-off from, and alighting on, the sea or inland waters,

(v) manoeuvring on, take-off from, and landing on slopes greater than 12°,

(vi) where deliberate frequent operation from a soft surface is intended; in this case a California Bearing Ratio (CBR) will be stated.

3.3 The extent to which any of the requirements of para 3.2 can be met shall be determined by dynamic analyses including, where appropriate, a statistical assessment of probability of the cases considered. Such tests as are necessary to confirm the assumptions on which the analyses are based, shall be done to a programme discussed and agreed with the Rotorcraft Project Director.

3.4 The requirements shall be met in all practical combinations of mass, cg position, and stores configuration, with appropriate usage of controls at and below the density altitude given in the Rotorcraft Specification.

4 WIND CONDITIONS

4.1 Surface wind velocities are difficult to predict for rotorcraft operations but meteorological statistics indicate that:

(i) the average variation due to gusts is less than ±25%,

(ii) the extreme variation due to gusts, accounting for terrain and other obstacles, is less than ±75%.

The increases or decreases in velocity are equally probable and may occur from any direction relative to the mean.

4.2 For the purposes of undercarriage design for surface handling, taxying, taking-off and alighting, it shall be assumed that the maximum surface wind to be encountered is 30 knots steady component from any direction relative to the rotorcraft with up to ±25% unsteady component from any direction relative to the steady component. In these wind conditions, undercarriage characteristics shall not prevent the achievement of Level 1 performance on smooth level surfaces. (For definition of Levels, see Chapter 302 para 2.1). (See also design values in Chapter 304 Table 3).

4.3 Rotorcraft can be expected to operate in more severe conditions than those given in para 4.2, i.e., higher wind speeds and gusts, and from sloping or uneven surfaces. In this case the Rotorcraft Specification will state the worst conditions, and associated acceptable performance degradation and limitations.

5 SEA CONDITIONS

5.1 The selection of sea conditions for design, appropriate to a particular Rotorcraft Specification, shall take account of the following influences:

(i) The circumstances under which the rotorcraft could encounter the specified conditions, namely:

(a) take-off and alighting as a normal operation,

(b) emergency alighting,

(c) survival after emergency alighting, and

(d) take-off after emergency alighting.

(ii) The probability of meeting the specified conditions. This is the combined probability of:

(a) a given sea state occurring, and

(b) the rotorcraft being on the surface at that time.

(iii) The estimated ability of a given design to withstand the specified conditions.

(iv) The desired period of survival.

5.2 The Rotorcraft Specification will give the following information:

(i) A Sea State number (see Table 1).

(ii) The survival time required.

5.3 Chapter 307 states requirements for ditching.

6 STEERING AND CASTORING

6.1 The Rotorcraft Specification will state whether steering through the wheels is required and the angle through which steering and castoring is required (see Chapters 302 and 303).

7 BRAKING

7.1 The Rotorcraft Specification will state if brakes are required and which of the following functions are applicable:

(i) To stop the rotorcraft during landing.

(ii) To stop the rotorcraft following an aborted take-off.

(iii) To provide a means of restraining the rotorcraft from unwanted movement,

(a) on land on any sloping terrain specified,

(b) on board ship in any combination of deck angle and acceleration specified.

(iv) To facilitate and augment smooth manoeuvring and directional control.

7.2 Detailed requirements for wheel brakes are stated in Chapter 310. Stressing cases arising from braking are stated in Chapter 302 and Leaflet 302/2.

8 UNDERCARRIAGE RETRACTION - SELECTION AND INDICATION

8.1 The Rotorcraft Specification will state whether undercarriage retraction is required or not.

8.2 Engagement of alternative undercarriage configurations or modes shall only be possible following positive selective action by the pilot or ground crew.

8.3 An unambiguous indication of the state of the undercarriage at any instant shall be available to the pilot.

TABLE 1

SEA STATE CODE
(WORLD METEOROLOGICAL ORGANIZATION)

Sea State Code	**Description of Sea**	**Significant Wave Height**	
		Metres	**Feet**
0	Calm (Glassy)	0	0
1	Calm (Ripples)	0 - 0.1	0 -
2	Smooth (Wavelets)	0.1 - 0.5	- 1
3	Slight	0.5 - 1.25	1 - 4
4	Moderate	1.25 - 2.5	4 - 8
5	Rough	2.5 - 4.0	8 - 13
6	Very Rough	4.0 - 6.0	13 - 20
7	High	6.0 - 9.0	20 - 30
8	Very High	9.0 - 14.0	30 - 45
9	Phenomenal	Over 14	Over 45

Notes: 1 The Significant Wave height (SWH) is defined as the average value of the height (vertical distance between trough and crest) of the largest one third (1/3) of waves present.

2 The Maximum Wave Height (MWH) is usually taken as 1.6 x Significant Wave Height e.g., SWH of 6 metres gives MWH of 9.6 metres.

3 Arbitrary Wave Slopes and wind speeds for Sea States (SS) 3,4,5 and 6, which may be required for design of floats are given in Leaflet 307/2 Table 5.

TABLE 2

OPERATION ON VARIOUS SURFACES
APPLICABLE REQUIREMENTS

Surfaces	Vertical Take-off and Alighting	Take-off Alighting with Forward Speed	Taxying and Turning	Towing
Normal	Chapter 300 Chapter 301 Chapter 302 Chapter 304 Chapter 305	Chapter 300 Chapter 301 Chapter 302 Chapter 303 Chapter 304 Chapter 305 Chapter 310	Chapter 300 Chapter 301 Chapter 302 Chapter 303 Chapter 305 Chapter 310	Chapter 300 Chapter 305 Chapter 308
Other	As above but see Chapter 300 para 3.2	As above but see Chapter 300 para 3.2	Not covered. Air taxying is assumed.	Chapter 300 Chapter 305 Chapter 308
Damaged and/or repaired runways	As above but see Chapter 300 para 3.2	Not covered. See Vol 1 Chapter 305 if required.	Not covered. See Vol 1 Chapter 305 if required.	Chapter 300 Chapter 305 Chapter 308

LEAFLET 300/1

DESIGN OF UNDERCARRIAGES - OPERATIONAL REQUIREMENTS

DEFINITIONS OF SURFACES

1 NORMAL SURFACES

(i) Hard, smooth and level, wet or dry.

(ii) Hard, smooth, sloping up to 12°, wet or dry.

(iii) Either of the above with a thin covering of mown grass, snow or ice.

(iv) Minor hard obstacles, steps, and bumps, in Classes A and B of Chapter 305.

(v) Soft ground, soft sand, and thick snow within limitations determined by the type of unit fitted.

(vi) The decks of ships in sea states up to and including Sea State (SS)3.

2 OTHER SURFACES

(i) Hard rough ground.

(ii) Soft ground, soft sand, and thick snow with a low California Bearing Ratio (CBR) stated in the Rotorcraft Specification.

(iii) Either of the above with a thin covering of mown grass, ice or snow.

(iv) Hard obstacles in Class C or D of Chapter 305 as required by the Rotorcraft Specification.

3 APPROPRIATE SURFACES

(i) For units fitted with Wheels, Skids, or Skis, the appropriate surfaces are all Normal Surfaces.

(ii) For units fitted with floats the appropriate surfaces are:

(a) The sea in Sea States up to and including SS3.

(b) All Normal Surfaces within limits to be determined by dynamic analysis and agreed with the Rotorcraft Project Director.

4 Vertical and near-vertical operations comprise take-off and alighting as defined in Chapter 304, Table 3, Chapter 305 and Leaflet 304/3.

5 Where precise definition of the surface qualities quoted above is required, the Designer should propose and agree values of the relevant parameters with the Rotorcraft Project Director.

CHAPTER 301

DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS

1 INTRODUCTION

This Chapter collects together design requirements which are of general applicability.

2 ROTORCRAFT DESIGN REQUIREMENTS

2.1 The arrangement of the undercarriage units shall be such as to ensure that the rotorcraft will not tip over in the specified operating conditions including run-up of the rotor system.

2.2 All undercarriage stressing cases are applicable to the rotorcraft as a whole (see Part 2).

2.3 The materials used for construction of the undercarriage units shall be approved as required by Chapter 400 and shall also be agreed with the Rotorcraft Project Director.

2.4 The operation of critical parts shall not be affected by slush, mud, debris, ice or salt accretion.

2.5 Undercarriage systems shall be compatible with all other rotorcraft systems which operate in conjunction with them.

3 WHEELS AND TYRES (WHEN FITTED)

3.1 The size and number of wheels and tyres on main and auxiliary undercarriage units shall be such that the rotorcraft can operate at the maximum design take-off mass, M_T from surfaces having the bearing strength stated in the Rotorcraft Specification. (See also Leaflet 301/1).

3.2 All wheels shall be designed to withstand at least 3 times the specified inflation pressure before bursting.

3.3 All tyres shall be designed to withstand at least 2.67 times the maximum inflation pressure before bursting.

3.4 The minimum clearances between the surface of a tyre and the rotorcraft structure shall not less than those specified in BS M 45 ('Aircraft Tyres').

4 TWIN WHEEL AND MULTI-WHEEL UNITS

4.1 Where any unit of the undercarriage has more than one wheel and tyre, all requirements shall be met in each of the following cases:

(i) All tyres correctly inflated.

(ii) The most adverse combination of wheel loading which could be caused by variations of up to ±10% of the correct tyre, pressure, and/or

(iii) The most adverse combination of wheel loading which could arise from fitting combinations of largest and smallest possible tyres arising from wear and growth.

4.2 If differential wheel loading arising from consideration of the requirements of paras 4.1 (ii) and (iii) taken together or separately is more than 60:40 in critical cases, the problem shall be discussed with the Rotorcraft Project Director.

4.3 With any possible combination of one tyre deflated and the rest correctly inflated, the requirements of Chapter 304 shall be met at a vertical velocity of descent of 60% of the corresponding all-tyres-inflated cases.

4.4 Allowance shall be made where relevant for the effects of pitch and roll angles at touchdown.

5 SHIMMY

5.1 The requirements of this section apply to all rotorcraft with wheels.

5.2 Shimmy or related dynamic instabilities of any of the undercarriage units shall not occur at any speeds up to 1.15 times maximum ground speed in any rotorcraft cg and loading conditions and on any required surface. All relevant positions of steerable and castoring wheels shall be considered.

5.3 Some sub-critical torsional oscillations in the shimmy mode may be acceptable provided that:

(i) they can be shown to be not dangerous,

(ii) they are considered in fatigue analyses,

(iii) they do not affect operational capability.

5.4 Compliance with the requirements of paras 5.1 and 5.2 shall be shown either by analysis or tests or both. Analysis is required if tests are limited to maximum ground speed or if testing under para 7 (ix) of this Chapter is not required. (See Leaflet 301/1 for an acceptable approach to this analysis). The acceptability of analysis without tests, if proposed, and of residual oscillations shall be discussed with the Rotorcraft Project Director.

6 DETAIL STRESSING REQUIREMENTS APPLICABLE TO ALL CASES

6.1 Changes in the forces acting on a landing gear unit arising, from kinematic changes in geometry of the unit shall be taken into account unless they can be shown to be negligible.

6.2 The undercarriage and airframe may be assumed to be rigid unless the forces on the undercarriage are significantly affected by deformation, in which case the effects should be estimated conservatively.

Note: Paras 6.3 to 6.6 apply to wheeled undercarriage units only.

6.3 For static and quasi-static analyses, the value of μ for hard dry surfaces shall be considered at all critical values up to 0.8. For dynamic analyses, values higher than 0.8 may have to be considered in some cases. The values selected for the μ/slip ratio curve for pin-up shall be discussed with the Rotorcraft Project Director (see Fig.1).

6.4 All side forces shall be applied at the ground contact point with full allowance for pneumatic trail where appropriate.

6.5 Where wheels may be locked for landing, all forces shall be applied at the ground contact point.

6.6 Drag forces induced by wheel spin-up and spring-back shall be applied at the wheel hub. Drag forces induced by braking shall be applied at the contact point. Drag forces applied by obstacles (Leaflet 302/2 and Chapter 305) shall be applied at the hub.

6.7 In Chapter 302, Leaflet 302/2 and in Volume 1 Chapter 305, reference is made to the bump factor F. The value of F is to be determined in each case by the tyre characteristics in relation to the size of the obstacle for the class of surface specified.

7 TESTS

7.1 Chapter 303 states test requirements relating to steering systems. Chapter 304 states requirements relating to the landing cases. Chapter 306 states requirements for retraction and lowering tests. The following additional tests should be considered and a programme shall be agreed with the Rotorcraft Project Director:

(i) Static Strength tests.

(ii) Fatigue tests (see Volume 1 Leaflet 301/4).

(iii) Stiffness tests.

(iv) Ground resonance tests.

(v) Environmental tests including high and low temperatures, temperature shock cycling, mechanical shock, noise and vibration, humidity and fungus, and dust and salt fog.

(vi) Functioning tests.

(vii) Endurance tests.

(viii) Castoring tests (see Chapter 302).

(ix) Shimmy tests (see Leaflet 301/2).

(x) Shock absorber static compression tests.

(xi) Drop tests in addition to those required by Chapter 304.

(xii) Shock absorber recoil damping tests.

(xiii) Wheel and tyre tests (see Leaflets 310/3 and 310/4).

(xiv) Brake system tests (see Chapter 310 and Leaflet 310/2).

(xv) Fragmentation Tests (see Leaflet 719/4).

8 SKIDS, SKIS, OR FLOATS ON LAND

8.1 When it is proposed to use skids, skis, or floats on land, whether wheels are also fitted or not, the following requirements apply.

8.2 Where their use is associated with a particular surface they shall meet all relevant requirements on that surface. Limitations on their use on other possible surfaces shall be in accordance with Chapter 300 para 3.

8.3 Where the energy dissipation requirements of Chapter 304 para 5.4 cannot be met at design V_V the Designer shall consider the maximum V_V at which they can be met throughout the whole range of environmental conditions in which such landings are operationally necessary, and discuss possible limitations with the Rotorcraft Project Director.

9 FLOATS ON WATER

9.1 When it is proposed to fit floats instead of, or in addition to, wheels, the following requirements apply.

9.2 The floats shall provide adequate static buoyancy to support the rotorcraft on fresh water at 1.6 $M_T g$.

9.3 The floats shall be designed to provide adequate stability in all modes of motion in the sea state and maximum wind speed specified.

9.4 With floats fitted, there shall be adequate directional control to manoeuvre the rotorcraft on water in the sea state and maximum wind speed specified.

9.5 Floats shall be designed so that no more than 10% total buoyancy is lost, and adequate static stability is maintained, after a single leak.

9.6 Alighting loads derived from the requirements of Chapter 304 shall be applied to each float in all positions which may occur when contact is made with one or more waves, according to float length, in sea states up to and including SS 3 as defined in Table 1 of Chapter 300 and Table 5 of Leaflet 307/2.

9.7 Wave slopes between 1:10 and 1:20 and all critical wind speeds between zero and 30 kn from any direction shall be considered.

9.8 No single wave hitting a single float, (or two waves hitting two floats) from any direction, shall cause any handling effects, when the rotorcraft is airborne just after take-off or just before alighting which might lead to a catastrophe.

10 STRUCTURAL SPRING UNDERCARRIAGES

10.1 Where the energy of alighting is substantially absorbed by friction, by elastic deflection, or at higher vertical velocities by plastic deformation of the structure of the unit, the following requirements apply.

10.2 The Designer shall consider and discuss with the Rotorcraft Project Director the amount of damping available, when alighting on specified surfaces and whether or not the energy dissipation requirements of Chapter 304 Para 5.4 can be met at design V_V. Structural yielding of the spring member(s), but not of the back-up structure, shall be permitted above a value of V_V to be agreed, with the Rotorcraft Project Director. (see also Chapter 307).

10.3 Where the energy dissipation requirements of Chapter 304 para 5.4 cannot be met, the Designer shall consider the maximum V_V at which they can be met throughout the whole range of environmental conditions in which the rotorcraft is required to operate, and discuss possible problems with the Rotorcraft Project Director.

11 GROUND RESONANCE

11.1 The characteristics of the rotorcraft and its undercarriage as a whole shall be designed to avoid catastrophic ground resonance in all practical ground operating conditions and rotorcraft configurations (see Leaflet 304/1 para 4.3 and Chapter 305).

11.2 Some sub-critical vibration in the ground response mode may be acceptable provided that:

(i) normal operational, procedures will avoid any dwell at these frequencies,

(ii) they can be shown to be not dangerous,

(iii) they are considered in fatigue analyses,

(iv) they do not, affect the operational capability of the crew.

11.3 Compliance with the requirements of paras 11.1 and 11.2 shall be shown either by analysis or by tests or both. Any doubt about the acceptability of residual oscillations shall be discussed with the Rotorcraft Project Director.

11.4 The design shall be capable of accommodating the effects of any, minor defects in the rotor and undercarriage system. Variations of shock absorber characteristics, tyre pressures and damper performance shall be considered (see Chapter 304 - Table 3). An assessment shall be made of the limits beyond which resonance may be caused. Any special limitations arising shall be discussed with the Rotorcraft Project Director. The cases of a flat tyre or of a defective oleo shall also be considered.

12 AUTOROTATION AND ALIGHTING FOLLOWING ENGINE FAILURE

12.1 The Designer shall estimate auto-rotation speed/height boundaries and agree them with the Rotorcraft Project Director. This shall include alighting in autorotation of single engined rotorcraft.

12.2 When relevant handling and performance parameters have been established by flight tests, the effect of undercarriage strength on the boundaries shall be investigated by dynamic analysis of rotorcraft behaviour to confirm the limitations for Service use (see Chapter 304 - para 3.2 (ii)).

13 FATIGUE

13.1 Fatigue spectra and fatigue life estimates shall be discussed with the Rotorcraft Project Director.

13.2 Undercarriage loads shall be included, where relevant, in fatigue spectra for all other components of the rotorcraft.

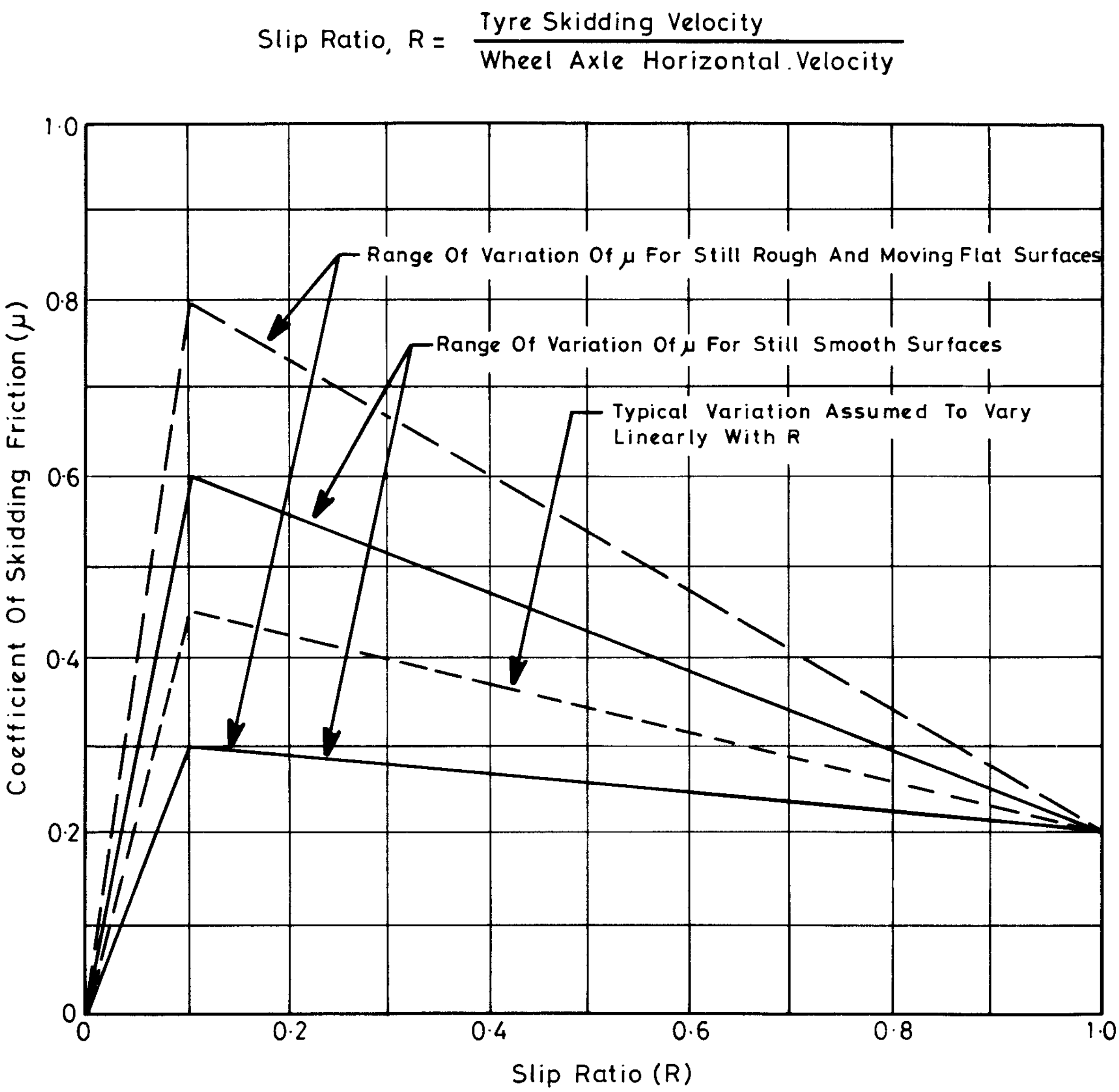


FIG.1 VARIATION OF COEFFICIENT OF SKIDDING FRICTION BETWEEN THE TYRE AND VARIOUS SURFACES WITH INSTANTANEOUS SLIP RATIO

LEAFLET 301/1

DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS
SHIMMY ANALYSIS

1 INTRODUCTION

1.1 Shimmy is defined as an oscillatory motion of an undercarriage unit which involves yawing and lateral motions of the wheels/tyres and the interplay of tyre-to-ground forces with elastic, inertia and damping forces. It is an instability phenomenon whose degree of severity depends on ground speed and in some cases the instability may be present only over a limited speed range. Those ground speeds at which there is a transition from stable to unstable motion are known as shimmy speeds.

1.2 Although shimmy is an instability phenomenon, the amplitude of oscillation in unstable conditions does not usually grow indefinitely but rather is limited by non-linearities of unit or tyre. It should not be confused with oscillations caused by periodic exciting forces such as are associated with wheel/tyre out of balance or brake judder, etc., which also show a speed dependence. If the gear is only marginally stable from a shimmy aspect, then these periodic exciting forces may provoke large oscillations which are best suppressed by improving the stability.

1.3 It is also theoretically possible for a non-oscillatory instability or divergence to occur, but in such cases the gear geometry and/or stiffness would probably be unusual. Although experience shows in a majority of cases that nose units are more likely to be susceptible to shimmy, due to inadequate torsional restraint, main wheel units are likely to shimmy if the combination of unit torsion, unit bending and tyre characteristics are critical.

2 MATHEMATICAL ANALYSIS

2.1 When a mathematical analysis is required, it should be used to assess the stability of the combined alighting gear/rotorcraft system throughout the operational ground speed range of the rotorcraft and not merely to find the shimmy speeds. The first phase of analysis need represent only the alighting gear, but a second phase incorporating rotorcraft structure should also be considered.

2.2 In order to develop a mathematical model in which the tyre elastic and ground forces are inevitably dominant, it is necessary to describe the forces exerted by a tyre when moving in an arbitrary manner along the ground (see Note). It is this feature which makes theoretical analysis of shimmy different from more familiar stability problems.

Note: Currently the best tyre mathematical model which can be recommended is based on work by Von Schlippe and Dietrich. This may be used unless a better method is available. Sufficient experimental data of the mechanical properties of tyres is available to enable the theory to be applied but it currently does not cover the full range of tyre types in use, (e.g., sizes, tyre pressure, ply rating, depth of tread etc.) and most of it has been obtained from non-oscillatory tests. Therefore the tyre data used in the theoretical investigation should be varied in order to allow for the uncertainties of the tyre dynamic characteristics.

2.3 The scope of the analysis should be such as to take account of the following:

(i) Ground speed.

(ii) Undercarriage unit geometry.

(iii) Tyre/wheel inertias, stiffness and damping.

(iv) Tyre-to-ground forces (e.g., self-aligning torque and side force).

(v) Tyre/wheel out of balance.

(vi) Unit inertias, stiffness (this includes backlash), structural damping.

(vii) Local attachment structure stiffness.

(viii) Normal modes of vibration (the unit and rotorcraft structure may be treated in a unified modal manner).

(ix) Gyroscopic forces.

(x) Steering system impedance.

2.4 It may be assumed that small displacement theory is adequate for use together with linearised tyre and structural equations of motion provided that the non-linear terms are covered by variations of the linearised constants, and the results interpreted in an appropriate manner. The characteristic roots (frequency and damping) of the equations of motion can then be found for a series of speeds for each landing gear configuration.

2.5 Systems which rely heavily on friction for shimmy suppression may require a different method of analysis from that of para 2.4, such as a time-history analysis in which the variation of decay of the motion in response to various levels of disturbance should be examined.

2.6 Important items of data should be varied in order to evaluate the sensitivity of the system's stability.

2.7 When mathematical analysis predicts shimmy, or stability which is only marginal, in the operational ground speed range of the rotorcraft, then a decision must be taken whether to modify the design or to undertake tests to demonstrate that the data used in the analysis does not in practice take values which are unacceptable.

3 ASSOCIATED TESTS

3.1 Tests in support of the theoretical analysis will often be necessary to obtain reliable values for the data required in the analysis. These may include:

(i) static stiffness tests of the unit and attachment structure,

(ii) impedance tests of the steering jack system,

(iii) resonance tests,

(iv) rolling (oscillatory or non-oscillatory) tyre tests.

4 DAMPING

4.1 The damping, required will be determined by consideration of the effect of shimmy on fatigue strength and on crew performance. Any oscillation which is not eliminated by the design/test process must be included in the fatigue analysis of all relevant parts of the structure and the results of all shimmy tests must be examined to ensure that the fatigue analysis provides for these residual oscillations. For preliminary theoretical calculations, damping should be set at a level (about 5% is suggested) which will ensure that there is adequate margin to provide for the worst value of any single parameter, or likely combination of parameters, so as to ensure that it is extremely improbable that the damping, will ever become negative. Adverse tyre size (caused by wear or incorrect pressure or a combination of the two) and both wet and dry surface conditions should be considered. Wear of mechanical parts, within permitted limits, should also be considered.

LEAFLET 301/2

DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS
SHIMMY TESTS

1 INTRODUCTION

1.1 This leaflet describes taxying tests which may be necessary specifically to check whether anti-shimmy requirements have been met.

1.2 When it is agreed with the Rotorcraft Project Director that tests are required in addition to or in place of analytical calculations to ensure that no shimmy, divergence or related dynamic instabilities can occur, then these tests must be potentially capable of exciting the phenomenon concerned. An acceptable method of testing is detailed below.

2 TESTS

2.1 Fix a plank to the runway surface at an angle to the direction of forward motion and of such a shape and height that the forces induced on the undercarriage unit excite the fundamental vibration mode of the undercarriage (side bending and/or torsion). The edges of this plank may be rounded to avoid tyre damage.

2.2 Taxi the rotorcraft over the plank at a number of speeds up to the speed which will allow calculations to be extrapolated accurately to cover all possible ground speeds both normal and emergency. Analyse the damping of the response to show that it does not fall below that required.

LEAFLET 301/3

DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS
GLOSSARY OF SYMBOLS USED IN PART 3

The following symbols are used throughout Part 3. Other conventions have been examined and it has been found impossible to adopt any one convention.

A	Auxiliary
a	Braking deceleration coefficient
CBR	California bearing ratio - A measure of soil bearing strength (%) of the ground from which it is required to operate. Assessed by use of a cone penetrometer. (See "US Army Handbook for Penetrometer, Soil, Hand-operated, Army Code No. 60285").
D	Drag force
F	Bump factor applicable to the vertical reaction
G_y	Lateral acceleration coefficient (ship motion)
G_z	Vertical acceleration coefficient (ship motion)
g	Acceleration due to gravity
H	Height of cg above ground
h	Height of cg above main wheel hub
K	Radius of gyration of rotorcraft
k	A value of a variate divided by $_N$ or $_R$ as appropriate
l_m	Horizontal distance of cg from main unit(s)
l_n	Horizontal distance of cg from nose unit
l_t	Horizontal distance of cg from tail unit
M	Main, or Mass of unsprung parts
M_E	Design take-off mass M_T but with minimum usable fuel
M_L	Design landing mass
M_T	Design take-off mass
M_L^T	Any value between M_T and M_L
N	Nose, or Number
n_y	Lateral acceleration coefficient

n_z	Normal acceleration coefficient
P	Towing load
P_r	Probability
p	Tyre pressure
R	Undercarriage design peak vertical reaction
R_A	Auxiliary unit design peak vertical reaction
R_M	Main unit design peak vertical reaction
R_N	Nose unit design peak vertical reaction
R_T	Tail unit design peak vertical reaction
RMS	Root -Mean Square
S	Side force
SS	Sea State number
S_A	Stroke of auxiliary unit shock absorber plus tyre
S_M	Stroke of main unit shock absorber plus tyre
S_N	Stroke of nose unit shock absorber plus tyre
S_T	Stroke of tail unit shock absorber plus tyre
V	Vertical force
V_A	Maximum permitted forward ground speed for alighting
V_B	Maximum permitted forward ground speed for take-off
V_C	Horizontal forward velocity (relative to surface)
V_L	Lateral velocity (relative to surface)
V_V	Vertical velocity (relative to surface)
V_x	Forward velocity (ship motion), or Closing Speed
V_y	Lateral velocity (ship motion), or Cross Wind, or Drift
V_z	Vertical velocity (ship motion)
V_1V_2	Wind velocities at heights h_1, h_2 above the surface
$\bar{x}$	Mean of a distribution or set of statistics

$\eta_M \eta_N \eta_T$	Overall efficiency of shock absorber/tyre combination - main, nose, tail unit
θp	Pitch attitude (relative to surface)
θ_R	Roll attitude (relative to surface)
θ_Y	Yaw attitude (relative to surface)
θ_x	Pitch attitude (ship motion)
θ_y	Roll attitude (ship motion)
θ_z	Yaw attitude (ship motion)
μ	Coefficient of friction
σ_N	Standard deviation of Normal Distribution
σ_R	Rayleigh distribution Shape Parameter

CHAPTER 302

DESIGN OF UNDERCARRIAGES - HANDLING QUALITIES, STRENGTH AND STIFFNESS ON SPECIFIED SURFACES

1 INTRODUCTION

1.1 This Chapter states general control requirements and associated strength and stiffness requirements which apply while the relevant undercarriage units are in contact with the appropriate specified surfaces (see Chapter 300 Table 2 and Leaflet 301/1).

1.2 Compliance with the requirements shall be demonstrated by dynamic analysis of rotorcraft behaviour and/or by tests agreed with the Rotorcraft Project Director.

1.3 The requirements shall be met whether undercarriage steering is provided or not and whether braking is provided or not. Where steering is provided the requirements of Chapter 303 are also applicable. Where braking is provided, the requirements of Chapter 310 also apply.

1.4 The extent to which the requirements provide a satisfactory standard when operating from the decks of ships in sea states higher than 3 shall be determined by dynamic analysis of rotorcraft behaviour based on the results of trials at sea and the strength of the undercarriage. Appropriate limitations shall be derived, discussed and agreed with the Rotorcraft Project Director.

2 HANDLING QUALITIES

2.1 In case of severe climatic conditions or failure of any one of the systems contributing to satisfactory control on the ground, the level of pilot work load necessary to achieve control may rise and the degree of control achieved may be lower. For this purpose the requirements are expressed in terms of levels of handling qualities defined as follows:

(i) Level 1 - Handling qualities clearly adequate for the manoeuvre.

(ii) Level 2 - Handling qualities adequate to accomplish the manoeuvre, but with some degradation in control or increase in the work load of the pilot, or both.

(iii) Level 3 - handling qualities such that the rotorcraft can just be controlled, or the total work load of the pilot is approaching the limit of his capacity.

2.2 All rotorcraft shall have Level 1 handling qualities when taking-off, alighting, or turning on the spot, on the Appropriate Surfaces for the type of unit fitted (see Leaflet 300/1).

2.3 All rotorcraft shall have positive stability in each major axis while any unit or pair of units are in contact with the Appropriate Surfaces (see Leaflet 302/1).

2.4 Where operation from the decks of ships is specified, the requirements of para 2.2 shall be met at Level 1 in sea state 3 and should be met if possible in sea state 6 at Level 2.

2.5 Where operation with forward speed is specified:

(i) satisfactory handing qualities shall be achieved at all speeds up to $1.15V_A$,

(ii) the take-off requirement of para 2.2 shall be met at all speeds up to $1.15V_B$,

(iii) straight taxying, transition to constant radius cornering (and the reverse) and turning at specified radii, shall be met with Level 1 handling qualities.

2.6 Where operation with forward speed on Other Surfaces or on damaged or repaired runways is specified, the applicability of the requirements of Chapter 305 and of Volume 1, Chapter 305 respectively shall be discussed with the Rotorcraft Project Director.

2.7 Further information on handling qualities for wheeled rotorcraft on the Appropriate Surfaces is given in Leaflet 302/1.

3 DIRECTIONAL CONTROL REQUIREMENTS FOR WHEELED STEERING AND CASTORING UNITS

3.1 The requirements of this section apply to all wheeled undercarriage units which have directional freedom whether steerable or not.

3.2 The dynamic behaviour of the undercarriage units shall not cause unacceptable oscillation, vibration or rotorcraft instability in any specified manoeuvres, either directly or through interaction with the rotorcraft and pilot.

3.3 The units shall not shimmy in any specified manoeuvres on specified surfaces (see para 2 above and Chapter 301).

3.4 Further design requirements for steerable or castorable undercarriage units are given in Chapters 303, 306 and 308.

4 STRENGTH AND STIFFNESS

4.1 The rotorcraft undercarriage and back-up structure shall have proof factors of not less than 1.0 and 1.125 respectively and an ultimate factor of not less than 1.5 in all applicable cases.

4.2 Where there is a requirement for taxying and turning on Normal Surfaces and brakes are provided on the wheels, all the stressing cases of Leaflet 302/2 shall be met. Where wheel brakes are not provided, the stressing case of Leaflet 302/2 para 3.2 only is applicable.

4.3 Where the stressing cases of Leaflet 302/2 refer to a bump, the requirements of Chapter 305 shall be met using the size of bump or step applicable (see Leaflet 300/1 and the Rotorcraft Specification).

4.4 The extent to which the requirements provide adequate strength, stiffness, and energy absorption for take-off and alighting with forward speed and for taxying over Other Surfaces and Damaged or Repaired Runways (see Leaflet 300/1), and in particular over hard rough ground, shall be determined by dynamic analysis and tests.

LEAFLET 302/1

DESIGN OF UNDERCARRIAGES - HANDLING QUALITIES, STRENGTH AND STIFFNESS ON SPECIFIED SURFACES HANDLING QUALITIES

1 INTRODUCTION

1.1 One aim of Chapter 302 is to ensure a basic standard of controllability of wheeled rotorcraft when operating from the specified surfaces. This Leaflet gives background information relating to this requirement.

1.2 Even where such operations are nominally vertical, some consideration must be given to alighting on an obstacle (see Chapter 305).

1.3 Even where wheel steering is provided, the most powerful means of directional control on the ground will be the tail rotor or main rotors used differentially. It will, therefore, be necessary to take account of rotor forces when considering stability and control of the rotorcraft on the ground.

2 STABILITY

2.1 The basic requirement is for positive stability in each axis (see Chapter 302 para 2.3). The two broad aspects which must be considered are:

(i) stability of the rotorcraft in pitch and roll,

(ii) the yaw stability as defined by the ability to follow a desired path without undue effort.

2.2 Chapter 301 para 2.1 requires that the rotorcraft shall not tip over in any operating conditions. It is considered that displacement of the rotorcraft vertical axis by 25° in any direction in the static condition is the minimum to meet this requirement and 30° is a desirable aim. On present experience, this static condition will adequately cover the requirement in Chapter 302 para 2.

3 CONTROL

3.1 Rotorcraft need a capability for accurate positioning on the ground. Steering control through the undercarriage however is not the only solution since a rotorcraft can be lifted off into the hover in order to reposition. Equally, rotor forces will produce more powerful control than wheel steering.

3.2 If wheel steering is provided, this alone should be sufficient to achieve a standard of turning performance agreed with the Rotorcraft Project Director.

LEAFLET 302/2

DESIGN OF UNDERCARRIAGES - HANDLING QUALITIES, STRENGTH AND STIFFNESS ON SPECIFIED SURFACES STRENGTH AND STIFFNESS

1 INTRODUCTION

1.1 This Leaflet applies to wheeled undercarriages on substantially flat smooth hard surfaces and gives stressing cases which will normally be acceptable to demonstrate compliance with the strength and stiffness requirements of Chapter 302. Strength requirements for landing are given in Chapter 304.

1.2 With appropriate values of relevant parameters, they may also be used for preliminary design for operation from other surfaces (see Leaflet 300/1). However, they may not be used as the sole means of demonstrating compliance with the requirements of Chapter 300 para 3.3 for which dynamic analysis and ground tests are also required.

1.3 These stressing cases are defined mainly for conditions which are static or quasi-static (in which the product of mass and the specified acceleration contributes to the overall balance of forces applied to the rotorcraft). In either circumstances, the tyre and shock absorber closures should be those which the forces applied to the undercarriage units would produce in a static condition. Alternatively, the descriptions of the situations in which the cases are assumed to arise may be used as a basis for dynamic analyses: the full range of appropriate forward speeds should then be covered.

1.4 Due account should be taken of the bump factor F. See Chapter 301 para 6.7 and Chapter 305 for dimensions of obstacles for which the class will be stated in the Rotorcraft Specification or by the Rotorcraft Project Director. Chapter 301 gives the points of application of drag and side forces.

2 CASES APPLICABLE TO NOSE UNITS ONLY

2.1 DYNAMIC BRAKING

2.1.1 Vertical loads should be derived from a dynamic analysis of the motion of the rotorcraft at the most adverse combination of mass and cg and most adverse pitching condition following the sudden application of all wheel brakes. It may be assumed that the nose unit is at zero castor angle with zero side force and that the drag force at the nose unit is zero if it is not equipped with brakes. A drag reaction at each wheel equipped with brakes and in contact with the ground should be assumed acting at the ground and combined with the vertical reaction. In the absence of data giving the brake drag response of all braked units, each drag reaction should equal the peak dynamic brake force but need not exceed 0.8 of the vertical reaction at the wheel.

2.2 STEADY BRAKING WITH BUMP

2.2.1 The rotorcraft is assumed to traverse a step or bump defined in Chapter 305 obliquely in crosswind with the nosewheel at any angle up to 20° either side of the rotorcraft longitudinal axis.

Vertical loads should be derived in the steady state condition at the most adverse combination of mass and cg position following the application of all wheel brakes. The total wheel braking force on the rotorcraft is $M_T ga$, where a $< \frac{1}{3}$ when wheel brakes acting alone provide the critical condition. The nose unit loads may be estimated from the following formulae:

(i) $$V = \frac{Fg}{1_m + 1_n}\left[M\frac{T}{L}.1_m + M_T Ha\right]$$

(ii) S = 0.25 V

(iii) D = 0.25 V without nosewheel braking

(iv) D = 0.4 V with nosewheel braking, where a drag force of 1.33 x drag corresponding to maximum nose undercarriage braking effort, is applied at the ground and the remainder of drag force D, if any, is applied at the hub.

3 CASES APPLICABLE TO NOSE AND MAIN UNITS OF NOSEWHEEL TYPE ROTORCRAFT

3.1 ASYMMETRIC BRAKING

3.1.1 For nosewheel rotorcraft, the rotorcraft is assumed to be in the normal taxying attitude with the vertical load factor at the cg equal to 1.0. One main unit should be assumed braked and developing a drag load at the ground equal to either 0.8 of the vertical reaction at that unit in the steady state or the peak dynamic brake drag whichever is the less. The rotorcraft is assumed to be in static equilibrium, with side loads at the main and nose units reacting the yawing moment, and with the vertical loads at the main and nose units reacting the pitching moment. The side load at the nose unit will be acting at the ground, and need not exceed the vertical reaction multiplied by a coefficient of friction of 0.8 for steerable nosewheels and 0.25 for freely castoring nosewheels. The nose unit will be at zero castor angle.

3.2 TURNING

3.2.1 The rotorcraft is assumed to execute a turn at constant forward speed and constant radius in which a vertical load factor at the cg of 1.0 is combined with lateral load factors up to 0.5 except that they shall not exceed a level at which the rotorcraft would overturn. It is assumed that the rate of turn is sustained, without the use of wheel brakes, either by use of nosewheel steering alone or by the use of tail or differential main rotor thrust. The range of radii of turn and forward speeds used will be that practicable for the rotorcraft. The distribution of side force amongst the wheels will be determined by an analysis which takes account of the tyre characteristics.

4 CASES APPLICABLE TO TAILWHEEL UNITS ONLY

4.1 REVERSE BRAKING

4.1.1 The rotorcraft should be in the three point attitude. The vertical load factor at the cg should be equal to 1.0 at the maximum design mass M_T with forward acting drag reaction at the ground of 0.8 of the vertical reaction or the peak dynamic brake drag whichever is the less. The tailwheel should be considered in both the castored aft and castored forwards positions.

5 CASES APPLICABLE TO MAIN AND TAIL UNITS OF TAILWHEEL TYPE ROTORCRAFT

5.1 TURNING

5.1.1 The rotorcraft is assumed to execute a turn at constant forward speed and constant radius by means of tail or differential main rotor thrust. The vertical load factor at the cg should be 1.0 and the tail wheel although castored should be assumed to be capable of resisting side load. A lateral load factor up to 0.5 should be applied at the cg except that it should not exceed a level at which the rotorcraft would overturn. The range of radii of turn and forward speeds used will be that practicable for the rotorcraft. The distribution of side force amongst the wheels will be determined by an analysis which takes account of the tyre characteristics.

5.2 TAXYING OVER A BUMP

5.2.1 The rotorcraft is assumed to be in the three point attitude at the maximum design mass M_T. The tailwheel is assumed to be capable of resisting drag and side forces. A bump factor should be applied to each unit in turn and all the forces recalculated for that unit using D = S = 0.25 V. The bump factor should not be less than 2.0, except when traversing an arresting cable (see para 8.2).

6 CASES APPLICABLE TO MAIN UNITS ONLY

6.1 DYNAMIC BRAKING

6.1.1 For nosewheel type rotorcraft, the rotorcraft attitude is assumed to be that corresponding to the auxiliary nose unit just clear of the ground with the auxiliary landing gear shock strut fully extended and the main landing gear and tyre compressed to their static position at the specified load factor. For tailwheel types, the rotorcraft reference axis is assumed to be horizontal. The vertical load factor acting at the cg should be 1.2 at maximum landing mass. A drag force at each wheel in contact with the ground is assumed to be acting at the ground. This will be equal to 0.8 of the vertical force or the peak dynamic brake drag, whichever is the less, and should be combined with the vertical force.

6.1.2 For rotorcraft with four main units, the rear units are assumed to carry a vertical load of 1.33 times their static load at maximum landing mass with a vertical load factor of 1.0 at the cg. A drag force equal to 0.8 of the vertical force or the peak dynamic brake drag, whichever is the less, is applied to the ground contact points of the rear wheels. The assumed rotorcraft attitude is with the forward wheels just clear of the ground.

6.2 REVERSE BRAKING

6.2.1 For nosewheel type rotorcraft, the rotorcraft is assumed to be supported by main units only with the nose gear fully extended just clear of the ground. For tailwheel types, the rotorcraft is assumed to be in the three point attitude. For rotorcraft with 4 main units, the rearmost units are assumed to carry a vertical load appropriate to the maximum which can be developed in static balance. The vertical load factor at the cg should be equal to 1.0 at the maximum design take-off mass M_T with forward acting drag force at the ground of 0.8 of the vertical force or the peak dynamic brake drag whichever is the less.

6.3 STEADY BRAKING WITH BUMP

6.3.1 One main unit is assumed to traverse a step or bump obliquely with all units centralised. The total vertical load on the main units should be derived in the steady state condition at the most adverse combination of weight and cg position following the application of all wheel brakes. The total wheel braking force on the rotorcraft is $M_T ga$, where $a \leq \frac{1}{3}$ when wheel brakes acting alone provide the critical condition. The vertical load on a main unit, V, in the steady state is:

(i) $$V = \frac{g}{2(1_m + 1_n)} \left[M \frac{T}{L} 1_n - M_T Ha \text{ for} \right]$$

2 main units aft of the cg

(ii) $$V = \frac{g}{2(1_m + 1_n)} \left[M \frac{T}{L} 1_n + M_T Ha \text{ for} \right]$$

2 main units forward of the cg

The bump factor F should be applied separately to re-estimate V for each main unit in turn. The side force for each main unit may then be estimated by S = 0.25 V and the drag force for each main unit by D = 0.4 V where only the drag developed by the brake acts at the ground.

6.4 TAXYING OVER A BUMP

6.4.1 The rotorcraft is assumed to be in the static state attitude at the maximum design mass M_T. A bump factor should be applied to each unit in turn and all the forces recalculated for that unit using D = S = 0.25 V. The bump factor should not be less than 2.0, except when traversing an arresting cable(see para 8.2).

6.5 PIVOTING OF A MAIN UNIT

6.5.1 With the brakes 'locked-on', on the wheels of the undercarriage unit about which the rotorcraft is pivoting, the rotorcraft will pivot about the centroid of the contact area of those wheels. The vertical load factor at the cg should be 1.0 and the tyre coefficient of friction should be 0.8 or such alternative coefficient of friction as defined in the rotorcraft specification.

7 TYRE DEFLATED CASES

7.1 In all possible cases of one tyre deflated and the rest correctly inflated, an ultimate factor of not less than 2.0 is to be achieved on the loads arising when stationary in the static attitude at the most critical mass and cg position.

7.2 On twin tyre undercarriages, the tyre loads in the tyres inflated cases are assumed to be applied to the remaining wheel but reduced by 50 per cent. However, the vertical load will not be less than that in the static condition with a vertical load factor of 1.0 at the cg.

7.3 On twin tyred nose undercarriages, for all possible cases of one tyre deflated and the rest correctly inflated, an ultimate factor of not less than 1.5 is to be achieved on the loads due to gradual full application of all wheel brakes (these being, correctly adjusted and excluding the brake on the wheel with the deflated tyre if it has a brake) while taxying at all rotorcraft masses up to M_T. The drag force at each braked wheel should equal the peak dynamic brake force but need not exceed 0.8 of the vertical reaction at the wheel. All castored or steered positions of the nosewheel should be considered.

8 TRAVERSING AN ARRESTING CABLE

8.1 This condition can arise at any time during ground manoeuvring and the take-off or landing runs. However, there is no requirement to consider the landing impact point of any undercarriage unit coincident with an arresting cable position. The size of the tyres must be adequate to provide for normal compression and the diameter of the arresting cable assumed need not exceed 32 mm.

8.2 This case applies to nose, tail and main units, and should be treated as specific size of bump for the purposes of paras 2.2, 5.2, 6.3 and 6.4.

8.3 For further information on airfield arresting cables, see Volume 1, Leaflet 311/1.

9 TOWING

9.1 Towing requirements are given in Chapter 308.

CHAPTER 303

DESIGN OF UNDERCARRIAGES - WHEEL STEERING

1 INTRODUCTION - GENERAL REQUIREMENTS

1.1 This Chapter states additional design requirements when a wheeled steering system is necessary. Rotorcraft steering on the ground is normally accomplished by use of the tail rotor. There are however some circumstances, in particular on board ship, where the provision of wheeled steering, during forward motion is necessary. In addition ship-borne use may require the selection of discrete wheel positions.

1.2 The steering system, in conjunction with the tail rotor, shall meet the directional control response requirements of Chapter 302 and of the Rotorcraft Specification.

1.3 The steering system shall not induce unacceptable vibration, rotorcraft instability or shimmy, with the system engaged or disengaged, at all required ground speeds.

1.4 Failure of any part of the steering system shall not:

(i) cause unacceptable oscillation or vibration,

(ii) degrade the handling qualities of the rotorcraft in vertical operation by more than one Level (defined in Chapter 302).

1.5 The probability of failure of any part of the steering system which could cause the wheels to turn when on or off the ground, shall be extremely remote (see BCAR Section G, Chapter G1-2 para 7.2.5). In the event of failure, the unit should self-centre or castor.

1.6 A failure effects and hazard analysis shall demonstrate compliance with the requirements above to the satisfaction of the Rotorcraft Project Director.

2 STEERING SYSTEM DESIGN REQUIREMENTS

2.1 When engaged the steering system shall position the wheels at the angle demanded by the cockpit controls and restore them to that position after an external disturbance.

2.2 Engagement of the steering system at any time shall not hazard the rotorcraft whether on the ground or in the air. In particular, it shall not prevent correct alignment of the wheels before touchdown.

3 STEERING REQUIREMENTS

3.1 The steering angles provided shall enable the rotorcraft to turn or rotate as defined in the Rotorcraft Specification.

3.2 For ground handling, a free castoring range shall be provided as required by the Rotorcraft Specification (see also Chapter 308).

4 STEERING TORQUE

4.1 The steering system shall be designed to provide sufficient output torque to meet the general directional control requirements of Chapter 302 and of the Rotorcraft Specification.

4.2 Hydraulic or electric system safety devices shall be designed to operate at 1.33 times the output torque required by para 4.1.

5 STRENGTH AND STIFFNESS

5.1 The rotorcraft structure, the undercarriage and its steering system shall have proof and ultimate factors of not less than 1.125 and 1.5 when meeting the requirement of para 4 on a level surface having a static coefficient of friction at the ground of 0.8 with a load factor of 1.0 at the cg.

5.2 The requirement of para 5.1 shall be met in the most adverse combination of mass and cg position.

5.3 The strength and stiffness of all components of the system when installed shall ensure operation of the system without failure under all design conditions.

5.4 The fatigue lives of all components shall be shown by analysis or test, using a spectrum of loads agreed with the Rotorcraft Project Director, to be at least equal to the life required by the Specification.

6 TESTS

6.1 In addition to relevant tests listed in Chapter 301 para 7, compliance with design requirements of this Chapter and of Chapter 302 shall be demonstrated during taxying trials at an early stage in development.

LEAFLET 303/1

DESIGN OF UNDERCARRIAGES - WHEEL STEERING
WHEEL STEERING SYSTEMS

1 GENERAL

1.1 This Leaflet amplifies the requirements for the design of rotorcraft undercarriage wheel steering systems given in Chapter 303.

2 STEERING CONTROL SYSTEM DESIGN

2.1 A mechanical input and follow-up linkage is preferred for controlling the steering power system. Where electric or electronic control systems are used, the circuit reliability will have to be included in the safety assessment of the system.

3 SHIMMY

3.1 The details of a mathematical approach required to demonstrate adequate shimmy damping are given in Leaflet 301/1.

3.2 Steering system parameters used in the shimmy analysis should be measured on the rotorcraft during early vibration tests and, if the parameters are found to be different from those assumed, the analysis should be repeated (see Leaflet 301/2).

4 STEERING TORQUE

4.1 Consideration should also be given to the effect of shock loads arising when the wheels strike the appropriate step or bump (see Chapter 305) at the most adverse position, on those parts of the unit which could be overstressed before any torque limiting device fitted can operate.

4.2 A check should also be made to ensure that the torque and strength requirements can be met at the highest and lowest temperatures and at the extremes of tyre and oleo pressures specified for the rotorcraft.

CHAPTER 304

DESIGN OF UNDERCARRIAGES - ALIGHTING

1 INTRODUCTION - GENERAL REQUIREMENTS

1.1 This Chapter states strength requirements for alighting on all surfaces.

1.2 The preliminary stressing cases of para 2 are mandatory for project design and initial stressing. The dynamic analyses of para 3 are mandatory to verify the design when the necessary data becomes available unless otherwise agreed by the Rotorcraft Project Director. The supplementary stressing cases of para 4 are optional.

1.3 Where a rotorcraft is required to alight according to more than one operational technique, the requirements shall be applied separately to each. If a particular technique involves design cases not covered by the requirements, these will be stated in the Rotorcraft Specification.

1.4 Where more than one operational loading or geometric configuration of the rotorcraft is specified, the requirements shall be met in each configuration. Where a particular loading or geometric configuration arises from a failure state in which alighting is required, the Rotorcraft Specification will state the conditions in the requirements shall be met.

2 PRELIMINARY STRESSING CASES - ALL PROJECTS

2.1 The cases given in Tables 1 and 2 are applicable to all rotorcraft, according to type and alighting surface, unless varied by the Rotorcraft Design Specification or by subsequent agreement with the Rotorcraft Project Director (see also Chapter 300 Table 2).

2.2 All applicable cases shall be considered in association with the design V_V.

2.3 Unless otherwise specified, all requirements shall be met at design take-off mass M_T and over the whole range of take-off cg positions. The requirements shall also be met at any critical combination of cg position with a mass less than M_T (see Leaflet 304/1 para 3.2).

2.4 In all cases, the forces applied to the main units shall be balanced by inertia forces and moments at the cg. The only air load which need be considered is the aerodynamic lift assumed equal to 2/3rds of the weight of the rotorcraft.

2.5 The cases given in Tables 1 and 2 shall be considered in association with the following requirements:

(i) The design reaction for the main units (R_M) nose unit (R_N) and tail unit (R_T) respectively, shall be the highest reaction arising in the appropriate attitude of Table 4 (see Leaflet 304/1 para 3.5).

(ii) For laterally disposed auxiliary units, the design reaction (R_A) shall be the maximum reaction obtained from a dynamic analysis of rotorcraft behaviour starting from initial touchdown in each of the appropriate attitudes defined in Table 4.

(iii) All critical values of side and drag force between zero and the values given in Table 2 shall be considered.

(iv) The values of shock absorber closure shall be as given in Table 2 (but see Leaflet 304/1 para 3.7.2).

(v) Design velocities may have to be increased if mechanical assistance to alighting is used (see Table 1).

3 DYNAMIC ANALYSES - ALL PROJECTS

3.1 Dynamic analyses of the forces in the undercarriage unit are required for the high drag, and spring-back cases and for the wheel-brakes-on cases of Table 2.

3.2 Dynamic analyses of rotorcraft behaviour are required and the results shall be submitted to the Rotorcraft Project Director for the following cases:

(i) To demonstrate compliance with the ground resonance requirements of Chapter 301 paras 2.1 and 11.

(ii) To establish height and speed limitations for the autorotative landing case of Chapter 301 para 12 within design reactions for the undercarriage.

(iii) To show satisfactory strength, stiffness and, where relevant, flight handling characteristics, when landing at design V_V, in each of the attitudes specified in Table 4 (see also Chapter 305 para 3.2).

(iv) To show compliance with the requirements of Chapter 309.

(v) To generate undercarriage fatigue spectra.

3.3 Where appropriate, these analyses shall include but not be limited to the following:

(i) The dynamic response characteristics of the tyre and shock absorber including internal frictional forces.

(ii) Elastic modes of the rotorcraft as a whole for all critical combinations of mass and cg position, including symmetrical and asymmetrical internal and external store loadings for which strength is required.

(iii) Gyroscopic forces.

(iv) Variations of the friction forces between the tyre and landing surface shown in Fig.1 of Chapter 301.

(v) Variations of the servicing conditions for the undercarriage shock absorbers and tyres from those specified.

(vi) Undercarriage unit drop test results.

3.4 These analyses are required to determine rotorcraft response in the specified conditions and to demonstrate that the design meets the requirements.

3.5 Results of these analyses, particularly the effects on critically loaded parts of the structure and on the store stations, shall be shown in the Rotorcraft Type Record and submitted to the Rotorcraft Project Director.

4 SUPPLEMENTARY AND SPECIAL STRESSING CASES

4.1.1 Where stated in the rotorcraft specification or subsequently agreed with the Rotorcraft Project Director, secondary stressing cases for any unit nay be substituted for or added to the preliminary stressing cases or dynamic analyses. These cases may be defined according to one of the following procedures:

(i) Arbitrary definition - in which a particular parameter is assigned a specific extreme value (or a small selection of parameters are assigned specific values) and the remainder are held at mean or other stated values.

(ii) Statistical definition - in which a specified selection of parameters are varied and the remainder are assigned statistically determined values. Analysis will then provide a number of design cases having a specified probability of occurrence.

(iii) Dynamic definition - analysis of undercarriage unit design parameters or of rotorcraft flight dynamic parameters, or both, additional to those stated in para 3 above.

4.2 The form of definition to be used, the probabilities required, and the parameters to be considered will be stated in the Rotorcraft Specification or shall be agreed with the Rotorcraft Project Director. The recommended values of Table 3 for vertical landing, or Leaflet 304/3 for alighting on ships, shall be used as appropriate, unless it can be shown to the satisfaction of the Rotorcraft Project Director that more appropriate data is available. For run-on landings, consult Volume 1, Chapter 304 Table 2 and agree values with the Rotorcraft Project Director.

5 ENERGY ABSORPTION AND DISSIPATION

5.1 Those units of the undercarriage which can act together within the attitude envelope, shall be capable of reducing to zero the vertical velocity of descent in all the cases specified in this chapter. These are termed the main units.

5.2 Where statistical or arbitrary definition is used, the attitude of the rotorcraft may be assumed to be unchanged during the absorption stroke of the main units but changes shall be included where relevant in auxiliary and nose unit calculations and in all dynamic analysis of rotorcraft behaviour.

5.3 Neither the tyre nor the shock absorber may bottom at vertical velocities up to the design vertical velocity. Either the tyre or the shock absorber but not both may be allowed to bottom at vertical velocities between design V_V and 1.2 x design V_V provided that the ultimate reaction is not exceeded and the energy absorption requirements are met.

5.4 The design of all units shall be such that the energy absorbed by the shock absorber is dissipated as far as possible during the initial compression stroke. For main units, the energy dissipated shall be not less than 67% of the total vertical energy of descent on the 2/3rds airborne condition at design V_V. However for spring type undercarriages see Chapter 301 para 10.

5.5 The recoil of all moving parts shall be so damped that the applied forces do not produce unacceptable stresses.

6 UNDERCARRIAGE STIFFNESS

6.1 For undercarriage units having conventional shock absorbers, where the ratio of design allowable proof stress to design allowable ultimate stress for the material proposed for the primary structure of the undercarriage unit is less than 0.75, it shall be demonstrated by calculation or test that in all cases the proof requirements as stated in Chapter 200 are met at all velocities up to design V_V.

6.2 For undercarriage units where energy absorption is by friction damping at normal vertical velocities and by progressive permanent set at higher vertical velocities, the requirements should be discussed with the Rotorcraft Project Director.

7 ROTORCRAFT STATIC STRENGTH

7.1 The whole rotorcraft shall have a proof factor not less than 1.0 and an ultimate factor not less than 1.5 in all cases of paras 2, 3 and 4.

7.2 The whole rotorcraft shall have an ultimate factor not less than 1.0 in all the cases of paras 2, 3 and 4 and at the ultimate vertical velocity 1.2 x design V_V.

8 DROP TESTS

8.1 Drop tests shall be performed at vertical velocities increasing progressively to the design vertical velocity under a mass equal to the appropriate proportion of the rotorcraft mass supported by the unit or under a mass calculated to provide energy equivalent to that for which the unit is designed whichever is appropriate.

8.2 The drop test cases selected shall include all those with combinations of pitch and roll angle considered critical, those necessary to permit the determination of the forces which comprise a fatigue spectrum, those required to confirm the energy absorption characteristics of the unit, and those required to confirm the estimated design reactions.

8.3 In drop tests representing alighting with forward velocity, arrangements shall be made to represent spin-up on a surface giving as closely as is practical the required value of the coefficient of friction. Cases with wheel-brakes-on should also be included where relevant.

8.4 Instrumentation shall be provided to enable the relevant design parameters to be calculated from the measurements made during the energy absorption stroke and during rebound to show that the requirements of this Chapter have been met.

8.5 In all drop tests up to design V_V, the unit shall suffer no significant permanent set or distortion and measured reactions shall not exceed design reactions.

8.6 The tests should be done on a surface agreed with the Rotorcraft Project Director.

TABLE 1

PRELIMINARY STRESSING CASES FOR ALIGHTING ON VARIOUS SURFACES UNDER CONDITIONS OF PARA 2

	Still Flat Surfaces	**Moving Flat Surfaces**	**Still Rough Surfaces**	**Water**
	Runways, Pads, Platforms, Frozen Lakes	Ships decks or platforms with or without haul-down	Open fields, scrub, clearings. Hard or soft ground with or without snow or ice	The sea and other open water
Design V_V	2 m/sec (6.5 ft/sec)	2+(0.28xSS)m/sec (see Note)	2 m/sec (6.5 ft/sec)	3 m/sec (9.8 ft/sec)
Wheels Skids Skis Floats	Table 2A)) Table 2A) except cases 5 and 6	Table 2B)) Table 2B) except case 4	Table 2C)) Table 2C) except cases 2 and 3	NA NA NA Table 2B

Note: SS is the maximum sea state number in which operation is required. This formula gives a value which should be used unless otherwise specified in the Rotorcraft Specification. This is the relative velocity between the rotorcraft and the surface. See also Chapter 304 para 2.5 (v) regarding design V_V with haul-down.

TABLE 2A - PRELIMINARY STRESSING CASES - ALL UNITS - ALIGHTING ON STILL FLAT SURFACES

No	Case	Vertical Force	Drag Force	Side Force	Vertical Axle Travel %
1	Combined drag and side load	R	±0.25R	±0.25R	30
2	Combined drag and side load	R	±0.3 R	±0.3 R	30
3	Side load inboard	0.5R	0	0.4 R	10
4	Side load outboard	0.5R	0	0.3 R	10
5	High drag (spin-up and spring-back)	0.8R	±0.64 R	0	15
6	High drag (Wheel-brakes-on)	0.8R	±0.64 R	0	15
7	Rebound of unsprung parts	20 Mg	0	0	0

Notes: 1 For particular requirements relating to this table, see Chapter 304, para 2.

2 For general requirements relating to this table, see Chapter 301, para 6.

3 For general notes relating to this table, see Leaflet 304/1, para 3.7.

4 Case 1 applies to freely castoring wheels and the drag and side forces are to be applied relative to the axis of the wheel, not relative to the rotorcraft.

5 Case 2 applies to non-castoring wheels; to castoring wheels when locked in any practical position (including that normally used for deck landing); and to skis, skids, and floats.

6 For a unit on the centreline of the rotorcraft, case 3 will apply to both port and starboard and will override case 4.

7 Case 6 may override case 5. However, the drag force may be limited to that which the brakes can generate if this is less than 0.64R. Cases 5 and 6 are not identical as the point of application of the drag force is different, see Chapter 301, para 6.

TABLE 2B - PRELIMINARY STRESSING CASES - ALL UNITS ALIGHTING ON MOVING FLAT SURFACES

No	Case	Vertical Force	Drag Force	Side Force	Vertical Axle Travel %
1	Combined drag and side load	R	±0.56R	±0.56 R	30
2	Side load inboard	R	0	0.8 R	30
3	Side load outboard	0.8 R	0	0.64 R	15
4	High drag	0.8 R	±0.8 R	0	15
5	Rebound of unsprung parts	20 Mg	0	0	0

Notes: 1 For particular requirements relating to this table, see Chapter 304, para 2.

2 For general requirements relating to this table, see Chapter 301, para 6.

3 For general notes relating to this table, see Leaflet 304/1, para 3.7.

4 All cases apply to wheeled rotorcraft with brakes on and castoring locks engaged. In case 4, the drag force may be limited to that which the brakes can generate if this is less than O.8R.

5 For a unit on the centreline of the rotorcraft, case 2 will apply to both port and starboard and will override case 3.

TABLE 2C - PRELIMINARY STRESSING CASES - ALL UNITS - ALIGHTING ON STILL ROUGH SURFACES

No	Case	Vertical Force	Drag Force	Side Force	Vertical Axle Travel %
1	Combined drag and side load	R	±0.4 R	±0.4 R	30
2	High drag (Spin-up and spring-back)	0.8 R	±0.8 R	0	15
3	High Drag (Wheel-brakes-on)	0.8 R	±0.8 R	0	15
4	Main unit obstruction	R	±0.5 R	0	30
5	Main unit obstruction - inboard	R	0	0.5 R	30
6	Main unit obstruction - outboard	0.8 R	0	0.5 R	15
7	Nose/Tail unit obstruction	R	±0.7 R	0	30
8	Nose/Tail unit obstruction	R	0	±0.7 R	30
9	Rebound of unsprung parts	20 Mg	0	0	0

Notes:

1 For particular requirements relating to this table, see Chapter 304, para 2.

2 For general requirements relating to this table, see Chapter 301, para 6.

3 For general notes relating to this table, see Leaflet 304/1, para 3.7.

4 For a unit on the centreline of the rotorcraft, cases 5 and 6 will apply to both port and starboard.

5 Cases 2 and 3 are not identical as the point of application of the drag force is different (see Chapter 301, para 6). In case 3 the drag force may be limited to that which the brakes can generate if this is less than 0.8R.

6 For wheeled rotorcraft, all cases except cases 4 and 5 should be considered with wheel-brakes-on and castoring locks engaged and also with wheels free of both these constraints.

TABLE 3 - SECONDARY STRESSING CASES
DESIGN PARAMETERS FOR DYNAMIC AND STATISTICAL ANALYSES - VERTICAL AND NEAR VERTICAL ALIGHTING ON STILL FLAT OR ROUGH SURFACES

	Symbols		Design Values			Note, Ref or Data Source
Rotorcraft Parameters	**Standard Aeronautical Term**	**Alternatives in use**	**Mean**	**Standard Deviation**	**Arbitrary Extreme Values**	
Mass	m	M	$\frac{M_T + M_E}{2}$	$\frac{M_T + M_E}{3.464}$	M_T M_E	Note 7
Lateral Accel:	n_y		0.00	0.03	±0.15	Volume 1
Normal Accel:	n_z		0.85	0.05	0.67	
Horizontal forward speed	u	V_c	2kn (1 m/s)	1 kn (0.5 m/s)	-1 to 5 kn (-0.5 to 2.5 m/s)	Note 8
Drift (lateral speed)	v	V_L	0	1 kn (0.5 m/s)	5 kn (2.5 m/s)	Note 8
Vertical Velocity	w	V_v	0.50 m/s	0.40 m/s	2 m/s	
Horizontal wind speed			0	11 kn (5.7 m/s)	50 kn (26 m/s)	GSOR14 Note 6
Pitch attitude	θ	θ_P	AD	3°	±15°	AR 56
Roll attitude	Ø	θ_R	AD	3°	±15°	AR 56
Yaw attitude	ψ	θ_Y	0	NA	NA	
Shock absorber:-						
gas/air pressure	NA	-	AS	±5%	±20%	AR 56
oil volume	NA	-	AS	±2.5%	±10%	AR 56
Tyre Pressure	NA	p	AD	±5%	±20%	AR 56

Notes:
1 AS = As specified in the Rotorcraft Specification.
2 AD = As dictated by other parameters specified.
3 NA = Not applicable.
4 All velocities, accelerations, and angles are for the rotorcraft relative to the mean surface.
5 For other notes and references see Leaflet 304/1 para 3.8. For explanation of the arbitrary extreme values see Chapter 304 para 4.1 and Leaflet 304/1 para 2.2.
6 Standard deviation of horizontal wind speed: 8 kn (4 m/s) crosswind is applicable if landing into wind. See also Leaflet 304/1, para 3.8.4.
7 M_E is defined for this purpose as M_T with minimum fuel instead of full fuel. See Leaflet 304/1 para 3.2. The distribution between M_T and M_E is assumed to be rectangular unless other evidence is available. Analysis of available operational data supports this.
8 Horizontal forward speed and Drift. The values apply when the intention is to alight vertically as opposed to a deliberate run-on landing.

TABLE 4 - ALIGHTING ATTITUDES

Number	**Rotorcraft Attitude**	**Description**
1	Multi-Point	All main units in contact with the ground. Zero forward speed
2	Single-Point	Each main unit in contact with ground in turn at all critical combinations of pitch and roll angles from zero to the arbitrary extreme values given in Table 3.
3	Tail-down	The tail unit or aft main unit(s) in contact with ground with rotorcraft at that pitch attitude defined in note 3 below. Zero roll angle.
4	Nose-down	The nose unit or forward main unit(s) in contact with the ground at a 15° nose down attitude, and zero roll angle.
5	Sloping Ground (12° slope)	Each main unit in contact with ground sloping 12° at all critical combinations of pitch and roll angle having a combined probability of 1:1000.

Notes: 1 These attitude definitions are required in connection with the requirements of Chapter 304, paras 2.5 and 3.2 (iii).

2 Attitudes are defined by reference to the rotorcraft datum and determined with shock absorbers fully extended and tyres undeflected.

3 The tail-down attitude shall be the greater angle determined for the most aft cg position for each of the following methods of alighting:

(i) Maximum Forward Deceleration Manoeuvre:

From level flight, perform that manoeuvre which will result in maximum horizontal deceleration. The pitch angle shall be the maximum angle attained during this manoeuvre. See also Leaflet 304/1 para 3.3.

(ii) Hover Manoeuvre:

The pitch angle shall be that angle required to hover in ground effect at the basic design gross weight with a 60 knot tail wind.

(iii) Autorotative Manoeuvre:

See Chapter 301, para 12.

LEAFLET 304/1

DESIGN OF UNDERCARRIAGES - ALIGHTING TOUCHDOWN CONDITIONS

1 GENERAL

1.1 This Leaflet gives background information, data, definitions and suggested methods of compliance relevant to the requirements of Chapter 304.

1.2 The requirements are based on the assumption that static or quasi-static analysis of the preliminary stressing cases of para 2 of Chapter 304 will usually be used for initial design and that they may need to be supplemented, either at the design stage or later, by a few carefully defined secondary stressing cases derived by other procedures.

1.3 Validation of the design for Service use must, in certain critical cases, be based on the mandatory dynamic analyses as will the extension of Service limits beyond any of the values used for the preliminary stressing cases.

2 BASIS OF THE REQUIREMENTS

2.1 DYNAMIC ANALYSIS

2.1.1 This may apply both to the behaviour of the rotorcraft as a whole or to the analysis of forces acting on the undercarriage unit during alighting and after touchdown. It may also be used for the analysis of forces in the unit. When applied to the rotorcraft as a whole it will be used to define the conditions when alighting from conditions defined at an earlier point. When applied to the undercarriage unit it may start from one of the empirical cases or from a set of conditions defined either arbitrarily or statistically.

2.1.2 In most dynamic analyses it will be adequate to assume that the rotorcraft airframe is rigid. However, if the airframe is flexible enough for its response to modify the ground loads or the resulting forces in any parts of the structure, then a study of flexibility and response should be undertaken.

2.2 ARBITRARY DEFINITION

2.2.1 The forces acting on an undercarriage unit may be derived by analysis of the velocity and acceleration vectors starting from a set of parameters one or more of which are considered at a specified high level considered likely to arise in service. Recommended values are given in Table 3 of Chapter 304 for landing and in Leaflet 304/3 for alighting on board ship. These are intended to be used individually at this level. For example it may be considered desirable to examine a case based on 25° roll angle at various values of vertical and lateral velocity, but not to design to meet the case of all three at their arbitrary extreme values together or to combine this with the design vertical velocity.

2.3 STATISTICAL DEFINITION

2.3.1 A number of parameters may be selected and varied statistically to provide a set of combinations having a specified probability. A selection from this set will then be used to define secondary stressing cases. If the rotorcraft specification calls for a statistical definition, a total probability will be stated and the parameters which may be varied will be listed. The mean and standard deviation of each parameter is given in Table 3 of Chapter 304 for landing and in Leaflet 304/3 for alighting on board ship. These values should be used if specific data more directly applicable to the project is not available.

2.3.2 Where a statistical definition of any combination of parameters at a stated probability level is attempted, it is assumed that a computer will be programmed to search the possible combinations at that level and select those which provide the highest load in each structural member of the unit. However, an acceptable alternative in many cases will be to construct a number of cases. This is discussed in Leaflet 304/3.

2.3.3 Where the distribution of one parameter is dictated by another, the more significant parameter should be included in the analysis and the other held at a selected arbitrary value. For example, there is no need to consider variation of bank and yaw if variation of forward speed and drift velocity or crosswind provide greater forces.

2.4 EXAMPLES

2.4.1 A dynamic analysis of rotorcraft behaviour between main wheel touchdown and nose wheel touchdown may be used to define a nose wheel energy absorption and stressing case. This might start from an arbitrarily defined maximum or minimum pitch attitude and vertical velocity and might use empirical values of drag and side forces as necessary or follow on from the dynamic behaviour analysis of the autorotation case.

2.4.2 An arbitrarily defined set of bank and yaw angles may be necessary to define stressing cases where there are 4 main units or for an outrigger. For the same type of unit a statistical definition of bank and yaw angles with vertical, lateral and horizontal velocities could be used to define a number of cases having a specified probability.

2.4.3 An analysis of rotorcraft behaviour starting before touchdown may be useful in certain circumstances, particularly for rotorcraft having automatic landing devices, to show whether the flight handling characteristics of the rotorcraft are such as to give a high degree of confidence that the design vertical velocity and associated attitude angles are adequate for specified pilot inputs, wind shear effects, cross winds, gusts, or other specified disturbances.

3 DESIGN PARAMETERS

3.1 DESIGN VERTICAL VELOCITY

3.1.1 Vertical velocity of initial contact is the principal parameter which determines the energy to be absorbed and the landing reactions. No method of rational analysis of a typical landing can be devised to give a design value which is high enough to provide an adequate margin against emergencies and incidents. Equally it is not reasonable to frame a requirement to cover the worst of all possible combinations of all relevant parameters. The design value must therefore be set arbitrarily at some intermediate value. See Table 1 of Chapter 304 for values which are mandatory unless otherwise specified.

3 .2 DESIGN MASS

3.2.1 All requirements are stated on the assumption that they will be applicable at design take-off mass M_T. If a lower mass will give a critical case it should also be considered. Such information as is available suggests that generally landing mass is evenly distributed between M_T and M_E where M_E is the minimum mass at which take-off is permitted. If the designer considers that meeting the alighting requirements at M_T is inappropriate for his rotorcraft, even allowing for the demands of training practices in service, then he should consult the Rotorcraft Project Director.

3.3 ANGLE LIMITS

3.3.1 Where the design geometry of the rotorcraft is such that any of the specified angles cannot be achieved before some other limit is reached (e.g., the tail hits the ground at a pitch angle less than that derived from Table 3 of Chapter 304) the value to be used will have to be agreed with the Rotorcraft Project Director and a bumper will need to be provided on that part of the rotorcraft which touches first.

3.4 DESIGN ENERGY

3.4.1 A shock absorber is normally designed to absorb a proportion of the kinetic energy and to dissipate most of this energy. A tyre and a spring undercarriage unit on the other hand will dissipate only a small proportion of the energy absorbed and will return most of it in rebound. The requirements stated apply to the whole of the unit in each case and provide the rebound requirements which must be substantiated by analysis of test results.

3.5 DESIGN REACTIONS R_A, R_N, R_M, and R_T

3.5.1 These reactions will eventually be established or confirmed by drop tests but estimates are required for initial design purposes. These estimates may be obtained by dynamic or quasi-static calculations or from empirical formulae. In some cases the object of the estimates will be to determine the reaction when the shock absorber stroke and the wheel size are directed by other considerations (available space in particular), in other cases the object will be to establish the stroke required when the reaction is dictated by the strength of the back-up structure.

3.5.2 In general R_A can be determined only by dynamic analysis. In some cases there may be an overriding structural limitation and the dynamic analysis will only be used to check that performance is satisfactory. In any case no simple formulae can be given. Statistical definition may be useful in some cases.

3.5.3 For a rotorcraft having two main and one nose unit, where a two-point touchdown is expected to be normal, the main unit reactions may be estimated from:

$$R_M = \frac{1}{\eta_M S_M} \cdot \frac{M_T g}{2} \cdot \frac{1}{1+1_m^2/K^2} \cdot \left(\frac{V_v^2}{2g} + \frac{S_M}{3} \right)$$

and the nose unit reactions may be estimated by dynamic analysis of rotorcraft behaviour or from:

$$R_N = \frac{M_T g}{\eta_N S_N} \cdot \frac{1}{1+1_n^2/K^2} \cdot \left(\frac{V_v^2}{2g} + \frac{S_M}{3} \right)$$

3.5.4 For a rotorcraft having two main units and a nose or tail unit, where a three-point touch down is expected to be normal and a two-point touch down improbable, the three-point reactions may be estimated from:

$$R_M = \frac{1}{\eta_M S_M} \cdot \frac{M_T g}{2} \left(\frac{V_v^2}{2g} + \frac{S_M}{3} \right) \left(\frac{1_n - 0.4h}{1_m + 1_n} \right)$$

$$R_N = \frac{1}{\eta_N S_N} \cdot M_T g \left(\frac{V_v^2}{2g} + \frac{S_M}{3} \right) \left(\frac{1_m + 0.4h}{1_m + 1_n} \right)$$

Where a Nose Unit is fitted

$$R_M = \frac{1}{\eta_M S_M} \cdot \frac{M_T g}{2} \left(\frac{V_v^2}{2g} + \frac{S_M}{3} \right) \left(\frac{1_t + 0.4h}{1_m + 1_t} \right)$$

$$R_T = \frac{1}{\eta_T S_T} \cdot M_T g \left(\frac{V_v^2}{2g} + \frac{S_T}{3} \right) \left(\frac{1_m - 0.4h}{1_m + 1_t} \right)$$

Where a Tail Unit is fitted

3.5.5 Where both two-point and three-point are within the design attitude envelope, then the value of R_M and R_N used for design and stressing should be the greater of the two given above.

3.5.6 In the above formulae, η is the assumed overall efficiency of energy absorption for the shock absorber plus tyre acting together at design V_V. For conventional tyres and efficient shock absorbers, a value of 75% may be used, but where a conservative estimate is necessary, 65% should be assumed.

3.5.7 Where the rotorcraft pitch attitude control is used to decelerate the rotorcraft, the effect on unit reactions must be considered and this can usually be done only by dynamic analysis. On some rotorcraft the nose/tail wheel may bounce and the reaction will be reapplied several times before damping reduces it significantly. On others, the nose wheel may be held off until the additional drag has been applied. In either case therefore, the effects should be treated as additive unless it can be shown that this is unlikely to be the case.

3.6 OTHER DESIGN PARAMETERS

3.6.1 See Chapter 301 para 6 for detail stressing requirements applicable to all cases.

3.6.2 The values of design parameters not mentioned above are given in Table 3 of Chapter 304. Some parameters (e.g., tyre pressure), may be taken at their nominal value for design calculations and varied only during drop tests.

3.7 NOTES ON TABLE 2 OF CHAPTER 304

3.7.1 For each unit the value of R is equal to the maximum value of R_A, R_N, R_T, or R_M as appropriate. See para 3.5.1.

3.7.2 In any of the cases, shock absorber closure may be determined by dynamic analysis if preferred.

3.7.3 Tyre closure appropriate to the unfactored design reaction may be assumed.

3.7.4 In the combined drag and side load cases, the value of the drag force coefficient must be maintained at the required value until spin-up is complete, but may be relaxed to zero thereafter.

3.7.5 For a unit on the centre line of the rotorcraft, the "side-load-inboard" case will apply in both directions and override the "side-load-outboard" case.

3.7.6 In the spin-up cases, the value of the vertical force coefficient and of the drag force coefficient may be determined by dynamic analysis, starting from the design vertical velocity on a surface having a coefficient of friction of 0.8, if preferred.

3.7.7 The difference between the two high drag cases is that in the Spin-up and Spring-back case, the force is applied at the hub; and in the wheel-brakes - on case, the force is applied at the ground.

3.7.8 The cases given in Table 2A of Chapter 304 cover all landings ashore including landing after take-off from a ship with the wheels braked and locked in the castored position.

3.7.9 When landing with forward speed on rough ground, the units will be subjected to a number of cycles of vertical drag and side loads generated by the roughness. This will affect fatigue spectra.

3.8 NOTES ON TABLE 3 OF CHAPTER 304

3.8.1 n_Z given in the table is the ratio of aerodynamic lift to rotorcraft mass at the time of touchdown. The value given may be assumed to apply throughout the energy absorption stroke of the main units unless the analysis includes representation of lift reduction as the vertical velocity decays.

3.8.2 All statistical distributions are assumed normal except that of M_L which is rectangular between the limits M_T and M_E (see however Leaflet 304/3).

3.8.3 Irrespective of the attitude of the rotorcraft, vertical velocity is always taken as normal to the horizontal and closing speed parallel to it.

3.8.4 Cross wind data is based on standard meteorological information collected at a height of 10 m. For design purposes this may be interpolated to the relevant height according to the law:

$$V_1 / V_2 = \left(h_1 / h_2 \right)^{0.15}$$

4 DIRECTIONAL AND RESONANT FORCES

4.1 The preliminary stressing cases of Chapter 304 provide for the drift/yaw conditions at main-wheel touchdown, during the spin-up phase, and during the main energy absorption stroke.

4.2 When considering alighting in crosswind, an analysis of directional forces may be necessary. The lateral forces generated and the lateral energy to be absorbed will be derived from the landing speed and crosswind with appropriate combinations of yaw and bank angle.

4.3 One major problem which must be avoided occurs when the perturbations caused by the main rotor, interact with the characteristics of the undercarriage to produce ground resonance. Motions of the rotor blades in the lag plane, and the associated shift in the centre of gravity together with the motion of the rotorcraft as a whole on its undercarriage, may create a condition which rapidly becomes catastrophic. The effect of an uneven surface on this problem must also be considered (see Chapter 305).

5 BACKGROUND TO STRESSING CASES AND SHOCK ABSORBER DESIGN

5.1 The parameters affecting touchdown reactions are the rotorcraft landing mass, moments of inertia, the instantaneous acceleration vector, the instantaneous velocity vector, the attitude vector, rotorcraft geometry, structural stiffness, aerodynamic forces acting on the rotorcraft, the ground surface, the shock absorber damping and spring stiffness, the moment of inertia of the wheel, tyre and brake assemblies and tyre stiffness. All are significant in determining the forces in the unit, but unfortunately the most significant parameters sometimes cannot be determined in advance.

5.2 The parameters of the shock absorber which should be considered include the shock absorber spring curve pertaining to the dynamic conditions under consideration, the damping characteristics of the shock absorber, the effects of friction in the bearings of the shock absorber and, where appropriate, lateral flexibility and mass distribution.

5.3 Reduction of rebound of the whole rotorcraft after the first touchdown of the main landing gears is important. Hence it is necessary to maximise the energy dissipated during the compression stroke. In the 2/3rds airborne condition at design V_V the total energy dissipated by the suspension system is required to be 67% of the total.

5.4 Drag forces on the wheels occur due to spin-up during initial touchdown, but are assumed to be negligible thereafter on hard smooth level surfaces until the brakes are applied.

5.5 The spring-back forces are to be assumed to be equal to the spin-up drag forces unless an analysis of the elasticity of the undercarriage unit structure shows a lower value to be expected in all cases. The tyre and structural damping forces should be included where necessary. A simplified analysis such as that given in ANC-2 may be used subject to the Rotorcraft Project Director's agreement. On cantilever units, a dynamic analysis of these effects should be done to check whether a resonant condition could be set up. On units with a high angle of rake, spring back forces may exceed spin-up forces and a dynamic analysis is therefore recommended.

5.6 Side forces should be consistent with the angle between the horizontal velocity vector at the wheel and the plane of the wheel and the appropriate side force characteristics of the tyres.

5.7 Calculations will normally be based on the expected characteristics of new tyres correctly inflated. Consideration should be given to the effects of growth, incorrect inflation, and tyre wear if likely to be significant.

6 CONDITIONS AT INITIAL TOUCHDOWN OF FIRST MAIN UNIT

6.1 These conditions provide the basis for the strength, stiffness and energy absorption requirements for the main units. The requirements apply at the design vertical velocity. The requirements assume that all landings are made on hard smooth level surfaces.

6.2 Statistical definition may also be used at this stage to assess the overall probability to be associated with a particular design case, but an analysis in which all parameters are varied is not considered desirable or practical as a basis for design. Another use of statistical analysis may be to define the loading spectrum in a particular member for a fatigue analysis or for re-design for production or for an up-rated version of the rotorcraft.

6.3 Arbitrary definition will be required for these conditions only where the primary stressing cases are considered to be inadequate.

7 CONDITIONS AT INITIAL TOUCHDOWN OF OTHER MAIN UNITS

7.1 Generally the requirements apply to the first main unit to touchdown, but assume that the second main unit and the nose unit in a three unit design will, in some cases, touch simultaneously and in others soon after the first. Where relevant, the conditions applicable to the second and subsequent main units can be derived from those applicable to the first unit by an analysis of rotorcraft behaviour similar to those described below for the first auxiliary unit. Such an analysis is necessary, firstly for rotorcraft with main units abreast of each other, where rolling velocities and accelerations at touchdown have a significant effect and secondly, for rotorcraft with main units in line, where the range of touchdown attitudes can include those in which pitching velocities and accelerations make a significant contribution. Rotorcraft with 4 main units will need both.

8 CONDITIONS AT INITIAL TOUCHDOWN OF AUXILIARY UNITS

8.1 The conditions at initial touchdown of each auxiliary unit cannot be defined in the same ways as were used at initial touchdown of each main unit. Dynamic analysis of rotorcraft behaviour between main unit and auxiliary unit touchdown is both more practical and more essential than it is prior to main unit touchdown particularly if it is possible to define relevant pilot input programmes through the controls and the brakes. This is particularly applicable to nose units in three unit rotorcraft which normally land two-point and where the pilot can hold the nose-unit off the ground for some time. The same form of analysis may alternatively be used to determine the limits which have to be imposed on pilot action to avoid over-stressing a nose unit which has already been designed.

8.2 Particular attention should be given to the requirement to consider the most adverse combination of mass and cg position. In some rotorcraft, higher forces than those generated in the normal cases can be generated in a nose or tail unit following main unit touchdown at an aft cg position associated with a lower landing mass.

8.3 Dynamic analyses of rotorcraft behaviour should start from the conditions when the first wheel touches down and should include consideration of spin-up, spring-back, and crosswind effects. They should explore the behaviour of the rotorcraft up to the completion of the energy absorption stroke of the auxiliary unit(s). Flexibility of structure may be considered if the rotorcraft is not stiff enough for a rigid body analysis to be considered sufficiently accurate. The extent to which the forces on the unit are affected by pilot action, after first main unit touchdown, through the controls or the brakes, must be considered. The effect of pitch and roll inertia of the rotorcraft must be included where appropriate.

9 TESTS - NOTES ON DROP TEST REQUIREMENTS

9.1 OBJECTIVES

The objectives of the Drop Test Programme are:

(i) to demonstrate the energy-absorbing capability of a landing gear without exceeding the design reactions at design V_V and without failure of the gear at 1.2 x design V_V,

(ii) to establish the parameters necessary to validate a theoretical model of the unit or any part of it and to confirm the validity of design assumptions used in performance analysis or stressing, and

(iii) to establish rebound characteristics of the complete unit and to show that the design requirements have been met.

9.2 ENERGY REQUIREMENTS

9.2.1 For main units, the energy input should be the total rotorcraft landing energy divided proportionately between the units. For auxiliary units, the energy input should be the energy appropriate to a unit load corresponding to a typical condition. Alternatively, for both main and auxiliary units, the energy input may be that energy obtained from a rational method of predicting the energy absorbed when landing considering the effects of external forces due to the surface and the rotorcraft aerodynamics including any pilot actions, appropriately divided between all the units. The energy input is often an equivalent energy using non-airborne conditions instead of two thirds airborne conditions.

9.3 TEST SIMULATION LIMITATIONS

9.3.1 The whole of the undercarriage and back-up structure need not be included in the drop test if it can be shown that adequate correlation between design conditions and drop test results can be obtained by calculation. However all relevant parts necessary for accurate assessment of energy absorption, dissipation of design reactions, and rebound characteristics must be included.

9.3.2 It is recognised that, at most drop test velocities, the exact simulation of actual touchdown conditions is to some extent difficult. In particular, if a free-fall drop test is done, the rebound will be incorrect for an airborne or partially airborne condition. In addition, the difficulties of correcting drop test results if both pitch and roll angles are included at the same time are appreciable. Nevertheless, in some units, roll angle can be more important than pitch angle and must therefore be considered for inclusion where relevant.

9.3.3 A constant friction coefficient can be achieved by the use of an inclined drop surface, but mechanical constraints often arise. A friction coefficient corresponding to wheel spin-up depends on the type of surface used, the tyre parameters and the spinning velocity. Pre-spinning of the wheel also creates a difference between the test condition where the wheel is spinning down and the actual landing where the wheel is spinning up.

9.4 DROP TEST PROGRAMME

9.4.1 The test programme should include a number of separate conditions:

(i) Design cases. These should use an energy input corresponding to the rotorcraft design take-off mass and should use velocities up to the design vertical velocity. The actual cases selected should be those which are critical for the design of the unit and its attachments.

(ii) Ultimate cases. The aim of these tests should be to obtain data which will enable the designer to predict the vertical velocity at which failure will occur, unless this can be predicted from the validated mathematical models at mean touchdown attitude and at any other attitude considered potentially critical. Vertical velocities above design V_V may be used and some permanent deformation may therefore occur.

(iii) Additional non-design cases. Variations of such parameters as tyre pressure, shock absorber inflation pressure, shock absorber filling and operating temperature, and mixing of new and old tyres on multi-wheel units can occur during service. The effect of operating at other than design conditions should be included in the test programme if practical. In particular the variations of shock absorber pressure caused by variations of ambient operating temperature within the limits of Chapter 101 should be considered. It may be possible however to do only a limited amount of additional testing to confirm an analysis which could then be used for further work.

9.5 TEST MEASUREMENTS

9.5.1 Sufficient measurements should be made to determine the particular parameters required:

(i) Energy absorption and dissipation. It is necessary to measure vertical velocity at impact, drop mass travel, rebound velocity and height, and simulated lift forces where relevant.

(ii) External forces. The ground forces in the vertical, draft, and side planes should be directly measured or derived from other measurements.

(iii) Alighting gear parameters. The forces along and normal to the shock absorber centreline together with the displacements of the shock absorber and tyre and, if possible, the internal pressures in the shock absorber, are all required in order to validate a theoretical model of the shock absorber.

(iv) Airframe attachment forces. It may be considered desirable to measure forces in three mutually perpendicular axes at each of the attachment points if those forces are to form part of the test programme. It should be appreciated that such measurements, or the results obtained for such measurements, apply only to the test inputs made with the gear in the test rig and are not necessarily representative of the rotorcraft installation. Special care will have to be taken if the test rig is to be representative of the rotorcraft structure. Alternatively, a theoretical representation of the test set up could be used to obtain rotorcraft attachment loads from the results obtained on a drop test. In any case it may be more accurate to calculate the three axis forces at the attachments from measurements made at other points on the undercarriage.

LEAFLET 304/2

DESIGN OF UNDERCARRIAGES - ALIGHTING
DESIGN OF OLEO-PNEUMATIC SHOCK ABSORBERS FOR CORRECT FUNCTIONING

1 INTRODUCTION

1.1 The design requirements imply that undercarriage shock absorbers should provide the specified performance when the undercarriage is in the correct position for landing and the rotorcraft is about to land.

1.2 On some rotorcraft, the attitude of the shock absorber unit when in the retracted position may be such that the gas and the oil of an oleo-pneumatic unit may become mixed. If this is so, the oil and gas may not resume their correct relative positions during the short time interval between initiation of undercarriage lowering and the actual landing. The loss of shock absorbing capacity can be considerable and may cause greatly increased loads in the undercarriage and consequent damage to both undercarriage and airframe.

1.3 This Leaflet gives general recommendations intended to prevent the occurrence of this trouble.

2 RECOMMENDATIONS

2.1 It is recommended that undercarriages employing oleo-pneumatic shock absorber units should be designed so that mixing of the gas and oil is prevented under all conditions (e.g., by a separator piston).

2.2 Proposals to use a shock absorber unit with unseparated gas and oil should be discussed with the Rotorcraft Project Director. Ground tests should be made to determine:

(i) to what extent the gas and oil become displaced in the most adverse circumstances of rotorcraft attitude, duration of flight and temperature lag effects in cold climates, and

(ii) the time needed, after the undercarriage has locked down, for the gas and oil to sufficiently regain their normal positions to meet the specified shock absorber performance.

The acceptability of the recovery time so demonstrated should be discussed and agreed with the Rotorcraft Project Director and should be consistent with the lowering requirements of Chapter 306.

2.3 An inert gas such as nitrogen will normally be used, but the unit should operate on air as an alternative when the inert gas is not available. Any special precautions or limitations to be observed when using air in lieu of inert gas should be discussed with the Rotorcraft Project Director.

LEAFLET 304/3

DESIGN OF UNDERCARRIAGES - ALIGHTING
STATISTICAL INFORMATION FOR ALIGHTING ON SHIPS

1 GENERAL

1.1 This Leaflet gives statistical data and methods of analysis to supplement the requirement of Chapter 304 and in particular para 4.1 (i) and (ii).

1.2 The values of parameters given in Table 3 of Chapter 304 are for the rotorcraft relative to the ground. The values of parameters given in this Leaflet are in two forms. In para 2 and Tables 1 to 4 inclusive, ship parameters have been derived from the final report on NST 6100 by geometrical summation at all headings to give an overall σ_R for Sea States (SS) 5, 6 and 7, then by interpolation to give SS 3 and 4 and, by extrapolation to give SS 8. The rotorcraft parameters of para 3 and Fig. 1 are derived from the results of ships trials and include typical ship motion.

1.3 Values of parameters not given in this Leaflet may be taken from Chapter 304, Table 3.

1.4 Methods of statistical analysis and some notes on the use of the Rayleigh distribution for the representation of trials results are also given below.

2 SHIP MOTION PARAMETERS

2.1 Two operational conditions are represented:

(i) Continuous operation in the Sea State number given.

(ii) General operation in all Sea States up to the Sea State number given using the frequency of occurrence of each Sea State for the North Atlantic as a weighting factor.

2.2 Values of Parameters are given for two ship configurations:

(i) With roll stabilisers operating.

(ii) Without roll stabilisers operating.

2.3 Each parameter given contains elements of all contributing motions. For example V_Z includes contributions from pitch and heave.

2.4 The values of parameters are given in Tables 1 to 4. These apply to all ships and may be used in lieu of specific data where this is not available.

3 ROTORCRAFT PARAMETERS

3.1 Films made during trials with Lynx rotorcraft have been analysed to provide Vertical Velocity data according to the Sea State in which they were recorded. The data are given in Fig. 1. The means and standard deviations have been used to derive curves having a probability of 1:1000 and 1:10000 and these values in turn have been used to deduce curves of σ_R for equivalent Rayleigh distributions by the methods of paras 5.1 and 5.2. The recommended design values of Chapter 304 Table 1 are also shown.

3.2 Data for one particular trial is available for three parameters V_V, V_L and θ_R. This has been analysed by Westland Helicopters Ltd., using a computer programme which calculates the Rayleigh distribution giving the best fit and also by the methods of paras 5.1 and 5.2 below. The results of these analyses and corresponding recommended arbitrary extreme values are given in the following table.

	σR			**Arbitrary**
	WHL	**Para 5.1**	**Para 5.2 (N=10000)**	**Extreme Value (1:10000 Approx)**
V_V(m/s)	1.22	1.26	1.05	5.0 m/s
V_L(m/s)	0.36	0.43	0.40	1.7 m/s
θ_R(deg)	3.39	3.57	3.31	15°

4 METHODS OF STATISTICAL ANALYSIS

4.1 GENERAL PROCEDURE:

(i) Select the parameters which affect the event under consideration.

(ii) Decide which ones will be held at specified values and which will be varied.

(iii) Decide the total probability to be used.

4.2 PARTICULAR PROCEDURES:

(i) A number of parameters which are of particular interest may be considered at specific probability levels (e.g. 1:100 or 1:1000) while others, which affect the event under consideration but are of less interest, are held at mean value (1:2).

or:

(ii) A particular parameter may be held at an arbitrary extreme value (P_r = 1:10000) while a small number of other parameters are considered at specific probability levels (e.g., 1:10, 1:50, or 1:100).

4.3 TOTAL PROBABILITY

4.3.1 A total probability of 10^{-9} is recommended for all multi-variate analyses where 6 or more parameters are considered. For exercises of the type described in para 4.2, the total probability should be of the order of 10^{-n} where n is 3 greater than the number of parameters included.

5 REPRESENTATION OF DATA BY THE RAYLEIGH DISTRIBUTION

5.1 RMS values from trials data can often be represented by a Rayleigh distribution of the form:

$$f(x) = (x / \sigma_R^2) \exp(-x^2 / 2\sigma_R^2)$$

by direct substitution of $RMS / \sqrt{2}$ for σ_R.

5.2 Trials data not given in RMS values may be represented by a Rayleigh distribution derived by the process described below:

(i) From the plotted curve representing the data, pick a pair of values of the parameter and the probability of exceedance Pr = 1/N.

(ii) Calculate $k = (2 \log_e N)^{1/2}$ and σ_R = parameter value/k.
See also Fig. 2 where values of k have been plotted.

Repeat this for several pairs of values. If the calculated value of σ_R is reasonably consistent over a range of probabilities then the Rayleigh distribution with σ_R equal to the calculated value may be said to fit the data over that range.

5.3 Trials data for which the mean and standard deviation have been calculated on the assumption that the data is normally distributed, may be represented approximately by a Rayleigh distribution by either of the above methods. If the best overall fit is required, use the method of para 5.1 above remembering that, for this distribution:

$$RMS = (x^2 + \sigma_N^2)^{1/2}$$

as illustrated in Example 1 below.

If however it is important to match the probability at the tail, then use the method of para 5.2 above as illustrated in Example 2 below.

5.4 It is unlikely that any mathematical distribution will fit a given set of data points exactly over the whole range of the data. The Rayleigh Distribution often gives a better overall fit than the Normal Gaussian Distribution and may therefore be more appropriate for extrapolation. To represent the whole of the trials data, a value of σ_R should be selected which is the average value of a large number of calculations over the whole range of the data points. For extrapolation of a tail, the value of σ_R should be selected to represent the tail values of the data points. Both these methods are illustrated in the example below (Example 3). In the first case, the probability of a given value will be greater than given by the normal distribution. In the second case it will be of the same order if not exactly the same. Before deciding whether a Normal or a Rayleigh distribution should be used for extrapolation it may be necessary to compare the data with each and do a Chi-squared test to determine the best fit.

Example 1: (Method of para 5.1)

From Fig.1 for SS 6.

$\bar{x}$ = 1.45 m/s

σ_N = 0.50 m/s

RMS = $(1.45^2 + 0.50^2)^{1/2}$

σ_R = $1.534 / \sqrt{2}$

= 1.08 m/s

Example 2: (Method of para 5-2)

Using the above data, the vertical velocity for a probability of 1:1000 on the Normal distribution is 1.45+3.09 x 0.5 m/s = 2.995 m/s. σ_R for a Rayleigh distribution having the same vertical velocity at the same probability will be:

σ_R = 2.995/k

where k from Fig. 2 is found to be 3.70

Then: σ_R = 2.995/ 3.70
= 0.81 m/s

Example 3: (Method of paras 5.2 and 5.4)

The following values of probability and vertical velocity are taken from Lynx trials and represent the average of all trials. In this case the values are in ft/sec.

V_V	1	2	3	4	5	6	7	8	9	10	11
P_r	.99	.90	.65	.36	.15	.06	.02	.0065	.0016	00035	.00006
N	1.01	1.11	1.54	2.78	6.67	16.67	50.00	153.8	625.0	2857	16667
k	0.14	0.45	0.93	1.43	1.95	2.37	2.80	3.17	3.59	3.99	4.41
σ_R	7.05	4.36	3.23	2.80	2.57	2.53	2.50	2.52	2.51	2.50	2.49

The mean value of σ_R is 2.96 and this will give a good general fit representing the whole of the data. A computer programme sampling the curve at smaller intervals, or sampling the original trials results, and fitting a curve by the method of least squares, might give a more accurate value. For extrapolation of the tail a value of σ_R of 2.50 or 2.51 might be more appropriate.

SHIP MOTION

TABLE 1 - VALUES OF σ_R AT THE ALIGHTING PLATFORM

Ships operating in the North Atlantic, at all headings continuously in the sea state given, with roll stabilisation.

Sea State No.	3	4	5	6	7	8
V_z(m/s)	0.20	0.40	0.75	1.05	1.30	1.65
G_z(g)	0.02	0.05	0.07	0.11	0.12	0.14
V_y(m/s)	0.18	0.30	0.45	0.60	0.80	1.00
G_y(g)	0.03	0.04	0.05	0.06	0.07	0.07
θ_x(deg)	0.30	0.50	0.85	1.40	1.95	2.20
θ_y(deg)	0.60	0.80	0.10	2.10	2.60	3.00

Ships operating in the North Atlantic at all headings in all sea states up to and including the sea state given with roll stabilisation.

V_z(m/s)	0.33	0.54	0.68	0.73	0.75	0.75
G_z(g)	0.02	0.05	0.06	0.07	0.07	0.07
V_y(m/s)	0.30	0.46	0.54	0.57	0.58	0.58
G_y(g)	0.04	0.06	0.07	0.07	0.07	0.07
θ_x(deg)	0.40	0.66	0.82	0.88	0.91	0.92
θ_y(deg)	0.88	1.31	1.52	1.61	1.65	1.66

SHIP MOTION

TABLE 2 - VALUES OF σ_R AT THE ALIGHTING PLATFORM

Ships operating in the North Atlantic, at all headings continuously in the sea state given, without roll stabilisation.

Sea State No.	3	4	5	6	7	8
V_z(m/s)	0.20	0.40	0.75	1.30	1.80	2.30
G_z(g)	0.03	0.04	0.06	0.13	0.17	0.20
V_y(m/s)	0.18	0.30	0.45	0.68	1.05	1.40
G_y(g)	0.01	0.02	0.03	0.04	0.05	0.06
θ_x(deg)	0.20	0.50	0.60	0.95	1.40	1.90
θ_y(deg)	0.60	0.90	1.40	2.50	3.50	4.00

Ships operating in the North Atlantic at all headings in all sea states up to and including the sea state given without roll stabilisation.

V_z(m/s)	0.33	0.54	0.68	0.74	0.77	0.77
G_z(g)	0.04	0.06	0.07	0.08	0.08	0.08
V_y(m/s)	0.30	0.46	0.54	0.56	0.58	0.59
G_y(g)	0.02	0.03	0.03	0.03	0.04	0.04
θ_x(deg)	0.33	0.60	0.71	0.75	0.77	0.77
θ_y(deg)	0.99	1.46	1.73	1.84	1.89	1.90

SHIP MOTION

TABLE 3 -σ_R AT WORST HEADING AT THE ALIGHTING PLATFORM

With Roll Stabilisers

Sea State No.	3	4	5	6	7	8
V_z(m/s)	0.50	0.75	1.00	1.40	1.85	2.35
G_z(g)	0.04	0.08	0.12	0.16	0.20	0.2
V_y(m/s)	0.30	0.40	0.70	0.90	1.10	1.20
G_y(g)	0.03	0.05	0.07	0.08	0.10	0.11
θ_x(deg)	0.50	0.80	1.20	2.00	2.70	3.30
θ_y(deg)	0.70	1.20	1.90	2.70	2.10	3.90

With NO Roll Stabilisers

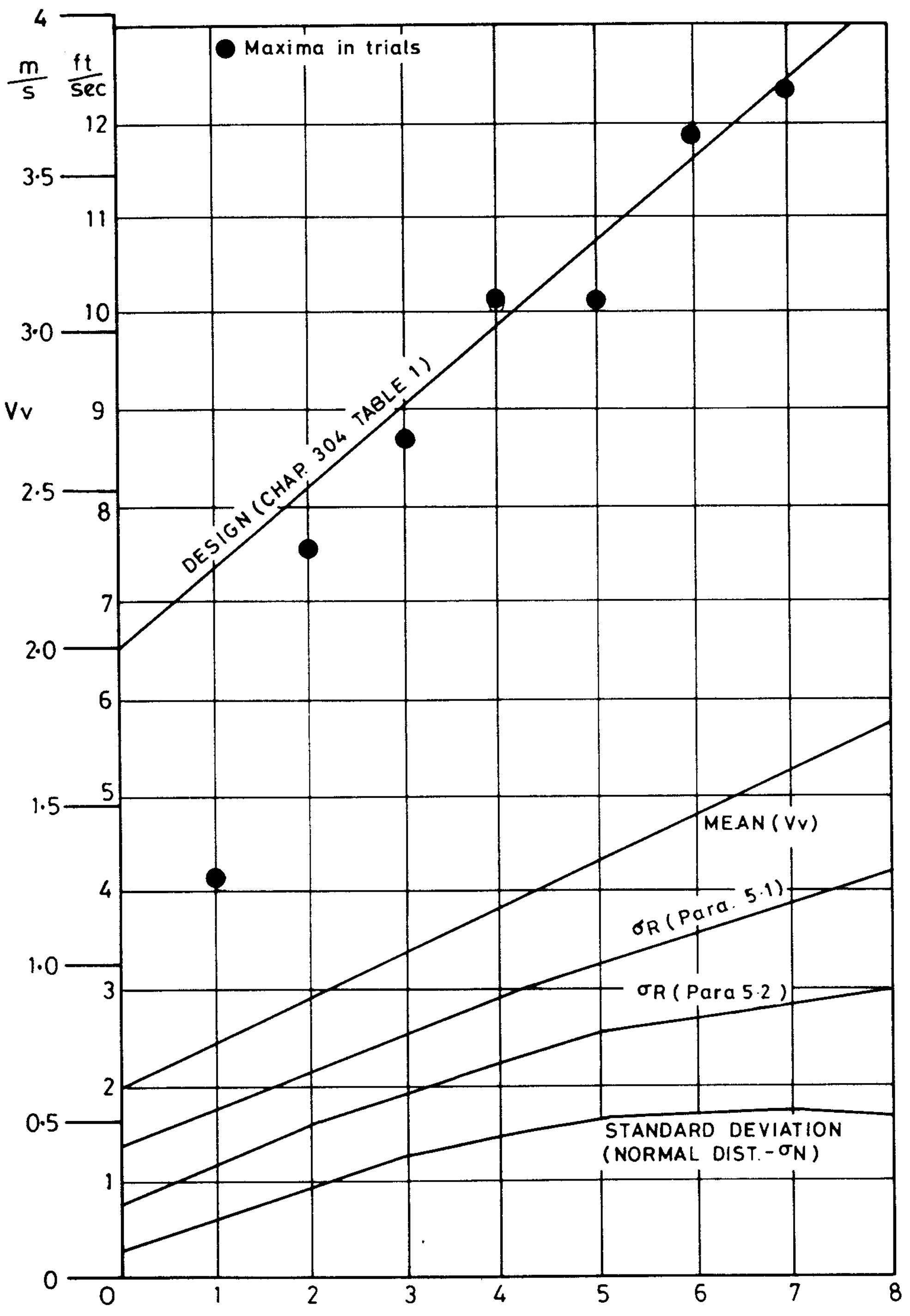
V_z(m/s)	0.40	0.70	1.20	2.00	2.70	3.70
G_z(g)	0.03	0.07	0.10	0.17	0.30	0.40
V_y(m/s)	0.30	0.50	0.80	1.20	1.60	1.85
G_y(g)	0.01	0.02	0.03	0.06	0.07	0.08
θ_x(deg)	0.20	0.40	0.70	1.20	2.00	2.80
θ_y(deg)	1.20	2.20	3.90	5.60	7.10	8.00

SHIP MOTION

TABLE 4 - ARBITRARY EXTREME VALUES (Pr = 1:10000)

(4.29 x largest σ_R in Tables 1 & 2)

Sea State No.	3	4	5	6	7	8
V_Z(m/s)	1.42	2.32	3.22	5.58	7.72	9.87
G_Z(g)	0.17	0.26	0.30	0.56	0.73	0.86
V_y(m/s)	1.29	1.97	2.32	2.92	4.50	6.00
G_y(g)	0.17	0.26	0.30	0.30	0.30	0.30
θ_x(deg)	1.72	2.83	3.65	6.00	8.37	9.44
θ_y(deg)	4.25	6.26	7.42	10.72	15.01	17.16



Sea State No.

FIG.1 RELATIVE VERTICAL VELOCITY - LYNX ROTORCRAFT TRIALS

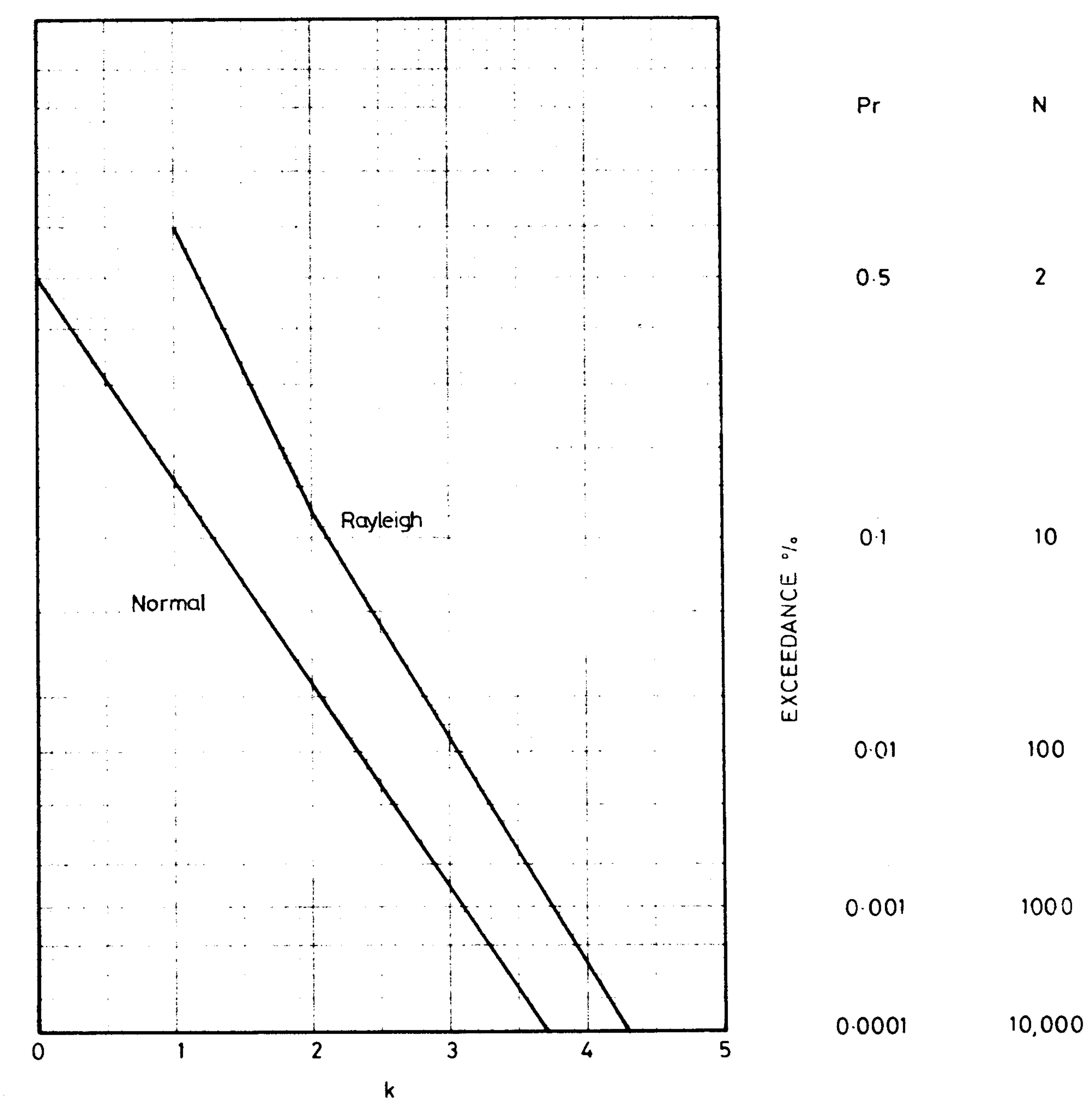


FIG.2 VALUES OF k FOR NORMAL AND RAYLEIGH DISTRIBUTIONS

CHAPTER 305

DESIGN OF UNDERCARRIAGES-

VERTICAL AND NEAR-VERTICAL OPERATION ON ROUGH SURFACES

1 INTRODUCTION

1.1 The requirements of Chapters 302, 303 and 304 are based on the assumption that the surface is smooth and hard. The extent to which the requirements of these Chapters provide adequate strength, stiffness, and handling qualities for operation on rough surfaces is uncertain.

1.2 This Chapter provides arbitrary standards against which the design may be assessed, with the aim of ensuring satisfactory vertical or near vertical operation from uneven hard surfaces.

2 SPEED VECTORS

2.1 The horizontal speed vector shall be any speed within a half-ellipse having a forward speed of 2.5 m/s and a lateral speed of 1 m/s.

2.2 The vertical speed vector shall be any vertical speed between zero and the maximum vertical lift-off speed (see also para 4.2 below).

3 OBSTACLES

3.1 The step is defined as a rectangular section (see Fig.1) but a 10mm radius of the corners may be presumed. It is of infinite length.

3.2 The bump is defined as a half buried sphere, (see Fig.2).

3.3 The log is defined as a half buried cylinder of infinite length and the same section as the bump.

3.4 The size of the obstacles is determined by the class of surface. See Leaflet 300/1 and the Rotorcraft Specification.

4 ASSESSMENT

4.1 The requirements of Chapters 302, 303 and 304 shall be considered in relation to any combination of the speed vectors of para 2 and each of the obstacles of para 3.

4.2 In para 3.2 (iii) of Chapter 304 the value of vertical velocity shall be taken as 0.8 times design V_V.

4.3 The results of this assessment shall be discussed with the Rotorcraft Project Director.

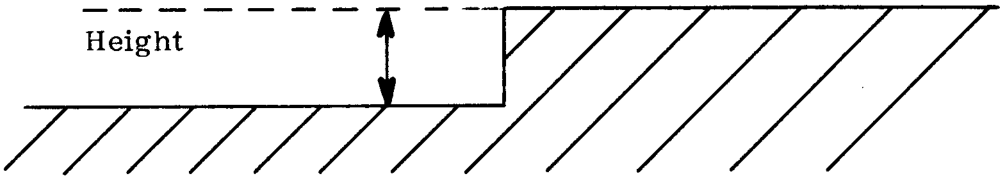


FIG.1 90° STEP

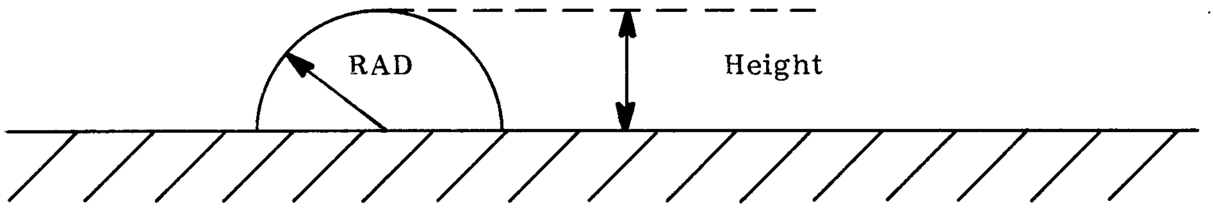


FIG.2 HEMISPHERICAL BUMP OR LOG SECTION

TABLE OF STEPS AND BUMPS

Class	Step Height	Bump Height (mm)	Surface (see Leaflet 300/1)
A	25	30	Normal
B	40	45	"
C	60	75	Other
D	100	120	"

CHAPTER 306

DESIGN OF UNDERCARRIAGES - RETRACTION AND LOWERING

1 INTRODUCTION

1.1 For rotorcraft with retractable undercarriage units, this chapter states operational, design, strength and test requirements for Retraction and Lowering, for Locking, and for Undercarriage Doors.

2 OPERATIONAL REQUIREMENTS

2.1 Retraction shall be possible at all forward speeds up to maximum speed in horizontal flight in the most adverse combination of attitude, acceleration, and environmental conditions. When acceleration exceeds typical values, retraction may stop but shall continue after the acceleration has reduced.

2.2 Lowering shall be possible at all forward speeds up to maximum speed on horizontal flight in the most adverse combination of attitude, acceleration, and environmental conditions. Lowering time shall be as short as possible (see Leaflet 306/1 para 2) and a time in excess of 5 seconds under any possible set of flight conditions shall be discussed and agreed with the Rotorcraft Project Director.

2.3 Level 1 handling qualities (see Chapter 302 para 1.4) shall be obtained throughout a declared undercarriage-down flight envelope. They shall not be degraded more than one level during retraction and lowering under the conditions of paras 2.1 and 2.2 above.

3 DESIGN REQUIREMENTS

3.1 The retraction and lowering mechanisms shall function satisfactorily under all operating conditions including operations from surfaces which are snow-covered, salt-treated, sandy, muddy or covered with mown grass. Particular attention shall be given to the prevention of any jamming or corrosion which might follow operation from such surfaces.

3.2 Gyroscopic effects which can occur, and braking torque which can be applied, during retraction following take-off with forward speed shall not cause serious vibration nor permanent deformation of the alighting gear nor impair its correct functioning.

3.3 A standby method of lowering the undercarriage in emergency shall be provided and no single failure in the normal system shall prevent the undercarriage being lowered by this standby system under the conditions of para 2.2 above. When the appropriate actions for lowering the undercarriage by the standby system have been correctly performed, the operation of lowering shall proceed until it is completed irrespective of any actions, correct or otherwise, which might have been made with the normal selector control. However, if the final actuator is a component of established reliability, this need not necessarily be duplicated if the Rotorcraft Project Director agrees. If not obvious, indication of operation of this standby system is required.

3.4 The design speed for emergency lowering shall be discussed and agreed with the Rotorcraft Project Director. If this is lower than the design speed for normal lowering, the parts of the system common to both normal and emergency lowering shall be designed to the higher speed.

3.5 Steering and/or castoring units shall centralise before retraction is complete, and retraction shall be achieved without fouling from any practical steered or castored position. Such units shall remain centralised whilst retracted.

3.6 It shall be impossible to retract the undercarriage by means of the normal selector control whilst the undercarriage is taking any part of the weight of the rotorcraft.

3.7 If the undercarriage is selected up, and this is prevented by any automatic interlock (e.g., castoring or weight-on-ground), the subsequent operation of that interlock shall not activate the dormant selection.

4 LOCKING

4.1 Unless the operating mechanism is irreversible, the main and auxiliary undercarriage units shall be held in both the safe landing and retracted positions under all appropriate flight conditions by positive locks maintained in engagement by mechanical means.

4.2 Hydraulic system pressure may be used to hold the undercarriage units in the retracted position provided the units are also prevented from moving from their retracted positions by mechanical means which must not be released by failure of hydraulic pressure (see also para 2.2 above).

4.3 All main and auxiliary undercarriage units shall be provided with a ground locking device in addition to the normal locking system to prevent unintentional retraction when the rotorcraft is being serviced or repaired and when normal locking or systems are ineffective. If take-off is made with a ground locking device left in place, neither attempted retraction of the undercarriage nor landing shall result in damage to the structure. The ground locking device when engaged shall not prevent the rotorcraft being manoeuvred on the ground within the range of steering or castoring angles specified.

4.4 Undercarriage-down lock mechanisms should be designed and positioned so that any failure to achieve a positive locked indication can be examined, and so that ground locks may be inserted if necessary, whilst the rotorcraft remains in the hover.

5 UNDERCARRIAGE DOORS

5.1 Under all relevant conditions the undercarriage doors if fitted should remain closed. Any gaping which may occur shall not cause unacceptable stresses, buffet or vibration.

5.2 When the undercarriage doors and/or door locks are also used as a means of retaining the undercarriage in the retracted position, the doors and door locks shall be designed to support any loads imposed including those arising from failure of the undercarriage jacks. If the wheels touch the doors during any flight manoeuvre, no damage to the tyres, wheels or undercarriage unit, or inability to lower, shall result.

5.3 Where any undercarriage door covers a servicing point, a door safety lock shall be provided to prevent inadvertent closure.

6 STRENGTH

6.1 Under normal operating conditions defined in para 2 above, the undercarriage and doors and their operating mechanisms shall have proof and ultimate factors not less than 1.5 and 2.0 respectively (see also para 3.1 above).

7 TESTS

7.1 Every new design of retractable undercarriage unit shall, prior to first in-flight retraction, undergo a schedule of tests to demonstrate compliance with the requirements of this Chapter. Guidance on the necessary tests is given in Leaflet 306/2 and the schedule shall be discussed and agreed with the Rotorcraft Project Director.

7.2 In all such tests the following shall apply:

(i) The mass of each wheel shall be increased by 15% to simulate mud and/or ice accretion (see Leaflet 306/1 para 3).

(ii) Power supplies shall be identical with those used in the appropriate condition in service.

LEAFLET 306/1

DESIGN OF UNDERCARRIAGES - RETRACTION AND LOWERING

1 INTRODUCTION

1.1 This Leaflet amplifies the requirements for undercarriage retraction and lowering.

2 RETRACTION AND LOWERING TIMES

2.1 The time from selection of the relevant control position to locking of the undercarriage and doors in the required position is required to be as short as possible consistent with adequate handling qualities. Times considered to be acceptable maxima under all likely operating conditions are:

Retraction time	..	..	10 seconds
Lowering time	..	..	5 seconds

3 NORMAL OPERATION

3.1 The power provided for retraction and lowering should be adequate to meet the operating requirements including an addition of 15% to the total weight of the wheels, brakes, and tyres to allow for weight growth and the accretion of mud and ice.

4 EMERGENCY OPERATION

4.1 Control of the emergency system for lowering the undercarriage should be as simple as possible. Emergency systems which require resetting after such operation are acceptable. If compressed gas or a combustible cartridge is used to supply emergency power, all cartridges, and compressed gas bottles, together with pipes and components through which the products of combustion have to pass, should be located so that there is a minimum risk of contamination in the cockpit or crew stations both when the system has been operated and if it has been damaged and so that there is a minimum risk of damage to normal and emergency systems due to any one cause. The operation of the emergency lowering system should not contaminate any other main system.

LEAFLET 306/2

DESIGN OF UNDERCARRIAGES - RETRACTION AND LOWERING

RETRACTION AND LOWERING TESTS

1 When rotorcraft are designed with retractable undercarriage units, a schedule of retraction and lowering tests must be agreed (see Chapter 306 para 7) which verify design assumptions and calculations. These tests should aim to demonstrate that the unit will correctly retract and, more importantly lower under:

(i) the most adverse flight conditions required for undercarriage operation, and

(ii) the most adverse tolerances of system adjustment and wear.

The problems of door gaping and vibration from residual wheel spin during retraction, both encountered on aeroplanes, are unlikely to require testing on conventional rotorcraft.

2 Features to be considered when establishing the test schedule should include the following:

(i) Loads resulting from the most adverse flight conditions required for undercarriage operation can cause:

(a) an increase in the time taken to complete the operation,

(b) distortion of the overall undercarriage system geometry,

(c) deformation of the rotorcraft structure supporting and surrounding the undercarriage components.

(ii) Large transient flight loads applied during the retraction or lowering sequence can cause the operation to stop momentarily.

(iii) The most adverse tolerances of system adjustment can occur in service either by chance or through component wear.

(iv) The undercarriage units should be hard against their locks when fully retracted or lowered and the locks should fulfill their locking function under the appropriate most adverse flight conditions.

(v) Tests of any emergency lowering system should parallel those of the normal lowering system.

3 Repeatability and consistency should be demonstrated in the tests. Any test sequence involving the normal retraction and lowering system should be repeated 25 times, and any test sequence involving an emergency lowering system should be repeated 5 times.

CHAPTER 307

CRASH LANDING, DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

1. GENERAL

1.1 The main objectives of Design for Crash Landing, Ditching and Precautionary Alighting on water are:

a) to preserve life, and to minimise injury to occupants, during impact,

b) to enhance survivability after impact,

c) to minimise damage to the rotorcraft and its equipment and to minimise repair costs. See Leaflet 307/2 para 3.

1.2 The structure, seats, harnesses (see Chapters 106 and 111) and all relevant systems shall be designed, so far as is reasonably practical, to attain the objectives stated in para 1.1 above by the provision of energy absorbing devices and adequate, but not excessive, structural strength. See Leaflet 307/2 para 1.

1.3 The Rotorcraft Design Authority must exercise discretion in choosing the extent to which each system contributes to the attainment of these objectives. Individual systems which contribute to occupant protection shall not be considered in isolation but shall be co-ordinated and interfaced to achieve a completely integrated and efficient crashworthy design.

1.4 Where velocity change or acceleration requirements are given in this chapter the values quoted are from the 95th percentile survivable crash, as defined in Tables 2 to 8 of this chapter. Where survivability less than 95% is required by the rotorcraft Specification, corresponding values from Tables 1, 2 and 3 of Leaflet 307/2 may be used in lieu according to the survivability.

1.5 Reference to the Rotorcraft Specification will be found in the following paragraphs of Chapter 307: 1.4, 3.1.1, 3.1.2, 3.1.3, 3.2.3, 3.3.1, 3.3.2, 4.7.1(iii), 4.7.3 and of the following: Leaflet 307/2 paras 1.9, 4.1, 16.1, and Leaflet 307/3 para 2.4.

2. DESIGN FOR CRASH LANDING AND DITCHING

2.1 EVACUATION (See also Chapter 102)

2.1.1 It shall be possible to evacuate the rotorcraft through half the available exits in 30 seconds or within flotation time whichever is less and whatever the attitude of the rotorcraft in the water.

2.1.2 Provision shall be made for rapid evacuation after either crash landing or ditching. In deciding the sizes and number of emergency exits, consideration shall be given to:

a) stability of the rotorcraft on the water after ditching or a precautionary alighting on water,

b) availability of the exits for use after the emergency,

c) the effect of roll-over following a crash landing,

d) the flotation time after ditching.

2.1.3 Means shall be provided so that each emergency exit can be opened from both inside and outside the rotorcraft. Jettison devices shall be similarly accessible. All emergency exists shall be designed to be squeezed outwards by structural deformation without jamming and should also be jettisonable. Release mechanisms shall not jam as a result of structural distortion.

2.1.4 Rotorcraft systems needed during evacuation shall be designed, as far as is reasonably practical, to function for the length of time it is estimated evacuation will take.

2.1.5 Seats, stretchers and their support structure shall be designed to minimise interference with evacuation. Harnesses shall be designed not to form obstructive loops when not in use. See also Chapter 111.

2.2 PROTECTIVE SHELL

2.2.1 The structure of the crew and passenger compartments shall be designed, as far as is reasonably practicable, to prevent inward buckling (see leaflet 307/2 para 11) and to provide a protective shell for seated occupants. Deformation shall be controlled, as far as reasonably practical, to provide maximum survivability and possibility of post-crash exit. The possibility of local penetration of the structure, by external parts of the rotorcraft which might cause injury to the occupants, shall be considered so that injury to the occupants can be avoided.

2.2.2 The interior of the protective shell shall be designed to minimise injury to any parts of the body of a seated and restrained occupant which flail during a crash landing or ditching. Particular attention shall be paid to the movement of the head. See Leaflet 307/2 paras 12-15.

2.2.3 The design shall ensure that the occupant and/or any part of his body cannot become trapped between the structure and any impacting surface following failure of doors, windows, canopies or hatches.

2.2.4 The movement, during a crash landing or ditching, of those parts of the rotorcraft in the vicinity of the crew or other occupants (particularly the massive items of para 4.11 below) shall not compromise the survivability of the occupants, as far as is reasonably practical, and the relevant parts of the structure shall be designed to absorb energy by controlled deformation. See Leaflet 307/2 paras 1.2 and 1.3.

2.2.5 For vertical impact not less than 42ft/sec (12.8m/s), on a rigid, hard, horizontal surface structure shall prevent ceiling collapse and maintain a survivable volume for the occupants. Neither seats nor stretchers shall be suspended from the ceiling unless this requirement is met. See also Leaflet 307/3 para 1.9 and para 1.4 above.

2.3 STRENGTH AND ENERGY ABSORPTION

2.3.1 All strength requirements based on the rotorcraft mass shall be met at Design Take-Off Mass M_T fully airborne unless otherwise stated ie Rotor Lift = M_T.

2.3.2 When the undercarriage or the relevant impact point (see leaflet 307/1) is subject to any velocity change, or combination of velocity changes in a half ellipsoid, given in Table 2, from zero to the values stated, within the impulse duration given, the accelerations at the base of the spine of any occupant shall be limited to the levels given in Table 1.

2.3.3 All parts affected by the requirements of this chapter shall have an ultimate factor not less than 1.0 under the forces arising from:

a) the static cases of paragraphs 4 and 5 below.

b) the dynamic cases of Table 2.

2.3.4 In the assessment of forces resulting from the specified velocity changes the distance over which the change occurs, and the efficiency of the energy absorption system, shall be considered. A factor for the efficiency of the energy absorption system between the impact point and part, determined by the Rotorcraft Design Authority preferably as the result of tests, shall be included. Excessive static strength shall however be avoided. See Leaflet 307/2 para 1.2.

2.3.5 Local increase in acceleration caused by whiplash effects or dynamic overshoot (see leaflet 307/2 para 1.3) shall be considered when assessing structural integrity and occupant survival.

2.3.6 See Leaflet 307/3 for recommendations on design for support of seats and stretchers and for data on the masses of aircrew and troops.

2.4 MATERIALS

2.4.1 Use of composite materials in the design shall allow for the directional nature of energy absorption in the materials when meeting multidirectional loading requirements. The design shall also provide for the flammability, toxicity, smoke and thermal decomposition of the materials used and the effects these may have post crash.

2.4.2 Exterior surfaces and structure which could be exposed to contact with the ground in a crash shall be constructed, as far as is reasonably practical, of materials which resist sparking as a consequence of abrasion.

2.4.3 Materials for seats and stretchers shall be selected on the basis of the best strength-to-weight ratio with sufficient ductility to prevent brittle failure. If any doubt exists about probable peak loads of critical members the elongation shall be at least 10%. Materials shall be considered for their resistance to burning and, in their finished condition, shall satisfy the requirements of Chapter 712.

2.5 CONTROLS

2.5.1 All controls shall be designed as far as practical so that they present no hazard to the crew during a crash. See Leaflet 307/2 para 15.

3. DESIGN FOR DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

3.1 GENERAL

3.1.1 The requirements of this section apply if Design for Ditching or for Precautionary Alighting or water is required by the Rotorcraft Specification as a result of a Service Requirement for operation over water for any significant time.

3.1.2 The Design Authority having designed the Rotorcraft to the Crashworthiness, Strength, Stiffness, Energy Absorption, and other requirements of this Chapter, shall estimate the resulting velocity and acceleration ellipsoids for Ditching, and for Precautionary Alighting on water, and incorporate them in the Rotorcraft Specification. Note : Impact onto water involves different modes. (See also para 6.4 below). Consideration shall be given to the differences in structural failure mode. (See Leaflet 307/2 para 1.6 and Leaflet 307/0 Ref 13)

3.1.3 The Design Authority shall consider the following cases and establish, in each case, at the limits of wind speed, wave-slope, and Rotorcraft attitude stated in the Rotorcraft Specification (See Table 1 and Fig 9 below), the limitations for each of the following cases :

CASE 1. Autorotation approaches, as close as practical to the critical points on the speed/height boundaries established as in Chapter 301 para 12.

CASE 2. Horizontal and near-horizontal speeds as close as practical to normal cruising speed as is reasonably possible.

CASE 3. Vertical and near-vertical alighting, at vertical velocities at or near the mean vertical velocity, in the combined (rotorcraft plus ship) vertical velocity spectrum derived from data for sea states up to the maximum given in the Rotorcraft Specification.

3.1.4 The design shall be such as to allow controlled fuel jettison following a precautionary alighting on water.

3.1.5 The design shall be such that, following a precautionary alighting on water, water ingress would not preclude a subsequent take-off within a reasonable period of time, by:

a) degrading essential rotorcraft systems.

b) increasing the mass above that which would allow take-off.

c) altering the C of G to a point where subsequent controlled flight would not be possible.

3.2 FLOTATION

3.2.1 Adequate buoyancy and stability shall be available to enable the rotorcraft to remain afloat and upright with rotors stopped in Sea State 3, and if possible up to Sea State 6.

3.2.2 Where flotation capability depends on the operation of emergency flotation aids, the requirements shall be met in any single failure case applied to the actuation system or with any single compartment of the flotation gear inoperative.

3.2.3 Flotation aids and associated equipment shall be designed and tested to show satisfactory operation following ditching in the most adverse environmental conditions called for in the Rotorcraft Specification and in the velocity and acceleration ellipsoids determined as the result of the analysis of para 3.1.2.

3.2.4 The use of a sea anchor may be assumed, if provided, during the deployment of liferafts and the escape of the crew and occupants, but shall be assumed to make no contribution to buoyancy and stability.

3.3 ESCAPE

3.3.1 Doors and windows and their local structure shall be designed to withstand local water pressures arising from the circumstances in which contact with water is expected in a precautionary alighting, when ditching, or on subsequent overturning. Local water pressure shall not prevent operation of emergency exits. The Rotorcraft Design Authority shall suggest applicable cases for insertion in the Rotorcraft Specification. See para 2.1.1 above and Table 2 Case 7.

3.3.2 The rotorcraft shall be designed to enable the crew and other occupants to salvage any special equipment required by the Rotorcraft Specification, to deploy liferafts, and to enter them with the maximum probability of survival. The deployment of liferafts shall not be impeded by deformation of exits.

3.3.3 Where liferafts are normally released directly overboard, an external means shall be provided for their release, accessible to a survivor outside the rotorcraft, in addition to any internal provision for their release.

3.3.4 Provision shall be made for underwater escape and inversion of the rotorcraft shall not compromise its success. See also Chapter 102 para 6.3.

4. DESIGN FOR CRASH LANDING

4.1 GENERAL

4.1.1 In addition to the general requirements of Section 1 and para 2.3 of this Chapter the following particular requirements shall be met. See para 6.1 below.

4.1.2 The velocity requirements of this section (paras 4.2 to 4.15) are based on the 95th percentile. See para 1.4 above.

4.2 DESIGN FOR LONGITUDINAL IMPACT

4.2.1 CASE 1. The basic airframe shall be capable of impacting a hard rigid vertical barrier longitudinally at 20ft/sec (6.1m/s) without reducing the size of the protective shell of the pilot, co-pilot and any designated members of the crew to the extent that their evacuation of the Rotorcraft and survival after the crash is prevented or impeded. The engine(s), gear boxes, transmissions, rotor systems, (including the rotor blades on the assumption that they do not impact a vertical surface) and any items of equipment or cargo, which could cause injury or unacceptable damage, shall remain intact and in place in the Rotorcraft. Particular attention shall be paid to the likely deformation of the rotor support structure and any deformation likely in the control system which may cause the rotor blades to be driven into occupied areas. See also para 4.11.

4.2.2 CASE 2. The basic airframe shall be capable of impacting a hard rigid vertical barrier longitudinally at 40ft/sec (12.2m/s) without reducing the length of the passenger/troop compartment by more than 15% and without causing the pilot, co-pilot and designated members of the crew to suffer accelerations greater than those of Table 1. Any consequent inward buckling shall not be hazardous to occupants nor prevent their evacuation after the crash.

4.2.3 CASE 3. The forward undersurface of the Rotorcraft shall be designed to minimise earth scooping regardless of the consistency or density of the soil within reasonable limits. The nose section shall neither plough nor scoop earth when the forward 25% of the fuselage has a uniformly applied local upward force of $10M_T$ and a rearward force of $4M_T$. See also Leaflet 307/2 para 10.

4.3 DESIGN FOR VERTICAL IMPACT

4.3.1 CASE 1. The Rotorcraft as a whole shall be designed for 42ft/sec (12.8m/s) vertical velocity on a hard rigid horizontal surface at any attitude within the envelope of Fig 1. The height of the crew and passenger/troop compartments shall not be decreased by more than 15% and the occupants shall not experience accelerations which are greater than those of Table 1, cause serious injury, or impede emergency evacuation. The engine(s), transmission, rotor system (except possibly the rotor blades), and any items of equipment or cargo which could cause injury or unacceptable damage, shall remain intact and in place in the Rotorcraft.

4.3.2 CASE 2. Where the undercarriage is retractable the Rotorcraft as a whole shall be designed to withstand, with the undercarriage retracted and within the envelope of Fig 1, a vertical velocity of not less than 26ft/sec (7.9m/s) on a hard rigid horizontal surface without reducing the height of any occupied compartment by more than 15% and not exceeding the accelerations of Table 1 for any occupant.

4.3.3 CASE 3. The undercarriage shall be capable of decelerating the Rotorcraft from a vertical velocity of 20ft/sec (6.1m/s) on a hard rigid horizontal surface without allowing the fuselage to touch the surface. This requirement shall be met at any attitude within the envelope of Fig 1. The undercarriage should continue to absorb some energy even when the fuselage has contacted the ground. Failure of the undercarriage shall not increase danger to any occupants by penetration of the airframe or by rupturing a fuel tank or by damaging missiles, rockets, and ammunition. Plastic deformation of the undercarriage is acceptable but the remainder of the rotorcraft structure, except possibly the rotor blades, should remain airworthy.

4.3.4 See also Leaflet 307/2 paras 8 and 9.

4.4 DESIGN FOR LATERAL IMPACT

4.4.1 The Rotorcraft as a whole shall be designed for a lateral impact velocity of 30ft/sec (9.1m/s) normal to the longitudinal and vertical axes of the rotorcraft. The width of the crew and passenger/troop compartments shall not be decreased by more than 15% and the occupants shall not suffer accelerations greater than those given in Table 1 or cause serious injury, or impede evacuation.

4.4.2 The engine(s), transmission, rotor system (except possibly the rotor blades), and any items of equipment or cargo which could cause injury or unacceptable damage, shall remain intact and in place in the Rotorcraft.

4.5 COMBINED CASES

4.5.1 CASE 1. The Rotorcraft as a whole shall be designed undercarriage down to withstand impact, as in a low-level approach, fully airborne, into ploughed soil, at a flight path angle of 8^{o}, at 100ft/sec (30.5m/s). The volume of the passenger/troop compartment shall be reduced by not more than 15%. The design should if possible prevent earth scoop or roll-over in these conditions. See Leaflet 307/2 para 10.
The engine(s), transmission, rotor system (except possibly the rotor blades, and any items of equipment or cargo which could cause injury or unacceptable damage, shall remain intact and in place in the Rotorcraft.

4.5.2 CASE 2. The Rotorcraft as a whole shall be designed undercarriage down to sustain a combined vertical/longitudinal impact on a hard rigid horizontal surface at a vertical velocity of 42ft/sec (12.8m/s) and a longitudinal velocity of 27ft/sec (8.2m/s) within the attitude envelope of Fig 1. The volume of the passenger/troop compartment shall be not be reduced by more than 15%.

4.6 ROLL-OVER

4.6.1 Cases 1, 2 and 3 below shall be met without permitting deformation of a magnitude that may cause injury to a seated and restrained occupant, or restrict their evacuation.

4.6.2 CASE 1. The fuselage roof is assumed to be buried in soil to a depth of 2" (5.1cm) over the forward 25% of the total length of the Rotorcraft excluding the blades. A load of 4 times the Design Take-Off Mass M_T shall be applied as in para 4.6.5 below.

4.6.3 CASE 2. The rotorcraft is loaded on its side in the same conditions as Case 1.

4.6.4 CASE 3. Where the Rotorcraft configuration is such that Case 1 cannot be applied then the Rotorcraft shall be assumed to be on the ground inverted in the most critical attitude for safety of the occupants. The following forces shall be applied individually:

a) $4M_T$ perpendicular to the ground or

b) $4M_T$ along the longitudinal axis parallel to the ground or

c) $2M_T$ along the lateral axis parallel to the ground.

4.6.5 In cases 1 and 2 above $4M_T$ load shall be applied, over the area defined, at any angle between vertical and horizontal in any direction, ie from perpendicular to the fuselage skin to parallel to it. All doors, hatches and windows take no load.

4.7 SEAT INSTALLATION - CREW AND SPECIFIED OCCUPANTS

4.7.1 The requirements of this section apply to the seats, restraint harness (see Chapter 111 para 1.1) attachments, and local support structure for:-

a) all members of the crew,

b) any occupant who might compromise the essential actions of the crew if his restraint failed,

c) any occupant designated by the Rotorcraft Specification.

4.7.2 The parts listed in para 4.7.1 shall meet the static strength requirements of para 2.3 in conjunction with Table 3. See Leaflet 307/3 para 2.

4.7.3 When determining the forces in the cases of para 4.7.2 the mass and dimensions of the crew and occupants, and the CG position assumed, shall be in accordance with the data given in Leaflet 307/3 Table 1 and Leaflet 105/3 for the percentiles given in the Rotorcraft Specification, and full allowance shall be made for the weight of the seat and any items of equipment, including armour, carried on the seat or by the crew. See also Chapter 113 and Leaflet 307/3.

4.8 SEAT INSTALLATION - OTHER OCCUPANTS

4.8.1 Seats, harnesses, attachments, and local support structure for occupants other than those covered by para 4.7.1, including those for passengers, troops, and paratroops, shall meet the static strength requirements of para 2.3 in conjunction with Table 4.

4.8.2 When determining the forces in the cases of para 4.8.1 the mass and dimensions of the occupants, and the CG position assumed, shall be in accordance with the data given in Leaflet 307/3 Table 2 and Leaflet 105/3 for the 5th and 95th percentiles, and full allowance shall be made for the weight of the seat and any items of equipment carried on the seat or by the occupant. See also Leaflet 307/3.

4.9 EQUIPMENT (See also Chapter 102) AND COMPONENTS OF SYSTEMS

4.9.1 The requirements of this section apply to all fixed and removable equipment and to components of systems. See para 5.1.2 below.

4.9.2 Wherever it can reasonably be foreseen that, if the equipment or component moved or broke free during a crash landing or ditching it would:-

a) cause injury to crew or other occupants,

b) cause a fire,

c) prevent the use of an emergency exit,

d) prevent the use of emergency equipment,

e) prevent access to items which might be needed by the crew after the crash,

then they shall be restrained (see Leaflet 307/2 para 6.1) to meet the static strength requirements of para 2.3 in conjunction with Table 5.

4.9.3 Emergency equipment and items which might be needed by the crew after the crash or ditching shall also meet the requirements of para 4.9.2.

4.9.4 Stowage space for unrestrained items or personal equipment shall also meet the requirements of para 4.9.2 and shall be designed so that items stowed in it cannot become a hazard to personnel in a survivable crash. See also Chapter 714 paras 2.5.3 and 2.5.4.

4.9.5 Where fixed or removable equipment or components of systems are located in such a manner that the requirements of para 4.9.2 and 4.9.3 do not apply, then the installation shall have the normal in-flight and ground load factors, the factors shown by trade-off studies (see Leaflet 307/2 para 5) or the ultimate factors of Chapter 714 whichever are the greater.

4.10 CARGO AND FREIGHT

4.10.1 Configuration requirements and normal strength requirements are stated in Chapter 714.

4.10.2 Where the cargo or freight might be a hazard to personnel (see para 4.9.1 above) energy absorption in the longitudinal forward and the lateral directions shall be provided. Static strength and energy absorbers in these directions shall be provided. Static strength and energy absorbers in these directions shall meet the requirements of para 2.3 above in conjunction with Table 6.

4.10.3 Static strength in other directions shall meet the requirements for equipment of para 4.9.2 or 4.9.5 as appropriate.

4.10.4 See Leaflet 307/4 for Operational Recommendations.

4.11 MOUNTING OF MASSIVE PARTS

4.11.1 All massive parts, including but not limited to; Engines, Gear Boxes, Transmissions, Rotor Heads, Blades, Empennage, Fuel Tanks or Auxiliary Power Units, shall meet normal flight cases stated in DEF STAN 00-970 Volume 2 Part 2 and any special cases stated in other chapters (eg Chapter 705 for transmissions). Their attachments and support structure shall be designed to prevent their displacement in a manner hazardous to occupants.

4.11.2 Where these parts can become a hazard in a crash landing or ditching they shall also comply with the static strength requirements of para 2.3 in conjunction with Table 7. See also para 4.2.1 above.

4.11.3 The rotor head and transmission shall not displace in a manner hazardous to occupants during:

a) Rollover as in Cases 1 and 2 of para 4.6 above.

b) The outer 10% of a main rotor blade striking a rigid hard cylinder of 20cm (8in) diameter.

4.12 STRETCHERS (LITTERS)

4.12.1 The normal requirements for stretchers (litters) are stated in Chapter 714.

4.12.2 Static strength for crash landing and ditching shall meet the static strength requirements of para 2.3 above in conjunction with Table 8 and a body mass of 114kg (250lb).

4.13 HAND GRIPS

4.13.1 Hand grips which may be used during a crash landing, together with their attachments, shall have an ultimate factor not less than 1.0 under a pull of 2000N (450 lbf) acting in any possible direction.

4.14 HARNESS ATTACHMENT

4.14.1 Where a harness is attached directly to structure allowance shall be made for seat movement caused by functioning of energy-absorbers.

4.15 EVACUATION

4.15.1 Emergency exists and those parts of the structure, the failure or excessive displacement of which, would prevent or impede evacuation, shall meet the requirements of para 2.3. Particular attention shall be given to manufacturing joints and transport joints.

4.15.2 Emergency lighting, escape identifications and markings shall be crashworthy to the requirements of para 4.9.2 or para 4.9.5 as appropriate.

5. DESIGN OF SYSTEMS

5.1 GENERAL

5.1.1 The rotorcraft and its systems shall be designed to minimise the probability of a fire developing at or soon after impact, whatever its attitude, by ensuring that the requirements of Chapter 712 are met. See also Chapters 702, 704 and 706.

5.1.2 All components of systems shall meet the requirements for equipment of para 4.9.2 or 4.9.5 as appropriate.

5.2 FUEL SYSTEM

5.2.1 The fuel system shall be designed, as far as is practical, to contain fuel during and after the crash in each of the cases of Table 2. See also Chapter 702 para 9.

5.2.2 Fuel tanks shall comply with the requirements of para 4.11, and where relevant of para 4.9 above, when 75% full.

6. VALIDATION OF DESIGN

6.1 DEMONSTRATION

6.1.1 Demonstration of compliance with the requirements of sections 1, 2, 3, 4 and 5 shall be by static and dynamic analysis, mathematical modelling and such tests as are practical. See Leaflet 307/2 para 2.1.

6.1.2 The need for a physical model and for full scale dynamic impact tests on the rotorcraft structure, in addition to static strength tests, shall be considered. See Leaflet 307/2 para 2.1.2.

6.1.3 Where complete system testing is not undertaken then mathematical analysis shall show that individual crashworthy elements of the system function together properly to achieve the desired overall level of crashworthiness.

6.1.4 Where design for ditching or precautionary alighting on water is required (section 3 above) the test programme shall include proposals for ditching model tests to show adequate buoyancy and stability on the water. See Leaflet 307/2 para 2.2.

6.1.5 The crew seats, energy absorbers, and relevant structure, together with the restraint system and other items defined in para 4.7.1, shall be sled-tested under dynamic loading conditions equivalent to those of Table 2 Cases 8, 9 and 10. The strength and stiffness of the attachments used shall be representative of the support structure. For details of dynamic tests on seats see References 8 and 9 of Leaflet 307/0 and STANAG 3950.

6.1.6 The results of the tests shall show compliance with the requirements of para 2.3 above.

6.1.7 Instrumentation suitable for making the measurements required by the above shall be provided.

7. RELIABILITY

7.1 Because of the emergency nature of the requirements of this chapter, particularly those applicable those applicable to seats, it is important that a high degree of reliability of all components affected is achieved. See Leaflet 307/2 para 17 and Chapter 307 para 1.3.

7.2 The Rotorcraft Design Authority shall take note of the requirements of DEF STAN 00-40 and 00-41 when compiling the test schedule for reliability tests.

TABLE 1

LIMITS OF HUMAN TOLERANCE (G)

Direction (See note 1)	Impulse Duration (ms) (plateau)									Max Jerk (G/sec)
	1000	500	200	100	60	40	20	10	1	(See note 2)
Forward	12	17	25	33	40	45	50	60	60	500
Aft	20	23	30	36	38	40	45	47	47	500
Lateral	(4)	(5)	(10)	12	(13)	(15)	(15)	(16)	(16)	500
Down	5	7	10	15	20	25	25	25	25	500
Up	5	7	10	12	13	17	17	18	19	100

NOTES ON TABLE 1:

1. Directions are those of the pilot's eyeball movement, during the impulse, relative to rotorcraft axes.

2. Jerk is also know as 'aggravation' and 'onset rate'.

3. Forward, aft, down and up values have been taken from Ref 1 of Leaflet 307/0.

4. For lateral accelerations the value 12 at 100ms has been taken from para 4.3.5 of USARTL-TR-79-22B. Other values have been derived by proportion using forward and aft values for comparison. These are bracketed.

5. The times given are the plateau durations. Total times will in each case be longer because of the time required for the initial jerk.

6. For design purposes acceleration factors given may be assumed to be whole body tolerance limits for all seated, fully-restrained occupants.

GENERAL NOTES ON TABLES 2 to 11:

1. Directions are given as in Note 1 above.

2. The figures given in Table 2 are taken from Ref 2 of Leaflet 307/0.

3. The figures given in Tables 3 to 8 are taken from Refs 2, 8 and 9 of Leaflet 307/0.

TABLE 2

DESIGN AND TEST CASES

CASE NO	PARTS AFFECTED	DIRECTION	VELOCITY CHANGE		PLATEAU DURATION
			m/s	ft/sec	ms
1	Cockpit	Longitudinal forward	6.1	20	100
2	Cabin	Longitudinal forward	12.1	40	100
3	All	Vertical down	12.8	42	50
4	All	Lateral	9.1	30	100
5	All	Vertical down and	12.8	42	50
		Longitudinal forward	8.2	27	100
6	All	Vertical down and	4.3	14	50
		Longitudinal forward	30.5	100	100
7	Airframe and Seats	Vertical down	7.9	26	50
8	All Seats	Vertical down,	13.0	42.6	50
		Longitudinal forward,	7.6	25.0	100
		and Lateral	2.6	8.7	100
9	All Seats	Longitudinal forward	13.2	43.3	100
		and Lateral	7.6	25.0	100
10	Crew Seats	Vertical down	12.8	42.0	50

Notes on Table 2:

Cases 1-6 Undercarriage down

Case 5. High angle approach, 50 ft/sec (15.2m/s) at 50^{o}.

Case 6. Low angle approach to ploughed field, 101 ft/sec (30.8m/s) at 8^{o}.

Case 7. Undercarriage up case. May also be used as guidance when considering ditching cases, if required. See paras 2.1.1 and 3.3.1

Case 8 & 9. Test cases 50ft/sec (15.2m/s) resolved.

Case 10. Test case for crew seats with 50% and 95% dummy.

Note: All case Impacts are against rigid hard surfaces. Except case 6.

TABLE 3

SEATS FOR CREW AND SPECIFIED OCCUPANTS
(Chapter 307 para 4.7 and Leaflet 307/3 para 2.1)

Direction	Acceleration Factor (G)	Body Mass lb	Body Mass kg	Deflection Limit in	Deflection Limit cm
Forward	35	250	114	2	5.1 Max
Aft	12	250	114	2	5.1 Max
Lateral	20	250	114	4	10.2 Max
Down (bottomed) *	48 (20-25) (15 preferred)	200	91	12	30.5 Min
Up	8	250	114	2	5.1 Max

* See Leaflet 307/2 para 7.1

TABLE 4

SEATS FOR OTHER OCCUPANTS
(Chapter 307 para 4.8 and Leaflet 307/3 para 3.1)

Direction		Acceleration Factor (G)	Body Mass lb	Body Mass kg	Deflection Limit in	Deflection Limit cm	
Forward (bottomed)*		35 (30-15)	242	100	-	-	-
Aft		12	242	110	-	-	-
Lateral	U&OR*	23 (23-10)	242	110	-	-	-
(bottomed)	A&CR*	20 (20-10)	242	110	-	-	-
Down (bottomed)		36 (14.5 ± 1)	200	91	12	30.5	Min
Up		8	242	110	-	-	-

U&OR = Utility and Observation Rotorcraft
A&CR = Attack and Cargo Rotorcraft
* See Leaflet 307/2 para 7.1

TABLE 5

HAZARDOUS EQUIPMENT AND COMPONENTS
(Chapter 307 paras 4.9, 4.10, 4.15 and 5)

Direction	Acceleration Factor (G)	Note
Forward	35	For acceleration factors applicable to Non-Hazardous Equipment and Components see Chapter 714
Aft	15	
Lateral	25	
Down	50	
Up	10	

TABLE 6

CARGO AND FREIGHT
(Chapter 307 para 4.10)

Direction	Restraint (G)	
Forward (bottomed) *	35 (16-6)	
Aft	As para 4.10.3	
Lateral (bottomed) *	25 (10-3)	BLANK
Down	As para 4.10.3	
Up	As para 4.10.3	

* See Leaflet 307/2 para 7.1

TABLE 7

MASSIVE ITEMS
(Chapter 307 para 4.11)

Direction	Individually (G)	Combined (G)			
Forward	20	20	10	10	) Longitudinal
		or	or	or	)
Aft	20	20	10	10	) Alternatives
Lateral	18	0	9	18	Alternating L&R
Down	20	10	20	10	) Vertical
		or	or	or	)
Up	10	5	10	5	) Alternatives

TABLE 8

STRETCHERS (LITTERS)
(Chapter 307 para 4.12)

Direction	Restraint (G)	
Forward	20	
Aft	12	
Lateral	20	BLANK
Down	25	
Up	8	

TABLE 9

SEA CONDITIONS

Sea State No	World Meteorological Organisation		Design Wave Slope	Design Wind Speed (kt)
	Description	Significant Wave Height (m)		
0	Calm (Glassy)	0	-	-
1	Calm (Ripples)	0 - 0.1	-	-
2	Smooth (Wavelet)	0.1 - 0.5	-	-
3	Slight	0.5 - 1.25	1:20	15
4	Moderate	1.25 - 2.5	1:16	20
5	Rough	2.5 - 4.0	1:12	25
6	Very Rough	4.0 - 6.0	1:10	30
7	High	6.0 - 9.0	-	-
8	Very High	9.0 - 14.0	-	-
9	Phenomenal	over 14	-	-

NOTES ON TABLE 9:

1. The Significant Wave Height (SWH) is defined as the average value of the height (vertical distance between trough and crest) of the largest 1/3 of waves present.

2. The Maximum Wave Height (MWH) is usually taken as 1.6 x Significant Wave Height (eg SWH of 6 metres give MWH of 9.6 metres).

3. The arbitrary values given in Columns 4 and 5 have been selected from North Atlantic data as those most suitable for use in design.

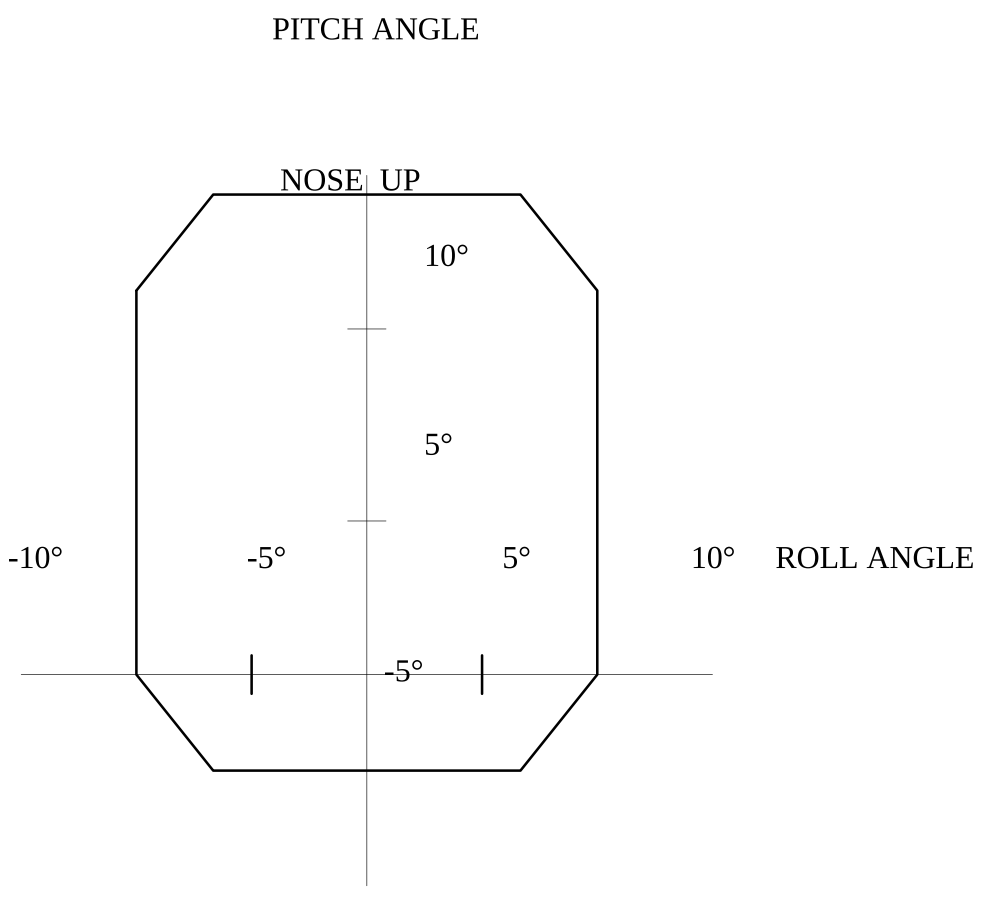


FIG 1 - PITCH/ROLL ENVELOPE

LEAFLET 307/0

CRASH LANDING, DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

REFERENCE PAGE

	Authors, References	Title(s) etc
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3.	Institute of Aviation Medicine (IAM) Report No 632, P Vyrnwy-Jones	A review of Army Air Corps helicopter accidents 1971-82.
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6.	Royal Aircraft Establishment RAE Report No TR 81120 M T Charlton (BR 82527)	Some statistical landing data derived from helicopter trials on ships at sea.
7.	Westland Helicopters Ltd Final Report Contract A26B/197 Research Paper RP 718 (1986) (BR 101287)	Helicopter structural crashworthiness research programme.
8.	American Helicopter Society and Georgia Institute of Technology (1986) BLDC 87/19283	A specialist meeting on Crashworthy Design of Rotorcraft.
9.	American Helicopter Society Fort Worth Texas MOD Ref 84/001455	41st Annual Forum Proceedings. May 1985

10.	Mil Spec Mil-S-58095A(AV) Jan 1986	Seat System: crash resistant, non-ejection, aircrew
11.	Mil Spec Mil-S-85510(AS) Nov 1981	Seats, helicopter cabin, Crashworthy
12.	STANAG 3950 AI (Edition 1) 25 Jan 1991	Helicopter crew seat design
13.	ASD-TR-76-30	Cargo and Spacecraft Restraint Criteria.
14.	ISVR Technical Report No 209 (Oct 1992)	Design Guide for the Ergonomic Aspects of Helicopter Crew Seating
15.	DYNAMIC SCIENCE 1500-71-43 Dept of Navy Office of Naval Research, Va USA.	A survey of Naval aircraft crash environment with emphasis on structural response. (Glancy JJ, Desjardines SP)

LEAFLET 307/1

CRASH LANDING, DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

DEFINITIONS

1. ALIGHTING

1.1 Descent and touchdown on any surface. Includes Normal, Heavy and Crash Landing, and both Precautionary Alighting on Water and Ditching.

2. NORMAL LANDING

2.1 Any landing within the design requirements of Chapter 304. No damage to the rotorcraft. No injury to the occupants.

3. HEAVY LANDING

3.1 Any landing above the design vertical velocity (V_V) of Chapter 304 in which control is retained. Some local yielding of structure. Possible collapse of the undercarriage above $1.2V_V$. No incapacitating injury to the crew. After a heavy landing the rotorcraft may be able in emergency to take-of, fly back to base, and alight, with negligible risk of further injury to the occupants, depending on the state of the undercarriage.

4. CRASH LANDING

4.1 Any landing involving high vertical impact velocities arising from irrecoverable loss of control or impact from any other direction.

5. PRECAUTIONARY ALIGHTING ON WATER

5.1 A pre-meditated controlled alighting on the sea or any inland water. Take-off may be expected in certain circumstances.

6. DITCHING

6.1 Alighting on the sea or any inland water following loss of power to the rotors or loss of control and with the intention of abandoning the rotorcraft.

6.2 Note that, in the civil requirements field, ditching is defined as a controlled alighting on water. This is equivalent to the military precautionary alighting on water.

7. IMPACT POINT

7.1 The relevant impact point of Chapter 307 paras 2.3, will be the point on the rotorcraft which will strike the surface or obstacle first in the attitude giving the velocity vector selected for analysis and test.

8. SURVIVABLE CRASH

8.1 A crash in which the range of impact conditions transmitted to the occupants (including jerk and the magnitude, direction, and duration of declarative forces), does not exceed the limits of human tolerance in which the structure supporting and surrounding personnel remains sufficiently intact, during and after impact, to permit survival.

9. EFFECTIVE MASS

9.1 In vertical crash cases: the mass of the body, clothing and equipment reacted by the seat.

9.2 Normally assumed to be the sum of: 80% of body mass, plus 80% of the mass of clothing less boots, plus 100% of the mass of equipment carried on the body above the knees.

10. SUBMARINING

10.1 In a crash landing or ditching with high downward and forward accelerations the body will tend to sink into the seat and, almost simultaneously, slide forward. Unless the restraint system is correctly designed the inertia forces on the hips will pull the torso under the lap belt.

LEAFLET 307/2

CRASH LANDING, DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

1. DESIGN

1.1 In amplification of chapter 307 paras 1, 2 and 3, the priority requirement is that the crew and occupants should be able to survive the specified impact conditions and then be able to get out of the rotorcraft. Referring particularly to paras 1.1(i) and (ii) the probability that the occupants of the Rotorcraft will survive a crash landing or ditching will be improved by the inclusion, at the initial design stage, of the following design features:

a) design of the airframe as a protective shell,

b) provision of high strength for the retention of massive parts,

c) evaluation of accelerations and consequent movements of each occupant,

d) consideration of hazards to each occupant during the crash,

e) consideration of hazards to each occupant during the escape and afterwards.

1.2 The energy of impact will be absorbed partly by the impacted structure but some energy will be absorbed by the seat, its local structure, and the energy absorbing devices provided for the purpose. The objective of good design for crashworthiness is to make the force-extension diagram for each relevant part of essential element of the rotorcraft as rectangular as possible and spread over as long a time as possible. To this end either the structure must be designed to absorb energy by progressive collapse, or impact-peak-reducing energy-absorbing mechanisms must be built into the load paths, or both. Leaflet 307/0 Ref 1 gives some examples of how this can be done. Provision of high static strength alone may result in excessively massive structure and in a considerable reduction in performance without necessarily giving any real guarantee of integrity in a crash landing or ditching.

1.3 On the other hand the uncontrolled partial absorption of energy which can arise from the bending of structural members within elastic limits can cause an increase in accelerating forces at points remote from the impact point. (The whiplash effect of dynamic overshoot). Care must therefore be taken to ensure, by proper design and testing of energy absorbers in all cases that this does not occur. See Chapter 307 para 2.3.4.

1.4 The requirements of Chapter 307 para 2.3.2 would be adequate on their own to determine all the crash landing and ditching forces in all parts of the airframe if dynamic analyses and tests were wholly accurate. As they are not it is necessary to supplement them with the arbitrary minimum static strength requirements given in Chapter 307 Section 4.

1.5 Where the rotorcraft has been designed to meet the normal alighting requirements of Chapter 304, the undercarriage, provided it has not been retracted, will provide some attenuation of peak loads, within its limitations, for vertical and near vertical crashes, but not for ditching or for horizontal crashes. See Chapter 307 para 4.3.3 and para 9 below.

1.6 In amplification of Chapter 307 para 3.1.2 it should be noted that the accelerations transmitted through a conventional structure in near-horizontal ditching may be more severe than in a crash landing where the forces are reacted largely by the frames. In a ditching the panels may fail and the frames penetrate the water with little vertical resistance, creating considerable horizontal deceleration. With proper design for energy absorption this problem can be avoided. Chapter 307 Table 2 Case 7 in conjunction with Table 9 may be used as a minimum ditching case for preliminary design.

1.7 The local accelerations in a rotorcraft descending vertically onto water will, in general, be higher than for an equivalent descent onto land due to the lack of energy absorbing devices. However the Rotorcraft Design Authority can assist in the reduction of these accelerations by such devices as "V-ing" and/or double skinning of the bottom of the fuselage. The area of the bottom should also be reduced to the minimum practical.

1.8 When a rotorcraft alights on water at a shallow angle and high speed the bottom produces a rearwards force which will pitch the rotorcraft nose-down and cause the forward fuselage to impact the water. The detailed design of the rotorcraft will determine the subsequent motion, particularly the rate of deceleration. An analysis will be necessary to determine the effect of the forces imposed on the front fuselage, the subsequent motion of the rotorcraft, and the ability of the fuselage to withstand these forces without failure.

1.9 Nose structure designs should be such that, when they are subjected to water impact, they maintain a surface which, even after partial collapse, will provide an upward acting force. This will keep the nose clear of the water and prevent the rotorcraft from turning over. The fuselage should be designed to accept an upward force at least sufficient to counteract the nose-down moment produced by the drag of the undercarriage or the bottom at the speed given in the Rotorcraft Specification. In particular the nose should not fail in such a manner as to leave a bluff structure which will then cause rapid deceleration of the rotorcraft and further endanger human life.

2. TESTS

2.1 STRUCTURE TESTS

2.1.1 Compliance with the dynamic requirements of Chapter 307 para.

2.3.2 cannot be established by static tests alone. Individual elements in the systems concerned (eg seats), can be dynamically tested to demonstrate adequate energy absorption and these tests will give some indication as to whether the pulse shape and size at specified points is acceptable. The frangibility of the support structure and its capacity to absorb energy by progressive deformation and collapse can only be substantiated by similar tests.

2.1.2 Consideration should be given to the provision of a fully instrumented test vehicle, fitted with masses representing the relevant elements, with the connecting structure, with all energy absorbing elements, and with essential systems, which can be tested in an appropriate manner.

2.1.3 Details of suitable static and dynamic tests are given in references 8, 9 and 10 of Leaflet 307/0. However, the manner in which these documents specify the requirements creates additional design cases. These are combined loading cases and are included in Table 2 of Chapter 307.

2.2 DITCHING MODEL TESTS

2.2.1 The behaviour of the rotorcraft in ditching and during a precautionary alighting on water should be investigated in ditching model tests unless the design can be accepted by comparison with satisfactory earlier designs. Scoops, flaps, projections and other features likely to affect the hydrodynamic behaviour of the rotorcraft should be simulated. The tests should also ensure, as far as is practical, that the doors and windows are capable of withstanding the probable water pressure they will experience.

3. REPAIRABILITY

3.1 In amplification of Chapter 307 para 1.1(iii) and to reduce cost of repair after the following low energy survivable accidents:

a) heavy landing undercarriage up, or heavy landing followed by undercarriage failure,

b) inadvertent undercarriage retraction (static or during landing/taxi-ing),

c) landing with one or more undercarriage units retracted,

d) any unit running off the landing surface into a ditch or other obstruction,

the design should provide for modular replacement of the energy absorbers, and of the energy absorbing structure affected, whenever possible.

4. SUPPLEMENTARY DYNAMIC ANALYSIS

4.1 In addition to the Dynamic Analysis of Chapter 307 para 2.3.2 necessary to meet the requirements for crash landing and ditching, an analysis of any of the following events may be required by the Rotorcraft Specification.

a) Landing on runway at design V_V with any practical combination of one or more undercarriage units retracted.

b) Main wheel running off runway into ditch or obstruction.

c) Nose or tail wheel running off runway into ditch or obstruction.

4.2 The speeds considered should cover the practical range for each case between zero and the maximum emergency landing speed.

4.3 The analyses should be conducted to show, as far as is practicable, whether any of these cases indicates a need for additional structural strength or energy absorption capability or special operating procedures.

5. TRADE-OFF STUDIES

5.1 In amplification of Chapter 307 para 4.9.5, where trade-off studies on the evacuation of occupants from a crashed or ditched Rotorcraft are undertaken to establish evacuation time, it may be found that static strength factors higher than those specified in Chapter 307 are necessary.

5.2 The results of these studies should be used as the basis for additional strength, stiffness or energy absorption capacity or for special instructions to aircrew, ground crew or passengers, as necessary.

6. ATTACHMENT AND RESTRAINT OF COMPONENTS, EQUIPMENT, CARGO AND FREIGHT

6.1 In amplification of Chapter 307 paras 4.9.2 and 5.1.2 where the static strength of the item, its attachments or local support structure, does not meet the requirements, it may be restrained by the provision of a bulkhead, or by the addition of strops or anchors, to provide the same acceleration factors as required by para 4.9.2. Energy absorbers may be included if advantageous.

7. STRENGTH OF ENERGY ABSORBERS

7.1 In amplification of Tables 3, 4 and 6 of Chapter 307, where acceleration factors are given as "bottomed" in the form 35 (30-15) the first figure is the estimated acceleration without energy absorber and the pair in brackets are the maximum initial and final values to which the energy absorber is required to reduce the acceleration. These figures are then the acceleration factors for minimum static strength of the energy absorber and associated parts.

8. VERTICAL IMPACT

8.1 The design features recommended in support of Chapter 307 para 4.3 are:

(a) locate massive items as low as possible and so that they will not intrude into occupied areas during a crash,

(b) design structure to prevent it buckling inwards and crushing occupants,

(c) provide energy-absorbing structure below cockpit and cabin floors and below massive parts,

(d) provide energy-absorbing undercarriage and

(e) provide energy-absorbing seats for all occupants.

9. UNDERCARRIAGE LAYOUT AND DESIGN

9.1 In a rotorcraft with three undercarriage units (whether nose or tail) the distance between the main and auxiliary units is of considerable importance in the near-vertical crash case and must be chosen carefully to maximise energy absorption and minimise the forces created.

9.2 The undercarriage and its support structure should be designed to collapse in a controlled energy-absorbing manner so that it does not penetrate the cabin, or any space occupied by crew or other personnel, or the fuel cells.

9.3 The undercarriage should be designed to absorb energy after the fuselage touches the ground to improve survivability.

9.4 Skid type undercarriages should be designed not to snag on rough ground or such obstacles as tree roots.

10. EARTH SCOOPING

10.1 The impact conditions of Chapter 307 paras 4.2.3 and 4.5.1 (low level approach to ploughed soil) are likely to lead to roll-over should any significant amount of earth scooping occur. Design features which should be considered to prevent this occurring are:

(a) provide a large flat undersurface to increase the tendency to slide over the ground.

(b) minimise inward local buckling of this area.

11. BUCKLING OF STRUCTURE

11.1 In amplification of Chapter 307 para 2.2.1 and to minimise hazards to occupants the structure of the protective shell should be designed to:

(a) provide adequate strength to prevent inward bending or buckling,

(b) position passengers and crew away from likely fracture areas,

(c) buckle outwards if possible.

12. HEAD STRIKE ENVELOPE (See also Chapter 106 para 7)

12.1 The head is the most vulnerable part of the body and the head strike envelope should be established in all axes. Design of the protective shell should incorporate the following features:

(a) No solid parts or structural members should be within the envelope,

(b) Any parts which must encroach into the head strike envelope should be smooth and made of frangible or ductile material, or be suitably rounded and padded. See para 14 below.

(c) Frangible parts should break away at a force not exceeding 1335 N (300 lbf).

(d) Visionic equipment should be padded and should also collapse or breakaway at a force not exceeding 1335 N (300 lbf).

13. FLAILING OF ARMS AND LEGS

13.1 The occupant's immediate environment should be designed so that, when parts of the body do flail and contact structure or equipment, the injury suffered is minimised.

13.2 Ductile materials, frangible parts, and energy absorbing padding should be provided to protect flailing arms and legs. Frangible parts should break away at a force of 3563 N (800 lbf).

13.3 Space between pedals and structure surrounding the feet should be designed so that failure will not trap them.

13.4 Data for constructing flailing envelopes is given in Leaflet 105/3, and in Ref 1 of Leaflet 307/0.

14. PADDING

14.1 Where padding is provided in accordance with paragraphs 12 and 13 above it should meet the following requirements.

14.2 The padding should reduce the velocity of the head from 7.6m/s (25ft/sec) to zero without it exceeding a deceleration of 150G. The mass of the head should be assumed to be 5 kg (11 lb).

14.3 Suitable stress-strain characteristics for padding are given in Ref 2 of Leaflet 307/0.

14.4 Edges and corners to be padded should have a radius of not less than 13mm (0.5in).

15. CONTROL COLUMNS

15.1 Control columns, including centre and side cyclic sticks, should be designed not to impale the pilot or co-pilot. Separation, telescoping, frangible joints or padding may be used.

15.2 Centre sticks should break flush with the floor.

15.3 Any separating or telescoping mechanism used should be incapable of premature actuation.

15.4 Frangible parts should break cleanly.

15.5 Control columns which pass longitudinally through the instrument panel should not be used. If unavoidable they should be collapsible.

16. WINGS

16.1 The Rotorcraft Specification will state whether wings should be frangible or not.

16.2 Wings used to support external stores help to prevent roll-over and should not be frangible but should allow the stores to separate. However the wing should break off before the fuselage itself collapses.

17. RELIABILITY OF SEATS

17.1 In amplification of Chapter 307 para 7 the reliability analysis should include a Failure Mode, Effects and Criticality Analysis.

17.2 Except for fabric parts the minimum life should be 5000 hrs of rotorcraft life and 5000 adjustments.

17.3 Fabric parts should meet strength requirements after 5 years and have a minimum shelf life of 30 years.

TABLE 1 - 80TH PERCENTILE POTENTIALLY SURVIVABLE CRASH

1	2		3	4	5
Direction (See Note 3)	Velocity Change (m/sec:ft/sec)		Peak Acceleration Factor (G)	Mean Acceleration Factor (G)	Impulse Duration (ms)
Longitudinal forward	11.5	37	14	7	168
Longitudinal aft	7	23	8	4	168
Lateral	7	23	8	4	168
Vertical, down	10	32	20	10	100
Vertical, up	5.5	18	7	3.5	168

TABLE 2 - 85TH PERCENTILE POTENTIALLY SURVIVABLE CRASH

1	2		3	4	5
Direction (See Note 3)	Velocity Change (m/sec:ft/sec)		Peak Acceleration Factor (G)	Mean Acceleration Factor (G)	Impulse Duration (ms)
Longitudinal forward	12	39	16	8	155
Longitudinal aft	7.5	24	9	4.5	155
Lateral	7.5	24	9	4.5	155
Vertical, down	10.5	34	24	12	90
Vertical, up	6	19	8	4	155

TABLE 3 - 90TH PERCENTILE POTENTIALLY SURVIVABLE CRASH

1	2		3	4	5
Direction (See Note 3)	Velocity Change (m/sec:ft/sec)		Peak Acceleration Factor (G)	Mean Acceleration Factor (G)	Impulse Duration (ms)
Longitudinal forward	13	43	18	9	150
Longitudinal aft	8	26	11	5.5	150
Lateral	8	26	11	5.5	150
Vertical, down	11.5	37	30	15	75
Vertical, up	6.5	21	9	4.5	150

TABLE 4 - 95TH PERCENTILE POTENTIALLY SURVIVABLE CRASH

1	2		3	4	5
Direction (See Note 3)	Velocity Change (m/sec:ft/sec)		Peak Acceleration Factor (G)	Mean Acceleration Factor (G)	Impulse Duration (ms)
Longitudinal forward	15	50	30	15	104
Longitudinal aft	9	30	18	9	104
Lateral	9	30	18	9	104
Vertical, down	13	42	48	24	54
Vertical, up	7	24	15	7.5	100

Notes on Tables 1 to 4:

1. The figures are derived from crash data gathered for rotorcraft during the period 1950 to 1970.

2. The essential requirements are stated in columns 2 and 5. Columns 3 and 4 give the consequential acceleration factors assuming a triangular pulse having a peak/mean ratio of 2.0.

3. Directions given are those of the movement of the pilot's eyeballs relative to the rotorcraft axes.

4. Where acceleration factors are required for static strength analysis of equipment attachment, the mean values may be used. Dynamic tests must however be done to the velocity change requirements.

LEAFLET 307/3

**CRASH LANDING, DITCHING AND
PRECAUTIONARY ALIGHTING ON WATER**

SEATS AND STRETCHERS (LITTERS)

1. ALL SEATS AND STRETCHERS

1.1 The energy absorption and strength requirements of Chapter 307 para 2.3.2 and Table 2 are applicable to all seats and stretchers. Table 1 gives the maximum accelerations a human body can tolerate and these must not be exceeded.

1.2 Specific static strength requirements are given in Chapter 307 paras 4.7, 4.8 and 4.12.
These are intended as additional arbitrary minima which apply if greater than the accelerations measured in dynamic tests.

1.3 Notwithstanding the minimum requirements for static strength and the maxima allowable in Chapter 307 Table 1 the Rotorcraft Design Authority should aim at limiting the vertical acceleration at the base of the occupant's spine to 15G if possible.

1.4 A further design aim should be to restrain all occupants throughout the crash and provide the maximum support in the most likely direction in most severe cases allowing for the operation of energy absorbers and the buckling of structure.

1.5 All seats and stretchers should be as comfortable as possible but comfort should not compromise safety. Submarining should be impossible.

1.6 Any part of a seat or stretcher which could be bumped or kicked, or used as a step or handhold should be design accordingly. See Chapter 307 para 4.13.

1.7 Covers for floor depressions or wells necessary to provide for seat stroking should be covered with frangible material which will allow the seat to stroke through the cover at a force not exceeding 667 N (150 lbf).

1.8 Design of the attachment of all seats and stretchers should allow for bulging and warping of structure. A movement of any single seat attachment in any direction by 10 ^{o}, or the relative movement of any pair of attachments by 5 ^{o} each in opposite directions, should be possible and the seat should still meet the static strength and energy absorption requirements.

1.9 Seats and stretchers may be attached through their local structure to the ceiling in addition to the floor, to help react crash forces, provided that the ceiling can be shown to be strong enough for this purpose without creating an additional hazard for the occupants.

1.10 The design of seats should be co-ordinated and interfaced with the design of other areas to achieve a completely integrated and efficient crashworthy design.

2. CREW SEATS

2.1 Note on Chapter 307 Table 3. On crew seats the requirements of Ref 8 of Leaflet 307/0 are framed to provide high static strength with small deflections except in the downward direction where an energy absorber is specifically required to attenuate the peak accelerations, caused by the 42ft/sec (12.8m/s) velocity change requirement of Chapter 307 Table 2, to 25G maximum (15G preferred, see para 1.3 above) within the stroke of the energy absorber, which should be at least 12ins (305mm).

2.2 The energy absorber should be capable of providing for the range of effective mass of the aircrew (see Leaflet 307/1 para 9 and Tables 1, 2, 3 and 4 of this leaflet) in three increments or by being continuously variable. Its strength must be adequate to resist the forces it generates without premature failure.

2.3 The seat motion should be parallel to the seat back tangent line within $\pm 10^{o}$.

2.4 Unless otherwise stated in the Rotorcraft Specification the seat should provide for the 5th and 95th percentile mass including clothing and equipment (see Tables 1, 3 and 4) and also for armour if carried. See Chapter 113.

2.5 The seat should provide for the 5th and 95th percentile dimensions. See Leaflet 105/3.

3. PASSENGER, TROOP AND PARATROOP SEATS

3.1 Note on Chapter 307 Table 4. On seats for passengers, troops and paratroops static strength is specified in Ref 9 of Leaflet 307/0, to meet the requirements of Chapter 307 Table 2, in only two directions. In the other three directions energy absorbers are specifically required to attenuate the peak accelerations to the "bottomed" value given within the stroke of the energy absorber. The strength of the energy absorber must be adequate to resist the forces it generates without premature failure.

3.2 The seats should provide for the 5th and 95th percentile mass and dimensions of fully clothed and equipped troops. See Table 2.

3.3 The preferred alignment of the seats (forward, aft, lateral) is stated in Chapter 714 paras 2.5.1, 2.5.2 and 2.6.3.

3.4 Individual (single) seats are preferred to units of 2 or more for ease of application of energy absorbers particularly of a multi-seat unit being occupied by only one person.

3.5 Provision should be made against whiplash effects of the head.

3.6 The energy absorber should include a snubber against rebound and some indication of energy absorber stroking, even partial, so that maintenance crew checking the effects of heavy landing would notice.

4. STRETCHERS (LITTERS)

4.1 Stretchers should be aligned laterally if possible. If not then special attention should be given to the design of the harness.

4.2 A structural pan should be provided if conventional (low strength) stretchers are to be used.

4.3 There should be adequate clearance beneath the lowest stretcher in a stack and the floor below it to allow for deflection in the 25G vertical case. The minimum should be 200mm (8 in).

4.4 Attachment of stretchers to the airframe should be designed so that it is easy to see if they are properly locked.

4.5 Two straps should be provided with each stretcher each capable of taking 8900 N (2000 lbf) in pure tension. At this load elongation should be not more than 100mm (4 in) in 1220mm (48in). Straps should be automatically self-adjusting for length and fitted with quick-disconnects.

TABLE 1

MASS OF AIRCREW (kg)
(SOURCE DEF-STAN 00-970 Vol 1 Leaflet 105/3

Item	3rd percentile	5th percentile	50th percentile	95th percentile	99th percentile
Body	59.46	61.33	74.46	90.02	96.50
Clothing) Boots)	14.00	14.00	14.00	14.00	14.00
Equipment	10.00	10.00	10.00	10.00	10.00
Total Mass	83.46	85.33	98.46	114.02	120.50
Effective Vertical Mass	67.77	71.26	81.77	94.22	99.40

TABLE 2

MASS OF TROOP OR PARATROOP SOLDIER (kg)
(SOURCE MIL-S-85510 (AS))

Item	3rd percentile	5th percentile	50th percentile	95th percentile	99th percentile
Body	55.34	57.3	71.0	91.6	100.17
Clothing	1.4	1.4	1.4	1.4	1.4
Boots	1.8	1.8	1.8	1.8	1.8
Equipment	15.5	15.5	15.5	15.5	15.5
Total Mass	74.04	76.0	89.7	110.3	118.87
Effective Vertical Mass	60.89	62.46	73.42	89.90	976.76

TABLE 3

MASS OF AIRCREW (kg) RAF
(SOURCE STANAG 3950 AI)

Item	3rd percentile	5th percentile	50th percentile	95th percentile	99th percentile
Body	59.5	61.4	74.5	90.0	96.5
Clothing) Boots)	15.5	15.5	15.5	15.5	15.5
Equipment	10.0	10.0	10.0	10.0	10.0
Total Mass	85.0	86.9	100.0	115.5	122.0
Effective Vertical Mass	68.6	70.1	80.6	93.0	98.2

TABLE 4

MASS OF AIRCREW (kg) USAF
(SOURCE STANAG 3950 AI)

Item	3rd percentile	5th percentile	50th percentile	95th percentile	99th percentile
Body	61.6	63.6	78.2	95.6	103.3
Clothing) Boots)	15.5	15.5	15.5	15.5	15.5
Equipment	10.0	10.0	10.0	10.0	10.0
Total Mass	87.1	89.1	103.7	121.1	128.8
Effective Vertical Mass	70.2	71.8	83.5	97.4	103.6

LEAFLET 307/4

CRASH LANDING, DITCHING AND PRECAUTIONARY ALIGHTING ON WATER

OPERATIONAL RECOMMENDATIONS FOR CARGO AND FREIGHT

1. GENERAL

1.1 These recommendations concern the restraint of cargo and freight and amplify the mandatory strength and energy absorption requirements of Chapter 307 para 4.10.

1.2 They are derived from para 5.3.1 of Ref 2 of Leaflet 307/0.

2. OPERATIONAL RECOMMENDATIONS

2.1 If the structure of the fuselage or floor is not strong enough to withstand the forces arising from the velocity changes of Chapter 307 Table 2 energy absorbers and space to operate them, additional to those required by Chapter 307 Para 4.10 should be provided.

2.2 Nets should be constructed of stiff material to reduce dynamic overshoot (see Leaflet 307/2 para 1.3) to a minimum.

2.3 Restraining lines (chains, strops etc) used for large cargo (over 3ft cube) should be arranged so that maximum load in each is reached simultaneously. Restraining lines having different force/extension characteristics should not be used on the same piece of equipment or cargo.

2.4 If load limiting energy absorbers are used restraining lines used on the same piece of equipment or cargo should be of low force/extension characteristics to ensure efficient energy absorption.

CHAPTER 308

STRENGTH FOR GROUND HANDLING

1 INTRODUCTION

1.1 The requirements of this Chapter apply to the rotorcraft as a whole and should be read in conjunction with the design, operational and standardisation requirements for transport, handling, and storage given in Chapter 801.

2 SLINGING

2.1 Proof and ultimate factors of not less than 2.25 and 3.0 respectively shall be achieved when either:

(i) the whole rotorcraft at its maximum design take-off mass less passengers and crew, or

(ii) any component of the rotorcraft which is required to be slung,

is hanging freely from the slinging gear (see Specification AD2/SRD/902/P - Lifting Tackle).

3 JACKING

3.1 Proof and ultimate factors of not less than 2.25 and 3.0 respectively shall be achieved at maximum design take-off mass less passengers and crew when supported on the jacking points. To allow for some surface slope and some abnormal rotorcraft attitude, this condition shall be met at angles up to 10° in any plane between rotorcraft vertical axis and the jacking axis.

3.2 Consideration shall be given to the horizontal loads present at the jacking points in the following conditions:

(i) On land in the conditions stated in para 3.1 and at such greater angles but lesser rotorcraft masses as may be appropriate to effect crash recovery.

(ii) At sea in the sea conditions given in the rotorcraft specification using appropriate ship motion data and assuming standard naval trestling and lashing practice whilst jacked.

4 TOWING - WHEELED ROTORCRAFT

4.1 The rotorcraft as a whole shall have adequate strength to provide proof and ultimate factors of not less than 1.5 and 2.0 respectively when towing in either direction in accordance with the detailed requirements of paras 5, 6 and 7 below.

4.2 Consideration may be given to reducing these factors to 1.125 and 1.5 respectively, if the towing load designs the auxiliary unit or any part of the rotorcraft structure, provided that failure in all modes (including lateral bending against a rigidly held wheel) is prevented by shear pins or equivalent devices which are made of a material sufficiently strong so that any substitute that may be used in service is unlikely to be stronger.

4.3 The requirements of paras 4.1 and 4.2 above shall be met at maximum design take-off mass.

5 TOWING STRESSING CASE - GENERAL

5.1 With the rotors stationary, the rotorcraft structure and in particular the rotor blades and their supporting structure shall be designed to withstand the loads resulting from the application during towing, on each undercarriage unit, of:

(i) a vertical load equal to 1.67 times the maximum static reaction, and

(ii) any combination of drag and side loads equal to from 0 to 0.25 times the vertical load.

5.2 For skid, ski or float equipped rotorcraft, this case assumes that the rotorcraft is standing on its wheeled means of ground handling required by para 9 below.

6 TOWING STRESSING CASE - WHEELED MAIN UNDERCARRIAGE UNITS (see Fig. 1)

6.1 Case 1. Conditions shall be as follows:

(i) Towing load P = 0.15 g M_T at each main unit towpoint separately and together balanced by a side load S having a value of P Sin θ and a drag load D having a value of P Cos θ at each mainwheel axle.

(ii) Appropriate auxiliary unit vertical reactions with zero side and drag reactions.

(iii) Appropriate main-unit vertical reactions at each main-wheel.

(iv) Balancing inertia loads at the rotorcraft cg if necessary (see Leaflet 308/1).

(v) Tow load angle θ at any critical value from zero to ±30° unless limited by the geometry of the rotorcraft.

6.2 Case 2. Conditions shall be as follows:

(i) Towing load P = 0.15 g M_T at each main-unit towpoint balanced entirely by inertia forces.

(ii) Appropriate auxiliary unit vertical reaction with zero side and drag reactions.

(iii) Appropriate main-unit vertical reactions at each main-wheel.

(iv) Zero drag and side loads at each main-unit.

(v) Tow load angle θ at any critical value from zero to ±30° unless limited by the geometry of the rotorcraft.

6.3 Couples in the vertical plane caused by the loads in these cases may be assumed to be balanced by pitch and roll inertia, but this may be ignored so far as stressing is concerned.

7 TOWING STRESSING CASES - WHEELED AUXILIARY UNDERCARRIAGE UNITS (see Fig.2)

7.1 The conditions shall be:

(i) towing load $P = 0.15\ g\ M_T$,

(ii) auxiliary unit alignment angle A at any critical value between zero and the maximum castoring angle,

(iii) tow load angle B at any critical value from zero to the maximum possible.

7.2 The applied loads shall be balanced by either:

(i) rotorcraft inertia alone, or

(ii) by a combination of a force Q applied at the axle equal and opposite to P together with such inertia forces as are needed to balance the rotorcraft.

7.3 Where a load-limiting device is incorporated in the tow bar its value shall be chosen having consideration to the torsional forces which could be applied to the auxiliary unit before the device operates.

7.4 For ship-borne rotorcraft, a load limiting device is not permitted. A dynamic analysis shall therefore be done to check if applied loads greater than $0.15\ g\ M_T$ could occur in the sea states specified and if they do they shall be used for design. In particular, the case of a sudden brake application and the effects of surface irregularities on a pitching, rolling deck shall be considered. See Leaflet 308/1, para 5.

8 TOWING - STEERABLE UNDERCARRIAGE UNITS

8.1 Where wheel steering is provided, it shall be possible, with the steering mechanism disconnected if necessary, to castor through angles required by the rotorcraft specification for ground handling, in all practical combinations of mass and cg position which may arise on the ground.

9 GROUND HANDLING - SKID, SKI AND FLOAT EQUIPPED ROTORCRAFT

9.1 The rotorcraft shall be provided with a wheeled means of ground handling on smooth, hard, flat surfaces at the maximum design take-off mass.

9.2 It shall be possible to fly the rotorcraft, throughout the full flight envelope, with the wheeled means of ground handling remaining fitted unless otherwise agreed with the Rotorcraft Project Director.

9.3 With the rotorcraft stationary on the surface on its skid, ski or float undercarriage as appropriate, the ground support equipment and manpower required to transfer the rotorcraft to its ground handling wheels or wheeled device shall be kept to a minimum and the technique agreed with the Rotorcraft Project Director.

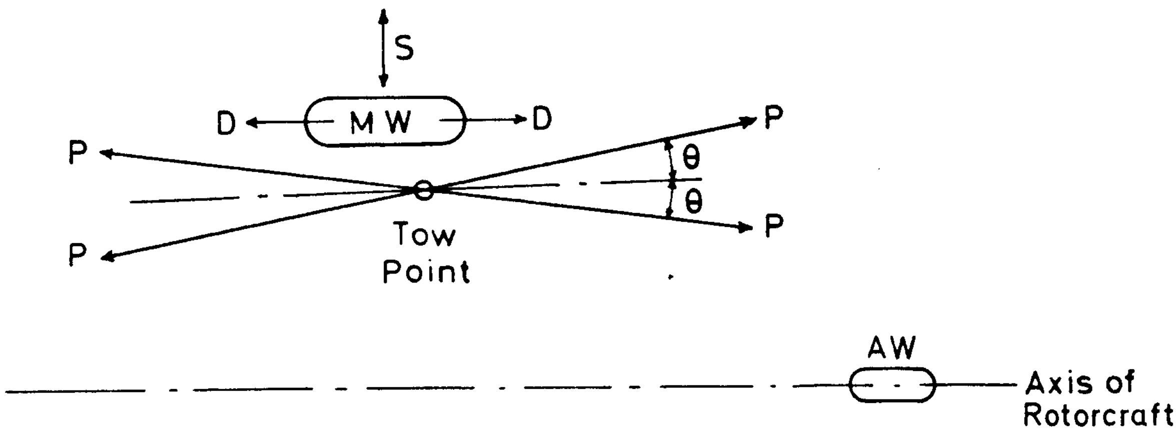

FIG.1 TOWING BY BRIDLE

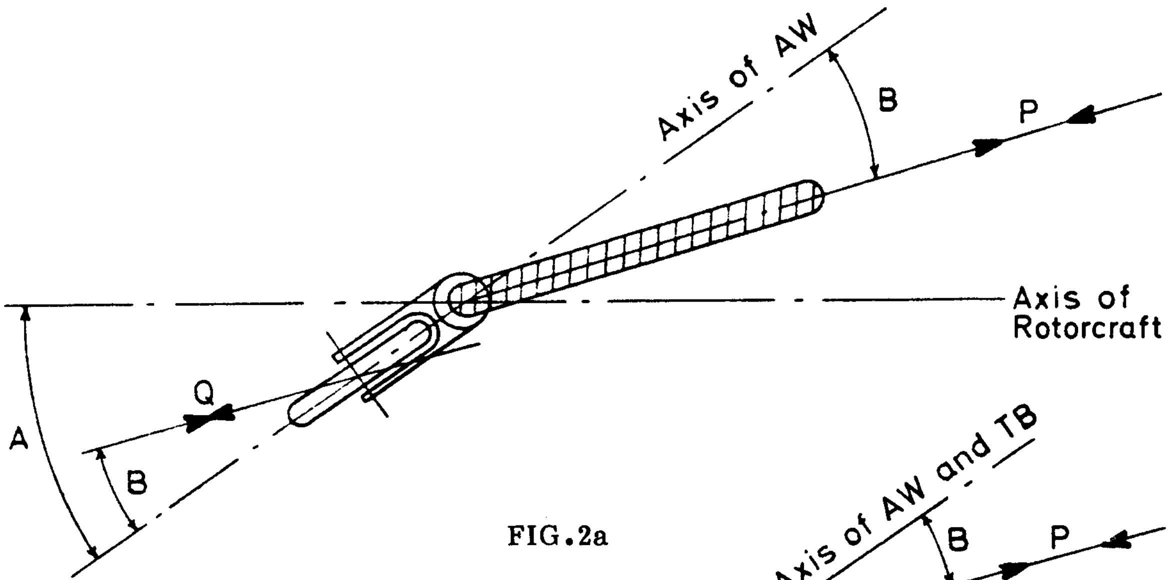

FIG.2a

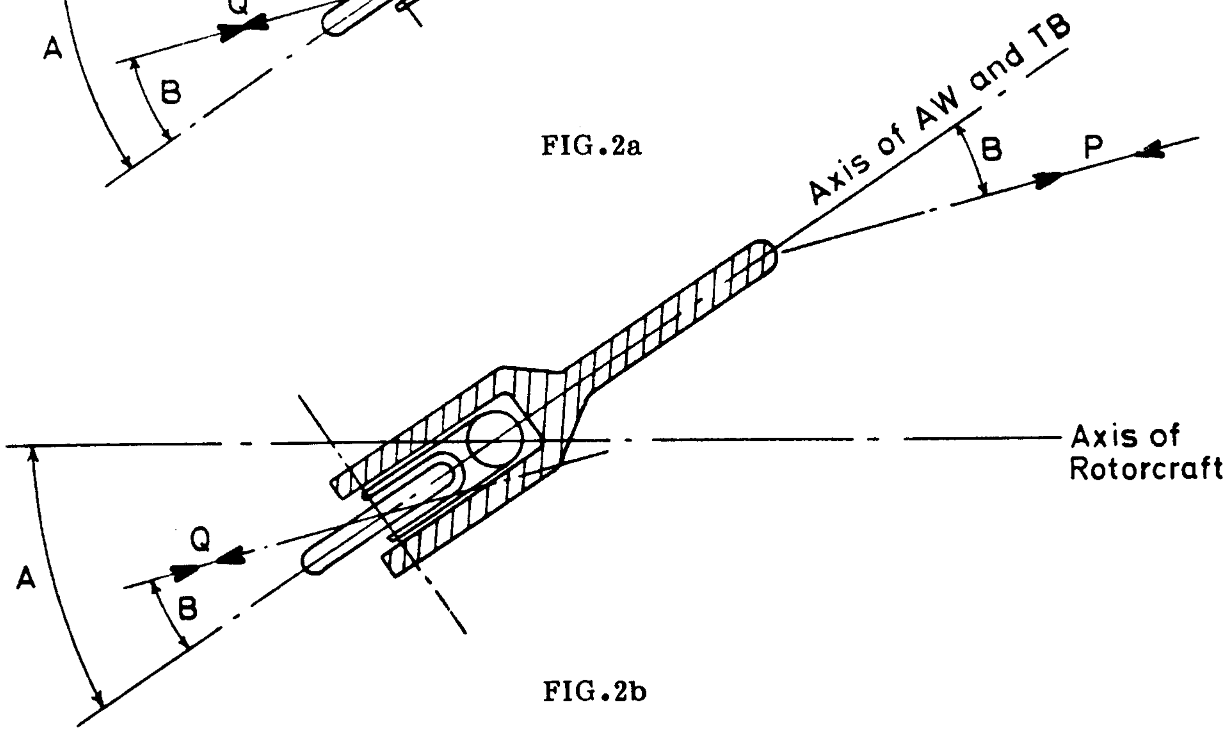

FIG.2b

FIG.2 (a & b) AUXILIARY-WHEEL TOWING

LEAFLET 308/1

STRENGTH FOR GROUND HANDLING

TOWING - WHEELED ROTORCRAFT

1 INTRODUCTION

1.1 Minimum mandatory strength requirements are given for towing in Chapter 308. This Leaflet gives background information and recommendations relating to those requirements.

1.2 Design, operational, and standardisation requirements are given for towing in Chapter 801. Note particularly that load-limiting devices are not allowed on tow bars provided for use on board ship.

2 GENERAL

2.1 The extreme cases which have to be considered are:

(i) towing from a point at or near a main or auxiliary unit which happens to be on a surface where the coefficient of friction is nearly zero (as for spilt fuel or wet mud on a firm surface) and where the only reaction is supplied by the inertia of the rotorcraft,

(ii) extracting a wheel which is bogged down, or similarly obstructed,

(iii) turning an auxiliary wheel with zero forward speed.

2.2 The applied towing force would, in practice, be balanced by a combination of some or all of the following:

(i) Frictional forces at points of tyre contact with the ground.

(ii) Gravitational forces due to inclined or uneven ground surfaces.

(iii) Re-distribution of vertical reactions at wheels.

(iv) Linear and angular inertia forces.

2.3 So far as possible, the need for detailed consideration of all those possibilities on every rotorcraft has been eliminated by specifying detailed stressing cases which, while not representing accurately any one particular condition, will normally cover all reasonable conditions except where the rotorcraft geometry is unconventional.

2.4 It is not intended that the requirements should be used to design major parts of the undercarriage or structure of the rotorcraft if satisfactory towing procedures can be devised which will provide for all normal rotorcraft movements and also the extreme cases described in para 2.1. It is the Designer's responsibility to agree these procedures with the Rotorcraft Project Director and include them in the Operating Manual.

2.5 The tyre and shock absorber deflections may be usually assumed to be those for the rotorcraft standing on level ground at maximum design take off mass (M_T) and associated cg position. However, appropriate allowances should be made if there are considerable geometrical changes under towing loads.

3 MAIN-UNDERCARRIAGE ATTACHMENT LOADS

3.1 The design loads illustrated for the main-undercarriage towing points in Chapter 308, Fig.1, are arranged to provide for the use of a towing rope, chain, or wire, on one main-undercarriage unit. Note that the stressing Case 1 of Chapter 308 para 6.1 implies a total force P of 0.3 g M_T when towing from two points even on smooth surfaces. Double point towing should therefore be used on any significant slopes (8½° to 17½°).

3.2 Two different methods of auxiliary wheel towing are represented in Chapter 308, Fig.2. Fig.2a illustrates the case when the towing arm, wire, chain, or rope, is attached to a strong point on a castoring auxiliary unit. Fig.2b illustrates the case when a towing arm is attached to the auxiliary wheel such that towing and steering occur at the same time. The applied loads will be opposed by side, draft, and torsional tyre forces.

3.3 Consideration should be given to variations in these loads and forces arising from:

(i) geometry of the auxiliary unit and wheel(s),

(ii) frictional forces from the castoring system,

(iii) the displacement of the tyre(s) relative to the ground, and

(iv) practical combinations of towing speeds and swivel rates, particularly the case of swivelling with zero forward speed.

4 AUXILIARY UNIT

4.1 Standardisation requirements call for a standard spool on each side of the auxiliary undercarriage unit for towing purposes. It is the designer's responsibility to ensure that the material chosen for this spool is such that the strength requirements of Chapter 308 are met.

5 SHIPBORNE ROTORCRAFT

5.1 Shipborne rotorcraft are not allowed to have a relief device in the tow bar (see Chapter 801) and must be stressed for the maximum loads expected in the worst sea state for which towing is required. A dynamic analysis is therefore required.

5.2 For preliminary stressing, a tow force of 0.8 g M_T should be used to represent sudden braking of the rotorcraft or tractor on a non-slip surface and the auxiliary wheel vertical load may be calculated from:

$$R_N = M_{T.\ g}\ (1_m + 1.6\ y)/(1_m + 1_n)$$

where y is the moment arm of the braking force about the rotorcraft cg. Then y = h for tractor braking with auxiliary wheel tow at the auxiliary wheel axle and y = H for main-wheel braking of the rotorcraft with any form of towing.

CHAPTER 309

PICKETING, TIE-DOWN AND RAPID SECURING SYSTEMS

1 INTRODUCTION

1.1 The requirements of this Chapter are to ensure that provision is made on all rotorcraft for picketing, and if specified, the tie-down for ground running of engines.

1.2 Requirements are also given for rapid securing systems associated with rotorcraft operation from small ships.

2 OPERATIONAL REQUIREMENTS

2.1 GENERAL

2.1.1 Provision shall be made for picketing the rotorcraft from picketing points located on or near each undercarriage unit. The selection of picketing points shall allow for securing jacked or trestled rotorcraft in adverse wind and sea conditions.

2.1.2 The picketing points shall be located so that tension in the lashings will keep the undercarriage struts in compression.

2.1.3 The picketing points shall accommodate sufficient standard lashings to meet the strength requirement of para 3.1.

2.1.4 When designing for picketing, account shall be taken of any forces provided by chocks or brakes.

2.2 PICKETING ON LAND

2.2.1 Rotorcraft picketed on land shall be safe in wind speeds up to 80 kn horizontally from any direction, and remain serviceable afterwards.

2.3 PICKETING ABOARD SHIP

2.3.1 Rotorcraft shall be safe when picketed on a ship according to the ship's picketing plan in the sea states given by the Rotorcraft Project Director, and remain serviceable afterwards. If sea states have not been specified the following values of wind strengths and ship motion are to be used:

(i) Wind speeds up to 80 kn from any direction between dead ahead and 45° each side of the ship's fore-and aft axis, or 65 kn from any other quarter.

(ii) Angles of up to ±10° pitch and up to ±30° roll concurrently, together with additional accelerations of 7.62 m/s^2 (25 ft/s^2) normal to the flight deck, 1.52 m/s^2 (5 ft/s^2) across and parallel to the flight deck, and 0.61 m/s^2 (2 ft/s^2) along the fore-and-aft axis of the flight deck.

2.3.2 The lashing configuration shall employ the minimum practical number of lashings and occupy the smallest spread compatible with the deck securing points, spacings of which are to be obtained from the Rotorcraft Project Director.

2.4 SECURING OF ROTOR BLADES AND CONTROL SURFACES

2.4.1 All rotor blades and control surfaces shall be safe-guarded against damage in wind speeds up to 80 kn by provision of lashing devices and lashing points, and locking devices as appropriate, both in the rotors spread and rotors folded configurations. If this creates a design case for part of the structure, the problem should be discussed with the Rotorcraft Project Director.

2.4.2 The securing of rotor blades either spread or folded and the locking of control surfaces shall not prevent the rotorcraft being ground-handled.

2.4.3 The design of locking devices for control surfaces shall be such that under routine checking procedures the pilot will be aware that they are applied. It shall be impossible to apply take-off power with any lock engaged.

2.5 TIE-DOWN

2.5.1 If tie-down for ground running of engines is specified, the same attachment points and securing devices as required for picketing shall be used unless otherwise agreed with the Rotorcraft Project Director.

2.5.2 Tie-down, if specified, shall provide for ground running of engines safely in wind speeds up to 40 kn horizontally from any direction.

2.6 RAPID SECURING SYSTEMS

2.6.1 The operation of a rapid securing system, both during alighting and take-off, shall be under the direct control of the pilot/aircrew. The pilot/aircrew shall be able to override the system in all circumstances in order to abort the operation of rapid securing. The system shall be capable of repeated operation without the need for resetting by groundcrew. The time to reset the system in the air shall be agreed with the Rotorcraft Project Director.

2.6.2 The design shall be such that the rotorcraft is not secured until after the touchdown has been accomplished but, once this has occurred, the securing action shall be complete within 2 seconds of being initiated.

2.6.3 The requirements of Chapter 301 para 11 shall be met when the rapid securing system is restraining the rotorcraft. The rotorcraft securing system shall remain serviceable after alighting in all combinations of velocity, acceleration and angle up to the maxima in Leaflet 304/3 in the sea state specified. Once operated, the system shall be able to maintain its restraining action for a minimum of 24 hours in sea states up to that specified.

2.6.4 The rotorcraft specification will state if the rapid securing mechanism shall permit the rotorcraft to change heading while still secured to the deck and the angles through which this is required.

2.6.5 Securing devices fitted to the rotorcraft shall not impair the correct operation or functioning of other installed equipment including winch cables, strops for the carriage of underslung loads, and radio aerials. When locked and tensioned to the deck, the rotorcraft shall retain sufficient ground clearance to permit the loading of the external stores and weapons listed in the Rotorcraft Specification. The securing system shall not cause oleo legs and tyres, when fitted, to become fully compressed in the conditions of ship motion specified.

2.6.6 Any installation, addition or modification to the deck that is part of a rapid securing system shall not impede the movement of the rotorcraft, weapon trolleys or necessary servicing equipment. Such deck equipment shall not damage the wheels or tyres of the rotorcraft during alighting, take-off or movement over the deck.

3 STRENGTH

3.1 STRENGTH FOR PICKETING AND TIE-DOWN

3.1.1 At all masses between the basic mass and the maximum design take-off mass, the picketing points and the surrounding structure shall have proof and ultimate factors not less than 1.125 and 1.5 respectively under the conditions specified in para 2.2 for picketing on land, para 2.3 for picketing on board ship, and para 2.5 for tie-down if specified.

3.2 STRENGTH OF ROTOR BLADES AND CONTROL SYSTEMS

3.2.1 Rotor blades and their attachments, and control surfaces and their attachments, shall have proof and ultimate factors not less than 1.125 and 1.5 respectively under the loads arising in the conditions specified in para 2.4.

3.3 STRENGTH FOR RAPID SECURING SYSTEMS

3.3.1 At all masses between the basic mass and the maximum design take-off mass, the rapid securing system shall have proof and ultimate factors not less than 1.125 and 1.5 respectively under the conditions specified in the Rotorcraft Specification for operations from ships and in para 2.6 above.

3.3.2 The rotorcraft structure shall have proof and ultimate factors not less than 1.0 and 1.33 respectively on the maximum failing load of the weakest part of the rapid securing system.

3.3.3 The estimated fatigue life, supported by tests where necessary, of each part of the rapid securing system shall be agreed with the Rotorcraft Project Director.

LEAFLET 309/1

PICKETING, TIE-DOWN AND RAPID SECURING SYSTEMS
RECOMMENDATIONS FOR PICKETING AND TIE-DOWN

1 INTRODUCTION

1.1 This Leaflet gives recommendations relating to the requirements of Chapter 309.

1.2 Chocks and brakes are normally used during picketing but their contribution should not be taken into account.

1.3 Tie-down for ground running of engines will not be required in many cases but may be specified for certain rotorcraft designs.

2 FORCES TO BE CONSIDERED

2.1 SIDE AND DRAG FORCES

2.1.1 During picketing, side forces, drag forces and torque caused by winds will be present on the rotorcraft. The effects of loads due to ship motion must also be considered. For calculations, the basic mass of the rotorcraft should be used together with a range of coefficients of friction from 0.1 to 0.8.

2.2 SHIP MOTION AND WIND FORCES

2.2.1 The most adverse combination of ship motion and wind forces should be considered. It may be assumed that provided the flight deck securing requirements are met, then the hangar case will be automatically met, since there are no wind forces and usually less ship motion in the hangar. The values of ship motion and wind forces contained in Chapter 309, para 2.3.1 are considered to be acting in the most adverse weather conditions. The forces imposed on secured rotorcraft, by the relative wind-overdeck, should be added to those resulting from ship motions.

2.3 TIE-DOWN

2.3.1 Forces similar to those described above, and of appropriate magnitude, should be considered for the tie-down case, when specified, together with the lift force associated with engine running rotors turning.

3 PICKETING AND SECURING POINTS

AL/2

3.1 To meet the requirements of Chapter 309 para 2.1.1, it is desirable to have a picketing point or points on the rotorcraft structure close to each under-carriage unit. Where this is not practical, a single point on each unit is acceptable provided that picketing can still be achieved in accordance with the picketing plan. Additional picketing points not associated with the rotorcraft back-up structure for the undercarriage are acceptable.

3.2 Picketing and securing points should be either built in to the structure of the rotorcraft or the undercarriage or take the form of detachable fittings, in which case provision for stowage in the rotorcraft is required.

LEAFLET 309/2

PICKETING, TIE-DOWN AND RAPID SECURING SYSTEMS

RAPID SECURING SYSTEMS

1 INTRODUCTION

1.1 This Leaflet gives recommendations and background information for the design of rotorcraft for rapid securing aboard ship, and for the design of rapid securing systems, in support of the requirements given in Chapter 309 para 2.6.

1.2 The principal purpose of a rapid securing system is to enhance the safety of a rotorcraft while it is on a ship's flight-deck and, in particular, just before take-off and immediately after alighting. The secondary purpose is to minimise, and if possible eliminate, the need for groundcrew to enter the alighting area while the rotors are turning. To achieve these purposes, a rapid securing system should be designed to be effective in all practical combinations of rotorcraft mass and configuration, ship motion and climatic conditions. It is also necessary for the securing system to be capable of acting as a picketing device on its own or in association with other systems (e.g., brakes and steering) in Sea States 1, 2 and 3.

1.3 By using a wide-track undercarriage and low cg position, a rotorcraft can achieve some resistance to sliding or toppling when operating from a moving deck, but with the layout of many designs it is not possible to achieve these features. By introducing a rapid secure system, either the same deck operating capability can be maintained with a narrower track and/or a high position of the rotorcraft cg., or, with a given geometry, it should be possible to extend the capability for deck operations in worsening conditions of wind and deck motion.

2 DECK CONSIDERATIONS

2.1 There is a minimum size of grid for a particular ship/rotorcraft combination. This depends on pilot view of the deck and rotorcraft handling qualities; it also depends on the wind-overdeck and motion characteristics of the ship. For Lynx/frigate, for example, this was 8 ft diameter.

2.2 If it is a feature of the securing system that part of the mechanism moves along the deck with the rotorcraft attached, as the latter is stowed or ranged, then the design of this part of the mechanism should be such that the deck is not damaged.

2.3 The deck grid should be strong enough to withstand without permanent deflection the load applied when alighting at ultimate V_V.

CHAPTER 310

DESIGN OF UNDERCARRIAGES - WHEELS, TYRES,

BRAKES AND BRAKING SYSTEMS

1 INTRODUCTION AND GENERAL DESIGN REQUIREMENTS

1.1 SCOPE

1.1.1 The requirements of this chapter are applicable to undercarriage wheels, tyres, brakes, and braking systems. It is divided into 3 main sections:

(i) Brakes and Braking Systems.

(ii) Wheel and Brake Assemblies, Tyres, and Tubes.

(iii) Rotorcraft Performance and Compatibility Tests.

1.2 LIFE

1.2.1 It is recognised that not all components of these systems need to have a life equal to that of the rotorcraft; therefore, estimated lives of all components, including shelf, wear, and fatigue lives, shall be agreed with the Rotorcraft Project Director.

1.3 TESTS

1.3.1 Prior to tests on the rotorcraft (para 4 below), brake dynamometer tests and/or rig tests shall be conducted if required either by the rotorcraft specification or by the Rotorcraft Designer.

1.3.2 The Rotorcraft Designer and the systems Designers shall mutually agree the extent of these tests and who shall perform them.

1.3.3 The parts to be tested and the schedule of tests shall be agreed between the Rotorcraft Designer and the braking systems Designers and submitted to the Rotorcraft Project Director.

1.3.4 The Test Conditions and Objectives of Table 1 are applicable to all tests to the extent agreed with the Rotorcraft Project Director.

1.4 INTERFACE REQUIREMENTS

1.4.1 Each system shall be designed to meet the interface requirements as defined for the rotorcraft. All critical dimensional limitations shall be agreed with the Rotorcraft Designer.

1.4.2 The following features shall be taken into account:

(i) Brake torque characteristics.

(ii) The brake metering system and its components.

(iii) Hydraulic flow requirements.

(iv) Rotorcraft and undercarriage dynamic characteristics, including shock absorber, brake, and tyre dynamics.

(v) Total rotorcraft stopping performance requirements.

(vi) Relevant characteristics of the tyres.

(vii) The rotorcraft electrical and electronic systems.

(viii) The operating environment of components.

1.5 RELIABILITY, FAULT AND FAILURE ANALYSIS

1.5.1 Quantitative reliability requirements shall be established for the system and its components consistent with the rotorcraft system requirements.

1.5.2 A failure mode and effects analysis shall be performed and submitted to the Rotorcraft Designer and Rotorcraft Project Director.

1.5.3 Failures whose effects cannot be adequately represented in the analysis alone, shall be simulated by appropriate tests.

1.5.4 It shall be verified that operational check-out procedures can be developed to detect all significant failures.

1.6 MAINTAINABILITY

1.6.1 Quantitative maintainability requirements shall be established for the system and its components, consistent with the rotorcraft system requirements. The requirements shall apply to maintenance in the planned maintenance and support environment. Factors to be considered shall include time, frequency, complexity, and their costs.

1.6.2 The assumptions and results shall be submitted to the Rotorcraft Designer and to the Rotorcraft Project Director for agreement and acceptance.

2 BRAKES AND BRAKING SYSTEMS

2.1 The Rotorcraft Specification shall state if a braking system is required, if a standby system is required and if a parking brake is required. Wheel locking systems (or sprag brakes) may be regarded as brakes for the purposes of these requirements.

2.2 OPERATION

2.2.1 If friction brakes are fitted, then the braking effect shall increase or decrease progressively as the force applied to the normal brake control is increased or decreased. There shall be no appreciable delay between operation of the cockpit control and operation of the brakes.

2.2.2 If a reserve power supply is provided, it shall be capable of operating the brake for a single stopping of the rotorcraft in the event of failure of the normal power supply. See para 2.4 for the required capacity of both normal and reserve power supplies.

2.2.3 If a standby braking system is required, it shall be capable of operating the brakes for a single stopping of the rotorcraft in the event of any single failure within the normal system (including, the power supply but excluding the brake unit).

2.2.4 All brakes shall become free after the control applying them has been taken off.

2.2.5 Variations in friction brake torque, of a cyclical or alternating nature, that are likely to cause structural damage shall not occur.

2.3 PARKING

2.3.1 If a parking brake is provided, it shall be capable, at any practical rotorcraft mass, of holding the rotorcraft on a 12° slope.

2.3.2 The parking brake shall be capable of remaining fully applied for at least 24 hours when all engines are stopped, when flying control locks have been applied, and no power is supplied from an outside source.

2.3.3 The parking brake control system shall be in accordance with current Defence Standards.

2.3.4 The parking brake system may be incorporated in either the normal or the reserve brake system.

2·4 CAPACITY OF POWER SUPPLY

2.4.1 The capacity of the normal power supply with main or auxiliary power units working and the capacity of the reserve power supply with no main or auxiliary power units working, shall each be sufficient for at least 10 full applications of the brakes.

2.4.2 The requirements of para 2.4.1 shall be met both when the brakes are new and when they have reached the limit of their allowable wear and adjustment. Compliance with this requirement shall not be affected by the operation of, or faults in, any other service.

2·5 KINETIC ENERGY ABSORPTION

2.5.1 Where wheel brakes are fitted, they shall be capable of absorbing, in the appropriate distance at sea level and such other altitudes and temperatures which the Rotorcraft Specification requires, the kinetic energy of the rotorcraft in the test case of Table 1 of this Chapter after allowance has been made for any operating conditions within the Rotorcraft Specification which affect braking performance, e.g., aerodynamic lift, drag, rotor and engine thrust, taxying and previous repeated brake operations.

2.5.2 Compliance with Table 1 of this Chapter shall be demonstrated on the prototype brake assembly by adequate laboratory tests on the general lines of those given in Leaflet 310/2.

2.6 PRESSURE INDICATION

2.6.1 Means shall be provided to indicate satisfactory brake system pressure, main and reserve, as agreed with the Rotorcraft Project Director.

2.6.2 Hydraulic pressure gauges or indicators shall be so installed or protected that failure of the gauge or indicator causes minimal loss of hydraulic fluid from the system.

2.6.3 Provision shall be made for fitting a pressure gauge at the brake for ground testing.

2.7 REPRESSURISING WITHOUT ENGINE POWER

2.7.1 Provision shall be made for simple repressurising of the brake system reservoirs without running the rotorcraft engines, to assist ground manoeuvring.

2.8 STRENGTH OF CONTROL CIRCUIT

2.8.1 The brake control circuit shall have proof and ultimate factors not less than 1.125 and 1.5 respectively under the appropriate loads of Leaflet 310/1 para 7.1.

3 WHEEL AND BRAKE ASSEMBLIES, TYRES AND TUBES

3.1 GENERAL REQUIREMENTS

3.1.1 An analysis of estimated wheel loads which affect static and fatigue strength shall be supplied by the Rotorcraft Designer for the wheel and brake designers. The wheel shall be designed to withstand the stresses resulting from these loads and to have the specified life when operating to this spectrum of loads (see Leaflet 310/3 para 5).

3.1.2 When compiling the fatigue spectrum, account shall be taken of the tyre pressure cycles induced by temperature effects.

3.1.3 For multi-wheel undercarriage units, account shall be taken of unequal wheel load distribution and flat tyres (refer to Chapter 301 para 4).

3.2 WHEEL DESIGN

3.2.1 The design of the wheel shall be of the demountable flange or divided hub type to facilitate tyre change. All demountable flanges shall be secured to the wheel so as to prevent loss of the flange if a flat tyre occurs when the wheel is rolling.

3.2.2 Mating tubeless wheel parts and valves shall be sealed to prevent leakage.

3.2.3 The wheel shall be designed for satisfactory operation at all ambient temperatures specified by the Rotorcraft Designer. The maximum tyre bead seat temperature under all rotorcraft operating conditions other than the rejected take-off shall be specified and agreed with the Rotorcraft Designer.

3.2.4 The wheel rim contour and form shall conform to European Tyre and Rim Technical Organisation (ETRTO) Standards.

3.2.5 The wheel shall be designed to ensure clearance with the rotorcraft structure and all non-rotating brake parts under all operating conditions (see BS M 45).

3.2.6 Consideration shall be given to the fitting of fusible plugs in braked wheels to relieve tyre pressure if the temperature of the tyre bead seat, or of any critical pressurised areas of the wheel, reaches a predetermined level during abnormal taxying or braking conditions. The fusible plugs shall not operate and shall not leak under normal service conditions.

3.2.7 Consideration shall be given to the need for wheel balancing. If balancing is considered necessary, split wheels shall be balanced separately so that pairing of halves in alternative positions, or pairing of different halves, will not cause the limits to be exceeded.

3.2.8 A pressure relief device shall be fitted to all rotorcraft wheels to prevent tyre over inflation by the ground crew. The type of device shall be agreed with the Rotorcraft Project Director, but a pressure release plug incorporating a fusion welded bursting disc is preferred. The device shall be designed and located on the wheel so that the ground crew inflating the tyre will not be injured by fracture particles when the disc bursts. The minimum pressure at which the bursting disc will operate shall be selected by the rotorcraft designer, and shall include an allowance for the acceptable transient tyre pressure increases to be expected in service, for example, due to environmental conditions and those caused by brake heating and landing load effects. A recommended bursting pressure is twice the normal tyre inflation pressure. Pressure relief devices of different pressure ratings shall be designed so that they are not physically interchangeable.

3.3 TYRES AND TUBES

3.3.1 The tyre manufacturer shall perform tests to a schedule agreed with the Rotorcraft Designer involving performance, environment, fatigue, and endurance.

3.3.2 Tubes, if fitted, are regarded as part of the tyre for all design and test requirements.

3.3.3 Tyre inflation valves shall comply with the requirements of BS C7 and shall have a high performance core[1].

3.3.4 The Rotorcraft Specification will state requirements for retreadable tyres if any.

4 ROTORCRAFT PERFORMANCE AND COMPATIBILITY TESTS

4.1 GENERAL

4.1.1 Performance and compatibility tests shall be done on the rotorcraft. The method of measuring performance efficiency and the level of system performance efficiency required shall be agreed by the Rotorcraft Designer and the Rotorcraft Project Director.

4.1.2 Suitable instrumentation shall be provided.

4.1.3 The tests shall show that the requirements of Table 1 of this Chapter have been met.

4.2 TESTS

4.2.1 Taxying trials shall be done on wet and dry surfaces to determine that:

(i) braking is progressive and satisfactory,

(ii) the requirements of Table 1 have been met,

(iii) the rotorcraft has acceptable dynamic characteristics.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	25/8	3209

TABLE 1

KINETIC ENERGY ABSORPTION - TEST CONDITIONS AND REQUIREMENTS
(Chapter 310 para 2)

Mass	Speed	Brake Power Supply	Cross Wind (Knots)	Air Temperature °C (See Note)	Requirements
M_T	V_A (see Leaflet 301/3)	Normal or Reserve	As specified in the Rotorcraft Specification or see Table 3 of Chap 304	As specified in the Rotorcraft Specification and in Chap 101	Any damage shall be limited to the wheels, brakes, tyres and tubes

Note: Due allowance shall also be made for the effect of solar radiation if relevant - See Chapter 101.

LEAFLET 310/1

DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS
DESIGN OF WHEEL BRAKE SYSTEMS

1 INTRODUCTION

1.1 This Leaflet contains explanations and recommendations relating to the requirements of Chapter 310.

2 ENERGY ABSORPTION CAPACITY

2.1 In the Case of Table 1 of Chapter 310, the rotorcraft mass and the distance over which the kinetic energy of the rotorcraft is to be absorbed will normally be given in the brake specification in the form of Load/Speed/Time characteristics, which have been calculated in order to achieve the requirements of the Rotorcraft Specification.

3 BRAKE SYSTEM CONTROL

3.1 For friction brakes, the characteristics of the brake-control system should be, if possible, as shown in Fig.1. Brake application should follow the line L_A and release should remain above the maximum lag line L_R as shown.

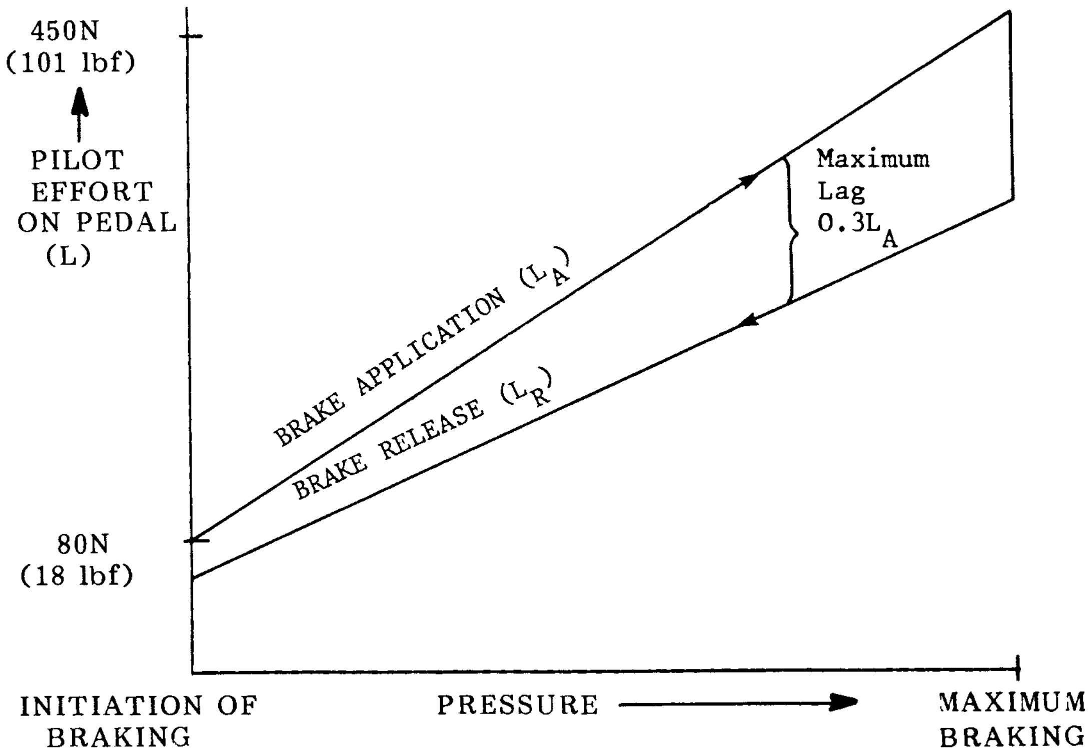

FIG.1 BRAKE CONTROL CHARACTERISTICS

4 BRAKING FORCE/TORQUE

4.1 The requirement of Chapter 310 para 2.4 sets a minimum for the static braking force which the brakes must be capable of applying. Consideration should also be given to the maximum torque which can be developed by the brakes without skidding the tyres. If the peak torque during any braking condition, within the rotorcraft speed and brake pressure range, exceeds the product of 0.8 times the static vertical load on each braked wheel at maximum design gross take-off mass times the static radius of the tyre, this should be discussed with the rotorcraft Designer. The effects on the nose wheel should also be considered.

5 PARKING AND GROUND MANOEUVRING

5.1 If brakes are fitted, it should be possible to lock them on for parking immediately after use without risk of damage due to heat but where this is inadvisable, suitable instructions should be given in the Aircrew Manual for the particular rotorcraft.

6 CONTROL FORCES

6.1 Table 1 of this Leaflet contains a summary of recommended values for the control forces under different conditions. These values should be regarded only as a guide since the position of the brake control, the range of travel and the mode of operation, all of which affect the control forces, vary considerably between different rotorcraft. Wherever possible, the best values should be determined by tests in the individual rotorcraft.

6.2 The initial operation of the brake should require sufficient effort to give the pilot the feeling that he is operating the control, while the force required to obtain maximum braking should be within the capabilities of the weakest pilot - see Table 1 of this Leaflet.

6.3 It is recommended that a pilot's brake pedal, if fitted, should be positioned such that the load is applied by the ball of the foot. Consideration should also be given to ensure that brake pressure is not applied inadvertently during yaw control operation for all pedal adjustment positions.

7 BRAKE CIRCUIT STRENGTH

7.1 A foot load of 1800 N (405 lbf) is the minimum to which all brake pedal circuits should be designed. It is not the maximum load that a pilot could apply when exerting his full strength under the optimum conditions of cockpit geometry. Such conditions are obtained when the line of action of the pedal requires a straight push with no appreciable ankle movement. At the same time, the leg becomes almost straight at full pedal deflection and the seat and shoulder harness must be able to provide efficient body restraint. Where such conditions apply, it is recommended that a load of 3,000 N (675 lbf) be taken as the maximum design load on each pedal singly in the asymmetric braking case and a load of 2,400 N (540 lbf) applied to each pedal simultaneously in the symmetric braking case. Whatever design pedal load is assumed, its point of application, while consistent with the foregoing, should be taken as that which would produce the highest stresses in the pedal and linkage.

7.2 The loads of para 7.1 are unfactored design maxima and do not compare with the loads given in Fig.1 and Table 1 of this Leaflet. See Chapter 310 para 2.8.1 for factors required.

TABLE 1

CONTROL FORCES FOR OPERATION

Type of Control	Recommended force to produce initial operation of the brakes	Recommended force to produce maximum braking effort	Estimated maximum force that can be exerted for 1 sec by the weakest pilot assuming the most disadvantageous position of the control
Toe Pedals	70 to 90N (15.7 to 20.2 lbf) per pedal	450N (101 lbf) per pedal	850N (191 lbf) per pedal
Hand Control (Squeeze type)	25 to 50N (5.6 to 11.2 lbf)	180N (40.3 lbf)	300N (67.2 lbf)

LEAFLET 310/2

DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS LABORATORY TESTING OF WHEEL BRAKE SYSTEMS

1 INTRODUCTION

1.1 This Leaflet gives details of laboratory tests suitable for demonstrating the ability of prototype wheel brakes to meet the requirements of Chapter 310.

2 OPERATING PRESSURE

2.1 Some specific tests may be required with the brakes in the unused condition. However, in order to determine the operating pressures (or loads for unpowered mechanical systems) required for subsequent tests, some running-in stops at low energy are permissible.

3 STATIC TESTS

3.1 At 0.25, 0.5, 0.75 and 1.0 times the maximum pressure, the static torque should be measured with the brake cold and also at the maximum temperature appropriate to a normal energy stop. At least four recordings of torque should be taken at each pressure, two with pressure increasing and two with pressure decreasing.

3.2 If the design of the brake is such that the torque in the reverse sense differs from that in the normal sense, the reverse torque should be measured as at para 3.1 above and should have similar characteristics, with the same limits on the design value.

4 DYNAMIC TESTS

4.1 TEST CONDITIONS

4.1.1 In the test case of Table 1 of Chapter 310, a record of the following should be made where relevant:

(i) Flywheel speed against time.

(ii) Brake pressure against time.

(iii) Brake torque against time.

(iv) Stopping time.

(v) Description, mass, and condition, of wheel, brake, tyre and tube.

(vi) Flywheel diameter, inertia equivalent, and kinetic energies.

(vii) Time/temperature relationships of: hydraulic fluid, wheel adjacent to fusible plug, bead-ledge above each brake, and other critical components.

(viii) Wheel load.

(ix) Fluid displacement due to brake wear and temperature changes.

(x) Time from stop to fusible plug release and energy level of stop.

(xi) Ability of tubeless tyre wheels to retain air satisfactorily under braking conditions.

4.1.2 The kinetic energy of the flywheel at the instant of brake application should be equal to the specified kinetic energy for the case under test. The rotational speed of the flywheel should be within ±10% of the corresponding equivalent speed for that case. These limits on speed may only be exceeded if the test machines available cannot be adjusted. Official concurrence should be obtained from the Rotorcraft Project Director in such cases.

4.1.3 The stopping time for each stop should be within ±15% of the required rotorcraft stopping time, and the instantaneous variation in torque within +50% to -20% of the required mean torque for the particular case.

4.2 KINETIC ENERGY ABSORPTION TEST (Chapter 310 Table 1)

4.2.1 Dynamometer tests should be conducted to demonstrate that the brakes will bring the rotorcraft to rest in accordance with the requirements of the Rotorcraft Specification. The brake is to be at ambient temperature at the start of any series of tests designed to demonstrate its durability, and artificial cooling between tests is acceptable.

4.2.2 The brake should be stripped at the end of these tests and thoroughly examined. The wear and any distortion of the friction surfaces should be measured and recorded.

4.3 ADDITIONAL TESTS

4.3.1 Consideration should be given to:

(i) structural torque tests (high pressure/high torque tests of wheel and brake),

(ii) functional and leakage tests (fluid compatibility and freedom from leakage at above normal pressure in static and dynamic conditions),

(iii) static pressure tests (no leakage or failure to a stipulated above normal pressure followed by pressure increase to failure),
(iv) endurance tests (number of pressure cycles specified by the Rotorcraft Designer to be endured without excessive leakage or malfunction),

(v) high and low ambient temperature tests,

(vi) simulated flight, taxy, and parking tests.

LEAFLET 310/3

DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS LABORATORY TESTING OF WHEELS

1 INTRODUCTION

1.1 This Leaflet gives details of laboratory tests suitable for demonstrating the ability of wheels to meet the requirements of Chapter 310.

1.2 The loads, pressures and tests discussed below should be agreed with the Rotorcraft Designer.

2 COMBINED LOAD TEST

2.1 Proof and ultimate tests should be based on the critical Design Combined Loads for both inboard and outboard directions of side load.

2.2 Consideration should be given to the necessity for proof and ultimate tests for both side-load-inboard and side-load-outboard conditions.

2.3 The load should be applied to the wheel through a tyre initially inflated to the wheel Design Maximum Inflation Pressure. Water may be used for the inflation. In this case the water should be bled off during loading to approximate the tyre deflection that would result if air were used. The inflation pressure at any stage should not exceed the pressure at maximum tyre deflection.

2.4 The corresponding brake assembly (where applicable) should be fitted during the proof test, to establish clearance under load.

2.5 Wheels intended to be fitted with tubeless tyres should be tested with tubeless tyres.

2.6 For the ultimate load test, bearing cones and rollers may be replaced by coned bushes. A tube may be fitted for this test if necessary.

3 RADIAL LOAD TEST

3.1 This test is only required if not covered by the Combined Load Test above.

3.2 Proof and ultimate tests should be based on the Design Radial Load.

3.3 The test procedures of paras 2.3, 2.4, 2.5 and 2.6 should be adopted.

4 BURST TEST

4.1 The wheel/tyre should be tested to establish the burst pressure of the combination and demonstrate that the requirements of Chapter 301 para 3 have been met. Water may be used for this test. The correct tyre and tube combination should be fitted.

5 ROLL TEST

5.1 The roll test should simulate a series of normal operations of the tyre and wheel assembly against a rotating flywheel to a programme representing as closely as possible the predicted usage of the wheel on the rotorcraft (see Chapter 310 paras 1.3 and 3.1).

5.2 Tyres should have the same size and ply rating as those to be fitted on the rotorcraft.

5.3 Before the roll test commences, wheels subjected to shot peening, roll burnishing, or any other such cold-working process, should be thermally conditioned to simulate the temperature/time history encountered during brake qualification tests excepting the rejected take-off test.

6 WHEEL PRESSURE RETENTION TEST (TUBELESS TYRES)

6.1 The tubeless tyre and wheel assembly should be inflated to 1.5 times the wheel Design Pressure and immersed in water. The leakage rate from the wheel should not exceed 4 bubbles per second.

6.2 After tyre growth has stabilised, the wheel should hold the wheel Design Inflation Pressure for 24 hours with no greater pressure drop than 5% or 5 psi whichever is the less.

6.3 The wheel assembly should be rolled under the design static load for 2.5 miles and then allowed to cool to the start temperature. The pressure difference between the pressure measured at the start and end of the test should not exceed 5% or 5 psi whichever is the less.

LEAFLET 310/4

DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS

TYRES

1 INTRODUCTION

1.1 This Leaflet contains recommendations relating to the requirements of Chapter 310.

2 INFORMATION REQUIRED

2.1 The following list contains the information which should be made available by the rotorcraft designer for the satisfactory development of a tyre:

1 Type of surface (paved, grass, matted).

2 If the tyre is on a braked wheel.

3 Tyre arrangement (single, twin or tandem, diabolo).

4 Rotorcraft take-off mass (M_T).

5 Date of latest issue of related documents.

6 List of referenced specifications to be used.

7 Addresses in UK where specifications may be obtained.

8 Statement regarding precedence of specifications.

9 Tyre performance:

(a) Rotorcraft design masses and speeds.

(b) Table of design tyre vertical, drag, side and torsional loads.

10 Tyre load/speed/time curves.

11 Envelope which the grown tyre is not to exceed.

12 Burst pressure (see Leaflet 310/3 para 4).

13 Tyre balance limitations.

14 Percentage increase in rotorcraft mass below which no redesign of tyre should be necessary.

15 Maximum allowable increase in tyre mass.

16 Environmental operating conditions:

(a) Maximum ground soak temperature operating.

(b) Maximum ground soak temperature non operating.

(c) Minimum ground soak temperature non operating.

(d) Minimum ground soak temperature operating.

(e) Minimum flight temperature sustained indefinitely at an altitude to be specified.

(f) Maximum flight temperature sustained for 2 minutes at an altitude to be specified.

(g) Maximum flight temperature at sea level and duration required.

17 Ambient pressures and rate of change ascending and descending.

18 Humidity specification to be met.

19 Contamination test specification and contaminants.

20 Vibration test specification.

21 Fungus resistance, salt, fog, rain, sand and dust, accelerator and icing test specifications.

22 Defect rate not to be exceeded.

23 Shelf life required (see Chapter 310 para 1.3).

24 Any special requirements for compatibility with wheels, and brakes.

25 Conductivity required.

26 Whether tubed or tubeless.

27 Cut resistance requirements.

28 Remouldability,

29 Identification and marking requirements.

30 Tyre contact pressure.

2.2 The following list contains the information which should be made available by the tyre designer to the rotorcraft designer and if necessary to the Rotorcraft Project Director:

1 Estimated tyre life.

2 Load/deflection/tyre pressure curves.

3 Maximum tyre mass.

4 Tread depth.

5 Maximum tyre deflection during normal operation. Reduction in tyre life if other deflections are used.

6 Load/deflection curves for both static and dynamic conditions at different pressures.

7 Vertical, lateral and fore and aft dynamic stiffnesses at relevant radial deflections.

8 Cornering power characteristics based on NASA TRR 64 for various loads and tyre pressures at relevant deflections up to a stated speed on dry and NATO Standard Good/Moderate wet runway conditions.

9 Tyre lateral rolling relaxation length characteristics based on NASA TRR 64 for speeds up to a stated maximum.

10 Moments of inertia for nose and main wheel tyres.

11 Pneumatic Trail.

PART 3 APPENDIX No. 1
STRUCTURAL STRENGTH AND DESIGN FOR OPERATION ON SPECIFIED SURFACES
MILITARY DERIVATIVES OF CIVIL ROTORCRAFT

1 INTRODUCTION

1.1 This Appendix covers requirements for structural strength and design for operation on specified surfaces, for military derivatives of civil rotorcraft.

1.2 Each Appendix paragraph consists of a table and amplifying notes comparing military and civil requirements for individual chapters from Part 3 of DEF STAN 00-970 Volume 2. In general, only those parts of the DEF STAN in which military and civil requirements differ are included in the Appendix and where no reference is made to DEF STAN chapter/paragraph numbers it is considered that the military and civil requirements are compatible and adequately covered by the military requirements (unless explained otherwise in the amplifying notes).

1.3 The issue or change number of the civil airworthiness requirements referred to in this Appendix are recorded in paragraph 9.5 of the main introduction to DEF STAN 00-970 Volume 2.

2 COMPLIANCE CHECKS/ASSESSMENT

2.1 Throughout this Appendix reference is made to the need for 'Compliance Checks or Assessments' of the existing civil rotorcraft against the military requirements of DEF STAN 00-970. A full explanation of the aims of compliance checks or assessments is given in paragraph 9 of the main Introduction to DEF STAN 00-970 Volume 2.

3 DESIGN OF UNDERCARRIAGES-OPERATIONAL REQUIREMENTS

3.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 300	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2 Basic Operational Requirements	3-5	231 235 471 473	231 235 471 473 (applies to FAR'S 27, 29 and JAR'S 27, 29)			

DEF STAN 00-970 CHAPTER 300 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
3. Operation form Specified Surfaces	2-7 2-2 3-5 7,3 4-10	231 235 473	231 235 473 521	231 235 473 519 521	231 235 473 521	231 235 473 591 521
4. Wind Conditions	3-2,2	143 237 485	143 485	143 485	143 485	143 485
5. Sea Conditions	3-5,7 4-10	--	521	519 521	521	519 521
6. Steering and Castoring	--	--	--	--	--	--
7. Braking	4-5,7	493 497 735	493 497 735	493 497 735	493 497 735	493 497 735
8. Undercarriage Retraction Selection and Indication	4-5,4	729	729	729	729	729
Table 1: Sea State Code (World Meteorological Organisation)	--	--	--	--	--	--
Table 2: Operation on Various Surfaces Applicable Requirements	--	--	--	--	--	--

3.2 The requirements for operating conditions of the military derivative are more severe than civil requirements. The military derivatives specification will specify the conditions severity.

3.3 The civil rotorcraft model specification will need to be checked for steering and castoring requirements.

4 DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS

4.1 MILITARY AND CIVIL REQUIREMENTS

<table>
<tr><th rowspan="2">DEF STAN 00-970
CHAPTER 301

Para Item</th><th colspan="2">BCAR'S

Section</th><th colspan="2">FAR'S</th><th colspan="2">JAR'S</th></tr>
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<tr><td>4. Twin Wheel
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<tr><td>5. Shimming</td><td>--</td><td>--</td><td>--</td><td>--</td><td>--</td><td>--</td></tr>
<tr><td>6. Detail Stressing
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521</td><td>471
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521</td></tr>
<tr><td>7. Tests</td><td>4-5</td><td>723
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727</td><td colspan="4">723
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727</td></tr>
<tr><td>8. Skids, Skis or
Floats on Land</td><td>2-7</td><td>501
505</td><td colspan="4">501
505</td></tr>
<tr><td>9. Floats on Water</td><td>2-7
6-12</td><td>521
563</td><td colspan="4">521
563</td></tr>
<tr><td>10. Structural Spring
Undercarriages</td><td>4-5</td><td>473
475</td><td colspan="4">473
475</td></tr>
</table>

DEF STAN 00-970 CHAPTER 301 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
11. Ground Resonance	2-10,6 3-9 3-13	241 552	241			
12. Auto-rotation and Alighting Following Engine Failure	4-9	691	691			
13. Fatigue	3-1,5 3-4,3	571	571			

4.2 A military derivative will have been designed to comply with civil requirements for shimmy or related dynamic instabilities of the undercarriage units. A compliance check to assess the derivative's suitability in its proposed military role will be necessary.

4.3 The tests specified under Chapter 301 paragraph 7 will, in addition, need to be considered for military derivatives, the programme being agreed by the Rotorcraft Project Director.

4.4 For notes on proof and ultimate factors see DEF STAN 00-970 Part 2 Appendix No. 2 paragraph 3.2.

4.5 Refer to DEF STAN 00-970 Part 4 Appendix No. 1 paragraph 2.2 for note on material properties, processing and protection.

5 DESIGN OF UNDERCARRIAGES - HANDLING QUALITIES, STRENGTH AND STIFFNESS ON SPECIFIED SURFACES

5.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 302 Para Item	BCAR'S Section G	 29	FAR'S 27	 29	JAR'S 27	 29
1. Introduction	--	--	--	--	--	--
2. Handling Qualities	2-7	231 235	231 235	231 235	231 235	231 235

DEF STAN 00-970 CHAPTER 302	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
3. Directional Control Requirements for Wheeled Steering and Castoring Units	2-7	231 235 237 493	231 235 493	231 235 493	231 235 493	231 235 493
4. Strength and Stiffness	3-1,4 3-5	235 237 303 305 307	235 303 305 307 309	235 303 305 307 309	235 303 305 307 309	235 303 305 307 309

5.2 The military handling qualities are defined in more detail than the civil requirements the operational limitation for the civil rotorcraft will need to be assessed against the military requirements and specification, particularly if operation is required from the deck of ships in sea states higher than 3 (reference Table 1 of DEF STAN 00-970, Chapter 300 Volume 2). The limitations established are to be agreed with the Rotorcraft Project Director.

5.3 DEF STAN 00-970 Chapter 302 paragraphs 4.3 and 4.4 require dynamic analysis and tests for rough surfaces including Damaged or Repaired Runways.

5.4 For notes on proof and ultimate factors see DEF STAN 00-970 Volume 2 Part 2 Appendix 1 paragraph 3.2.

6 DESIGN OF UNDERCARRIAGES - WHEEL STEERING

6.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 303	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. Introduction	1-2, 7.2.5	235 251	235 251	235 251	235 251	235 251
2. Steering Systems Design Requirements	2-7	235 237	235	235	235	235

<table>
<tr><th>DEF STAN 00-970
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Para Item</th><th colspan="2">BCAR'S

Section
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<tr><td>3. Steering Requirements</td><td>2-7</td><td>235
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<tr><td>4. Steering Torque</td><td>2-7</td><td>235
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<tr><td>5. Strength and Stiffness</td><td>3-1,4
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<tr><td>6. Tests</td><td>2-7
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& 5</td><td>571
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727</td></tr>
</table>

6.2 A military derivative will have been designed to suit civil operational requirements. In its proposed military role it is probable that the fatigue loading characteristics will be more severe. In the case of a new rotorcraft it will be necessary to perform a detailed comparison between civil and military requirements and establish safe life and recording techniques, etc., appropriate to its military role. Where the rotorcraft has been in civil operation, a similar detail comparison to establish the equivalent military fatigue life still remaining must be made (see Part 2 Appendix 1 paragraph 4 for further details).

6.3 The strength and stiffness requirements are more severe for military rotorcraft than they are for civil. It will be necessary to carry out an investigation into reserve factors available on permitted strength values for the civil rotorcraft design, as part of its assessment into the suitability for military operations. The major differences being the 1.0 to 1.125 proof factor requirements (see Part 2 Appendix 1 paragraph 3.2 for further details).

7 DESIGN OF UNDERCARRIAGES - ALIGHTING

7.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 304 Para Item	BCAR'S Section G	BCAR'S 29	FAR'S 27	FAR'S 29	JAR'S 27	JAR'S 29
1. Introduction - General Requirements	2-7,2 3-5,2 & 3 4-5,4	75 471 473 493	75 471 473 493			
2. Preliminary Stressing Cases - All Projects	3-5	473 479 to 512 519 521	473 479 to 511 519 521			
3. Dynamic Analyses - All Projects	3-9,3 3-12	629 663 687	629 663 687			
4. Supplementary and Special Stressing Cases	3-13 3-5,7	-- --	-- --			
5. Energy Absorption and Dissipation	4-5,3 & App 3-9,3 3-12	687 727	687 727			
6. Undercarriage Stiffness	3-1,4	303 305 307	303 305 307			
7. Rotorcraft Static Strength	3-5 4-5	303 305 307 473 479 to 485	303 305 307 473 479 to 485			

(The FAR'S/JAR'S entries are printed once, centred under a brace spanning FAR'S 27, FAR'S 29, JAR'S 27 and JAR'S 29.)

DEF STAN 00-970 CHAPTER 304	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
8. Drop Tests	3-5,2 4-5	723 725 727	723 725 727			

7.2 A military derivative will have been designed to suit civil operations requirements; in its proposed military role it is probable that the civil design rates of descent will be exceeded. Therefore, for military derivatives it will be necessary to carry out a compliance check to assess the derivative's suitability for operation in its proposed military role.

7.3 The Stressing Cases and Factors required for military aeroplanes may differ from those for civil aeroplanes, hence it may be necessary to carry out a compliance check to assess the derivative's suitability for operations in its proposed military role.

7.4 The Energy Absorption and Dissipation requirements are more specific for military aeroplanes, in particular the energy dissipated by the shock absorber for the main units is specified to be not less than 67% of the total vertical energy of descent. There is no similar requirement for civil aeroplanes. Therefore, for military derivatives it will be necessary to carry out a compliance check to assess the derivative's suitability for operations in its proposed military role.

7.5 The undercarriage stiffness requirements for military aeroplanes are more demanding than those of civil aeroplanes because of the 1.125 to 1.5 proof to ultimate strength ratio i.e., 0.75. Although in this instance a proof factor of not less than 1.0 is acceptable. Therefore, for military derivatives it will be necessary to carry out a compliance check to assess the derivative's suitability for operations in its proposed military role. (See Part 2 Appendix 1 paragraph 3.2 for further details).

8 DESIGN OF UNDERCARRIAGES - VERTICAL AND NEAR-VERTICAL OPERATION ON ROUGH SURFACES

8.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 305	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
2. Speed Vector	--	--	--	--	--	--
3. Obstacles	--	--	--	--	--	--
4. Assessment	--	--	--	--	--	--

8.2 A civil rotorcraft may have been designed to operate from smooth hard runway surfaces only. A compliance check to assess the derivatives suitability for operation from surfaces other than smooth hard runways, in accordance with the requirements of DEF STAN 00-970 Volume 2, Chapter 305 will be necessary.

9 DESIGN OF UNDERCARRIAGES - RETRACTION AND LOWERING

9.1 MILITARY AND CIVIL REQUIREMENTS

<table>
<tr><th>DEF STAN 00-970 CHAPTER 306</th><th colspan="2">BCAR'S</th><th colspan="2">FAR'S</th><th colspan="2">JAR'S</th></tr>
<tr><th>Para Item</th><th>Section G</th><th>29</th><th>27</th><th>29</th><th>27</th><th>29</th></tr>
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<tr><td>7. Tests</td><td>4-5
4.6</td><td>729</td><td colspan="4">729</td></tr>
</table>

9.2 The military requirement for the retraction and lowering of the undercarriage is that it shall be completed in a specified time, this being 10 and 5 seconds respectively. There is no equivalent civil requirement in either BCAR's, FAR's and JAR's. The time for lowering is, instead, stated in the aeroplane's maintenance manual. Hence, for a military derivative it will be necessary to check the derivative's lowering and retraction times for compliance by referring to its maintenance manual.

9.3 The military strength requirements for undercarriage and doors and their operating mechanisms is that they shall have proof and ultimate factors of not less than 1.5 and 2.0 respectively. The corresponding civil requirements being a proof of 1.0 and ultimate 1.5. It will be necessary to carry out an investigation into reserve factors available on permitted strength values for the civil aeroplane design, which would be required as part of its assessment into the suitability for military operations.

9.4 The test requirements for retractable undercarriage units are similar for both civil and military aeroplanes, although in the latter case the requirements are more detailed. It will therefore be necessary to carry out a compliance check against the more detailed military requirements to assess the derivative's suitability.

10 CRASH LANDING AND DITCHING

10.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 307	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
1. General	4-3	561 801 803	561 801 807	561 801 803	561 801 807	561 801 803
2. Ditching	3-5 3.2 5 6.6.3 3-8 4-10	561 563 801 to 813	561 562 563 801 807	561 562 563 801 to 813	561 562 563 801 to 807	561 562 563 801 813
3. Seats, Harnesses, Equipment and Local Attachments	3-8 4-4	561 563 785	2 561 562 563 785	2 561 562 563 785	561 562 563 785	561 562 563 785

DEF STAN 00-970 CHAPTER 307 Para Item	BCAR'S Section G	BCAR'S 29	FAR'S 27	FAR'S 29	JAR'S 27	JAR'S 29
4. Post Crash Fire Hazard	5-8,3	853 to 863	853 to 863	851 to 863	853 to 863	851 to 863
5. Validation of Design	3-1,4 3-8 404 4-3,5	305 307 561 563 785 1413	305 307 561 562 563 785 1413	305 307 561 562 563 785 1413	305 307 561 562 563 785 1413	305 307 561 562 563 785 1413

10.2 The military requirements for impulse duration and size are specified in paragraph 1.7 of DEF STAN 00-970, Chapter 307. Emergency landing dynamic conditions are specified in FAR 23, 25 and JAR 23 and 25 paragraph 562 but not in BCAR Sections G and 29. A detail design compliance check will be required to assess the derivatives suitability for use in its proposed military role.

10.3 The military requirement states that the Rotorcraft Specification will give a minimum flotation time or crew evacuation time. The corresponding civil requirement would be included in the flight manual of the derivative.

10.4 For military aeroplanes where the flotation or buoyancy and stability are unsatisfactory for the basic aeroplane, flotation aids may be used to enhance the required characteristics. Therefore it will be necessary for the military derivative to be checked to ensure it complies with the military requirements for flotation etc., and if necessary provision made for the fitting of flotation aids, if agreed by the Rotorcraft Project Director.

10.5 Differing requirements for accelerations and strength factors for crew seats and local fittings, a compliance check to establish the derivatives suitability for use in its proposed military roles. However, the inertia factors required for passenger seats and local fittings for military aircraft Categories II and III are generally similar to the listed civil requirements.

10.6 There is no equivalent civil requirement which specifies the strength of hand grips which are to be used in the event of a crash landing. It will be necessary to carry out a compliance check on the military derivative to establish the strength characteristics of any existing handles.

10.7 The military requirement for the validation of design of the seat, energy absorbers and relevant structure, together with the restraint system and other items (paragraph 5), is that they shall be sled-tested under dynamic loading conditions. The civil requirement is that seats and berths etc., shall be of approved types. Therefore for military derivatives it will be necessary to carry out a compliance assessment of approved civil seats and berths etc., proposed for use in the aeroplanes proposed military role.

11 STRENGTH FOR GROUND HANDLING

11.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 308	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
Complete chapters	4-1,12		Appendix A, paragraph 3(a)(4)			

11.2 Ground Handling is not specifically identified in the civil airworthiness requirements except in general notes concerned with maintenance to satisfy the military requirements an analysis will need to be carried out.
Reference should then be made to the Civil Operations Manual to identify the ground handling provisions for review against the military requirements. The result of the review is to be advised to the Rotorcraft Project Director.

12 PICKETING , TIE DOWN AND RAPID SECURING SYSTEM. CHAPTER 309

12.1 MILITARY AND CIVIL REQUIREMENTS

There is no direct reference to these items in the civil airworthiness requirements. As for Chapter 308 reference should be made to the Civil Operations Manual to identify if such provision has been made and then review accordingly. The results of the review to be advised to the Rotorcraft Project Director.

13 DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS

13.1 MILITARY AND CIVIL REQUIREMENTS

DEF STAN 00-970 CHAPTER 310	BCAR'S		FAR'S		JAR'S	
Para Item	Section G	29	27	29	27	29
Complete Chapter	4-5	723 of 37	723 to 737			

13.2 Military operations are more severe than the civil counterpart. The items in this Chapter will need to be reviewed against the military derivative specification.

PART 3 APPENDIX No. 2
STRUCTURAL STRENGTH AND DESIGN FOR OPERATION ON SPECIFIED SURFACES
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 300: DESIGN OF UNDERCARRIAGES - OPERATIONAL REQUIREMENTS

300	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-1290	LIGHT FIXED-AND ROTARY-WING AIRCRAFT CRASH RESISTANCE
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-B-8584	BRAKE SYSTEMS, WHEEL, AIRCRAFT, DESIGN OF
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8812	STEERING SYSTEMS: AIRCRAFT GENERAL REQUIREMENTS FOR
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY. GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

300 1.	MIL- D -23222A	PARA: 3.15.1

2. BASIC OPERATIONAL REQUIREMENTS

300 2.	MIL-STD-1290	PARA: 5.1.6
	MIL-STD-1472C	PARA: 4.4
	MIL-S-8698	PARA: 3.4
	MIL-F-83300	PARA: 3.1.10.3.1, 3.8.9.2.2, 3.8.10.2
	MIL-L-87139	

3. OPERATION FROM SPECIFIED SURFACES

300 3.	MIL-H -8501A	PARA: 3.5
	MIL-T -8679	PARA: 3
	MIL-A-8863B	PARA 3.9

4. WIND CONDITIONS

300 4.	MIL-H -8501A	PARA: 3.5
	MIL-S-8698	PARA: 3.4.6.2
	MIL-D-23222A	TABLE 3
	MIL-F-83300	PARA: 3.6.5, 3.6.7

5. SEA CONDITIONS
300 5.

6. STEERING AND CASTORING
300 6. MIL-S-8812

7. BRAKING

300 7.	MIL-B-8584C	PARA: 1.1
	MIL-A-8863B	PARA: 3.12.1

8. UNDERCARRIAGE RETRACTION - SELECTION AND INDICATION

300 8.	MIL-STD-250D	PARA: 5.1.10.(a).(3), 5.1.10.(c).(3), 5.1.12

TABLE 1 SEA STATE CODE (WORLD METEOROLOGICAL ORGANISATION)

TABLE 2 OPERATION ON VARIOUS SURFACES. APPLICABLE REQUIREMENTS

CHAPTER 301: DESIGN OF UNDERCARRIAGES - GENERAL REQUIREMENTS

301	MIL-W-5013	WHEEL AND BRAKE ASSEMBLIES, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-T-6053	TESTS, IMPACT, SHOCK ABSORBER LANDING GEAR, AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-L-8552	LANDING GEAR, AIRCRAFT SHOCK ABSORBER (AIR-OIL TYPE)
	MIL-B-8584	BRAKE SYSTEMS, WHEEL, AIRCRAFT, DESIGN OF
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8812	STEERING SYSTEM: AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY. GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

301 1.

2. ROTORCRAFT DESIGN REQUIREMENTS

301 2.	MIL-W-5013K	PARA: 3.2.1
	MIL-L-8552C	PARA: 3.2, 3.2.6
	MIL-B-8584C	PARA: 3.1
	MIL-S-8698	PARA: 3.4.1, 3.4.5
	MIL-S-8812D	PARA: 3.3, 3.4
	MIL-F-83300	PARA: 3.6.5.3

3. WHEELS AND TYRES (WHEN FITTED)

301 3.

4. TWIN WHEEL AND MULTI-WHEEL UNITS

301 4.

5. SHIMMY

301 5.	MIL-H-8501A	PARA: 3.7.3
	MIL-S-8812D	PARA: 3.4.2.7
	MIL-D-23222A	PARA: 3.3.5.1

6. DETAILED STRESSING REQUIREMENTS APPLICABLE TO ALL CASES

301 6.	MIL-S-8698	PARA: 3.1.3.1, 3.4.5.1, 3.5
	MIL-A-8863B	PARA: 3.11.1, 3.12

7. TESTS

301 7.	MIL-W-5013	PARA: 4.3
	MIL-T-6053C	PARA: 3.3
	MIL-T-8679	PARA: 3.1.2, 3.1.8
	MIL-S-8812D	PARA: 3.1
	MIL-D-23222A	PARA: 3.3.5.1, 3.7.2

8. SKIDS, SKIS, OR FLOATS ON LAND

301 8.

9. FLOATS ON WATER

301 9.	MIL-H-8501A	PARA: 3.5.4.5

10. STRUCTURAL SPRING UNDERCARRIAGES

301 10.	MIL-L-8552C	PARA: 3.5
	MIL-S-8698	PARA: 3.1.3, 3.4.2

11. GROUND RESONANCE

301 11.	MIL-H-8501A	PARA: 3.7.3
	MIL-D-23222A	PARA: 3.3.5.1
	MIL-F-83300	PARA: 3.8.11

12. AUTOROTATION AND ALIGHTING FOLLOWING ENGINE FAILURE

301 12.	MIL-H-8501A	PARA: 3.5.5
	MIL-D-23222A	PARA: 3.3.5.2.1

13. FATIGUE

301 13.	MIL-S-8698	PARA: 3.1.9, 3.4.1

FIGURE 1 VARIATION OF COEFFICIENT OF SKIDDING FRICTION BETWEEN THE TYRE AND VARIOUS SURFACES WITH INSTANTANEOUS SLIP RATIO

301 FIG 1	MIL-A-8863B	FIGURE 4

CHAPTER 302: DESIGN OF UNDERCARRIAGES - HANDLING QUALITIES, STRENGTH AND STIFFNESS ON SPECIFIED SURFACES

302	MIL-L-8552	LANDING GEAR, AIRCRAFT SHOCK ABSORBER (AIR-OIL TYPE)
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8812	STEERING SYSTEM: AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY. GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

302 1.	MIL-L-8552C	PARA: 3.5.3
	MIL-T-8679	PARA: 3.3.1.1.1, 3.3.1.1.2
	MIL-S-8812D	PARA: 3.4.2.7, 4.5
	MIL-A-8863B	PARA: 3.9
	MIL-F-83300	PARA: 6.5.1
	MIL-L-87139	

2. HANDLING QUALITIES

302 2.	MIL-A-8863B	PARA: 3.10.2, 3.11
	MIL-F-83300	PARA: 1.5, 3.1.10.1, 3.1.10.3.1, 3.8.9.2.2

3. DIRECTIONAL CONTROL REQUIREMENTS FOR WHEELED STEERING AND CASTORING UNITS

302 3.	MIL-S-8812D	PARA: 3.4, 3.4.1, 3.4.2.7, 3.4.2.10, 3.5.1

4. STRENGTH AND STIFFNESS

302 4.	MIL-T-8679	PARA: 3.1.4
	MIL-S-8698	PARA: 3.1.2, 3.1.4, 3.4.1, 3.4.3, 3.4.5, 3.4.6
	MIL-A-8863B	PARA: 3.14.1

CHAPTER 303: DESIGN OF UNDERCARRIAGES - WHEEL STEERING

303	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8812	STEERING SYSTEM: AIRCRAFT, GENERAL REQUIREMENTS FOR

CONTROLLED DISTRIBUTION:

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION - GENERAL REQUIREMENTS

303 1.	MIL-H-8501A	PARA: 3.7.3
	MIL-S-8812D	PARA: 3.4.1.(a), 3.4.1.(c), 3.4.2.3.1, 3.4.2.5, 3.4.2.7, 3.4.2.10, 3.7.4
	MIL-L-87139	

2. STEERING SYSTEM DESIGN REQUIREMENTS

303 2.	MIL-S-8812D	PARA: 3.4.2.9, 3.4.2.10

3. STEERING REQUIREMENTS

303 3.	MIL-S-8812D	PARA: 3.4.2.3, 3.4.2.4, 3.5

4. STEERING TORQUE

303 4.	MIL-S-8812D	PARA: 3.4.2.1

5. STRENGTH AND STIFFNESS

303 5.	MIL-S-8698	PARA: 3.1.1, 3.1.9
	MIL-S-8812D	PARA: 3.4, 3.4.2.1, 3.4.7.3

6. TESTS

303 6.	MIL-S-8698	PARA: 3.4.6
	MIL-S-8812D	PARA: 3.4.2.2, 3.5.2, 4

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304	MIL-STD-1290	LIGHT FIXED-AND ROTARY-WING AIRCRAFT CRASH RESISTANCE
	MIL-T-6053	TESTS, IMPACT, SHOCK ABSORBER LANDING GEAR, AIRCRAFT
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTERS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION - GENERAL REQUIREMENTS

304 1.	MIL-L-87139	

2. PRELIMINARY STRESSING CASES - ALL PROJECTS

304 2.	MIL-STD-1290	PARA: 5.1.6
	MIL-T-6053C	PARA: 3.3.1, 3.3.2, 3.3.3
	MIL-S-8698	PARA: 3.1.4, 3.1.5, 3.1.6, 3.1.7, 3.4.2, 3.4.5
	MIL-F-83300	PARA: 3.8.10.2

3. DYNAMIC ANALYSIS - ALL PROJECTS

304 3.	MIL-S-8698	PARA: 3.6.3, 3.7.1
	MIL-F-83300	PARA: 3.8.10.2
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4. SUPPLEMENTARY AND SPECIAL STRESSING CASES

304 4.	MIL-T-6053C	PARA: 3.3

5. ENERGY ABSORPTION AND DISSIPATION

304 5.	MIL-STD-1290	PARA: 5.1.6
	MIL-T-6053C	PARA: 3.3.4

6. UNDERCARRIAGE STIFFNESS

304 6.	MIL-STD-1290	PARA: 5.1.6
	MIL-S-8698	PARA: 3.1.3.2

7. ROTORCRAFT STATIC STRENGTH

304 7.	MIL-S-8698	PARA: 3.1.1, 3.1.2, 3.4.1, 3.4.2, 3.4.3
	MIL-D-23222A	PARA: 3.7.1

8.	DROP TESTS		
304 8.	MIL-STD-1290	PARA: 10.4	
	MIL-T-6053C	PARA: 3.1, 3.3.1	
	MIL-T-8679	PARA: 3.1.10.7, 3.3, 3.3.1.1.2, 3.3.1.2, 3.3.1.3, 3.3.1.4	
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305	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

305 1.	MIL-S-8698	PARA: 3.4.5.1, 3.4.5.2
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	MIL-L-67139	

2. SPEED VECTORS

305 2.

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306	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-S-8812	STEERING SYSTEM: AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

306 1.	MIL-L-87139	

2. OPERATIONAL REQUIREMENTS

306 2.	MIL-F-83300	PARA: 3.8.9.2.2

3. DESIGN REQUIREMENTS

306 3.	MIL-STD-250D	PARA: 5.1.10.(c).3, 5.1.11
	MIL-S-8812D	PARA: 3.4.2.5

4. LOCKING

306 4.	MIL-S-8698	PARA: 3.1.3.3

5. UNDERCARRIAGE DOORS

306 5.	MIL-H-8501A	PARA: 3.7.3
	MIL-S-8698	PARA: 3.6.3
	MIL-F-83300	PARA: 3.8.11

6. STRENGTH

306 6.	MIL-S-8698	PARA: 3.1.1, 3.1.2, 3.1.3.3, 3.4.3
	MIL-D-23222A	PARA: 3.7.1

7. TESTS

306 7.	MIL-H-8501A	PARA: 3.1.10.7, 3.2.9.3.4
	MIL-D-23222A	PARA: 3.7.1.2, 3.7.2, 3.12.7.1

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	MIL-STD-1290	LIGHT FIXED-AND ROTARY-WING AIRCRAFT CRASH RESISTANCE
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	MIL-H -8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-B -8584	BRAKE SYSTEMS, WHEEL, AIRCRAFT, DESIGN OF
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	MIL-T -9166	TIE DOWN, CARGO AIRCRAFT, CABLE NET, 10,000 LB CAPACITY, TYPE A-2
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	MIL-T -27260	TIE DOWN, CARGO, AIRCRAFT
	MIL-T -27422	TANK, FUEL, CRASH - RESISTANT, AIRCRAFT
	MIL-H -46855	HUMAN ENGINEERING REQUIREMENTS FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-S -58095	SEAT SYSTEM: CRASH - RESISTANT, NON-EJECTION, AIRCREW, GENERAL SPECIFICATION FOR
	MIL-R-81729	RESTRAINT SYSTEMS AIRCREWMAN'S
	MIL-F -83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT
	MIL-F-87168	FIRE AND EXPLOSION HAZARD PROTECTION SYSTEMS, GENERAL SPECIFICATION FOR
	AFGS-87238	SURVIVAL AND FLOATATION SYSTEM, AIRBORNE, SPECIFICATION FOR

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	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. GENERAL

307 1.	MIL-STD-1290	PARA: 4.1, 4.2, 4.3, 5.1, 5.1.1.1, 5.1.1.3. 5.1.2.1, 5.1.6, 5.2, 5.4
	MIL-STD-1472C	PARA: 4.4. (b) , 4.4. (i) , 4.4. (o) , 4.5, 5.1.7, 5.13.1, 5.14.4.1.3, 5.14.4.2, 5.14.4.3.7, 5.14.4.3.8.3
	MIL-B-8584C	PARA: 3.2
	MIL-S-8698	PARA: 3.4.7
	MIL-F-83300	PARA: 3.8.9.2.3
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	MIL-STD-1472C	PARA: 5.14.3.4, 5.14.4.3.7, 5.14.4.3.8.4
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3. SEATS, HARNESSES, EQUIPMENT AND LOCAL ATTACHMENTS

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4. POST CRASH FIRE HAZARD

307 4.	MIL-STD-1290	PARA: 4.3, 5.2.1.2.2, 5.5, 5.5.1.1, 5.5.1.1.1, 5.5.1.2, 5.5.1.4, 5.5.2.4, 10
	MIL-STD-1472C	PARA: 5.13.4.1, 5.13.7.4.1
	MIL-D-23222A	PARA: 3.10.11, 3.12.3
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307 5.	MIL-STD-872	PARA: 4
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	MIL-D-23222A	PARA: 3.3.5.2.4, 3.7.2, 3.12.8.2, 3.12.11, 3.12.12
	MIL-H-46855B	PARA: 3.2.2.1.1
	MIL-S-58095A	PARA: 4.7

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308	MIL-STD-805	TOWING FITTINGS AND PROVISIONS FOR MILITARY AIRCRAFT, DESIGN REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY. GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

308 1.	MIL-D-23222A	PARA: 3.7.1, 3.12.8.3
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2. SLINGING

308 2.	MIL-S-8698	PARA: 3.4.6.3
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3. JACKING

308 3.	MIL-S-8698	PARA: 3.4.6.3
	MIL-A-8863B	PARA: 3.13.2

4. TOWING - WHEELED ROTORCRAFT

308 4.	MIL-STD-805	
	MIL-S-8698	PARA: 3.1.1, 3.4.6

5. TOWING STRESSING CASE - GENERAL

308 5.	MIL-STD-805	
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	MIL-A-8863B	PARA: 3.13.1

6. TOWING STRESSING CASE - WHEELED MAIN UNDERCARRIAGE UNITS

308 6.	MIL-STD-805	
	MIL-S-8698	PARA: 3.13.1
	MIL-A-8863	

7. TOWING STRESSING CASE - WHEELED AUXILIARY UNDERCARRIAGE UNITS

308 7.	MIL-STD-805	
	MIL-S-8698	PARA: 3.4.6
	MIL-A-8863B	PARA: 3.13.1

8. TOWING - STEERABLE UNDERCARRIAGE UNITS

308 8.	MIL-STD-805	
	MIL-S-8698	PARA: 3.4.6

9. GROUND HANDLING - SKID, SKI, AND FLOAT EQUIPPED ROTORCRAFT

308 9.	MIL-S-8698	PARA: 3.4.6
	MIL-A-8863B	PARA: 3.13.1

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309	MIL-B-8075	BRAKE CONTROL SYSTEMS, AIRCRAFT WHEELS, GENERAL SPECIFICATION FOR
	MIL-S--8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY, GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-A-8865	AIRPLANE STRENGTH AND RIGIDITY, MISCELLANEOUS LOADS
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION, AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-T-81259	TIE-DOWNS, AIRFRAME DESIGN, REQUIREMENTS FOR
	MIL-T-85075	TIE DOWN ASSEMBLY, HELICOPTER BLADE, GENERAL SPECIFICATION FOR

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1. INTRODUCTION

309 1.	MIL-D-23222A	PARA: 3.12.8.3
	MIL-T-31259A	PARA: 3.1

2. OPERATIONAL REQUIREMENTS

309 2.	MIL-S-8698	PARA: 3.4.6.1, 3.4.6.2
	MIL-F-9490D	PARA: 3.2.6.7.2, 3.2.6.7.2.1
	MIL-T-81259A	PARA: 3.1, 3.6, 4.2, 4.3.3.1

3. STRENGTH

309 3.	MIL-S-8698	PARA: 3.1.1, 3.1.2, 3.5
	MIL-A-8863B	PARA: 3.13.4
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	MIL-D-23222A	PARA: 3.7.1
	MIL-T-81259A	PARA: 3
	MIL-T-85075A	PARA: 4.6

CHAPTER 310: DESIGN OF UNDERCARRIAGES - WHEELS, TYRES, BRAKES AND BRAKING SYSTEMS

310	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-W-5013	WHEEL AND BRAKE ASSEMBLIES, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-B-8584	BRAKE SYSTEMS, WHEEL, AIRCRAFT, DESIGN OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION AND GENERAL DESIGN REQUIREMENTS

310 1.	MIL-W-5013K	PARA: 3.3.7, 3.3.14.1, 3.3.15, 3.3.17
	MIL-B-8584C	PARA: 3. 1, 3. 3, 3. 3. 2. 2. 1. (b) , 3. 3. 2. 3, 3. 3. 4. 2, 3. 3. 4. 3, 3. 3. 4. 4, 3. 3. 4. 6, 3. 3. 4. 7, 3. 3. 6, 3. 3. 15
	MIL-D-23222A	PARA: 3.17, 3.4.2.5
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2. BRAKES AND BRAKING SYSTEMS

310 2.	MIL-STD-250D	PARA: 5.1.13.(c), 5.1.14
	MIL-W-5013K	PARA: 3.3.7, 3.3.8
	MIL-B-8584C	PARA: 3.3.2, 3.3.2.3, 3.3.3, 3.3.4.1, 3.3.5.1. 3.3.7, 3.3.8, 3.3.11, 3.3.14.1, 3.3.14.2

3. WHEEL AND BRAKE ASSEMBLIES, TYRES AND TUBES

310 3.	MIL-B-8584C	PARA: 3.3.1
	MIL-W-5013K	PARA: 3.3

4. ROTORCRAFT PERFORMANCE AND COMPATIBILITY TESTS

310 4.	MIL-W-5013K	PARA: 3.3, 3.3.28, 4.3
	MIL-H-8501A	PARA: 3.7.3
	MIL-D-23222A	PARA: 3.7.2, 3.12.7.2

REFERENCES

310 REFERENCES	MIL-W-5013K	PARA: 6. 5

TABLE 1 KINETIC ENERGY ABSORPTION - TEST CONDITIONS AND REQUIREMENTS

310 TABLE 1	MIL-W-5013K	TABLES I, II

PART 4

DETAIL DESIGN AND STRENGTH OF MATERIALS

CONTENTS

* In Preparation

CHAPTER 400

GENERAL DETAIL DESIGN

1 GENERAL

1.1 In the detailed design of the Rotorcraft, the variety of materials and parts shall be kept to a minimum consistent with structural efficiency and without unduly sacrificing structure weight. The design of parts and fittings shall be such as to facilitate their manufacture in quantity.

1.2 Components and items of equipment shall have at least the same factor as the main structure in all the appropriate stressing cases. Local loads at parts of attachment to the main structure shall be carefully considered to ensure that the main structure is not unduly weakened thereby.

2 GRADING OF PARTS AND ASSEMBLIES

2.1 INTRODUCTION

2.1.1 The term "parts" relates to all parts of the Rotorcraft except missiles (DEF STAN 08-5), engines (DEF STAN 00-971) and propellers. The grading requirements also apply to the structural and mechanical parts of all systems and equipment in the Rotorcraft.

2.1.2 In order to ensure that the material and processes used in the manufacture of a part are of suitable quality and that the part is satisfactory, quality control and testing must be appropriate to the design requirements and the application of the part. To this end all parts, except standard parts as defined in Para. 3, shall be designated Grade A or Grade B, taking cognizance of strength and stiffness requirements as promulgated in this publication, quality requirements, maintainability and inspectability requirements, and also such factors as failure by leakage, malfunction, or other defect.

2.1.3 Standard Parts shall not be graded but shall be selected according to the grade of their application.

2.1.4 The grading requirements apply whether the part is designed to Damage Tolerance Requirements or not.

2.2 GRADING REQUIREMENTS

2.2.1 Grade A - A part shall be Grade A if the deformation or failure of the part would result in one or more of the following:

(i) structural collapse at loads up to and including the design ultimate load,

(ii) loss of control,

(iii) failure of motive power,

(iv) unintentional operation of, or inability to operate, any systems or equipment essential to the safety or operational function of the Rotorcraft,

(v) incapacitating injury to any occupant,

2.2.2 Grade A non-redundant parts subject to fatigue loading, whose failure would cause a catastrophe, shall be graded GRADE A - VITAL unless the probability of failure can be shown to be extremely remote (See Leaflet 400/1).

2.2.3 Grade B - A part may be Grade B at the designers discretion if none of the provisions of Para 2.2.1 apply.

2.2.4 Advice on the interpretation of these definitions and requirements is given in Leaflet 400/1.

2.3 DRAWINGS AND QUALITY CONTROL

2.3.1 All drawings shall state the quality control requirements applicable to the grade of the part or assembly and the reference number of the specification or design designation of the material or process.

2.3.2 Where parts, or designated areas of parts irrespective of their grade, require quality control in addition to visual and dimensional examination (eg, non-destructive tests) the additional inspection requirements shall be detailed or referenced on the drawing.

3 STANDARD PARTS

3.1 The requirement of the appropriate Defence Standards and other Standards listed in Chapters 1 and 2 of Defence Standard 00-00 (Part 3) Section 1 shall be met. Where these requirements standardize a given item, no other items shall be used to perform functions for which this standard item is suitable, unless the use of an alternative item is:

(i) authorised by the Rotorcraft Specification,

(ii) permitted by the requirements of this publication, or

(iii) approved by the appropriate Rotorcraft Project Director - see also the Introduction to DEF STAN 00-00 (Part 3) Section 1.

3.2 When a suitable item is not covered by Defence Standards but is available in one of the following series:

(i) AGS parts,

(ii) SBAC standard parts,

(iii) BSI standard parts,

(iv) AECMA standard parts,

these items are suitable for use.

3.3 All standard items incorporated in the Rotorcraft shall be in accordance with the latest approved issue of the relevant drawings, but the issue number of such drawings shall not appear on the Rotorcraft drawings.

4 MATERIALS AND PROCESSES (See Chapter 407, Para. 6 for materials and processes used for protective treatment)

4.1 For Grade A parts the material and process of manufacture shall conform to an Aerospace Specification.

4.2 For ease of reference the situation on some national and international standards is as follows:

Approved Aerospace Specifications

(i) British Standards: Aerospace Series, including ISO and EN standards published by BSI in the Aerospace Series.

(ii) pr EN Standard specifications issued by AECMA.

(iii) DTD Specifications, including DTD 900 series.

Unapproved

(i) British Standards: General Engineering Series. These will require to be covered by a contractors' specification.

4.3 If no suitable specification is available, the contractor shall prepare and maintain an appropriate material or process specification.

4.4 Grade B parts may be manufactured from materials specified in Para 4.1 above, or from less closely controlled materials (eg, general engineering materials) at the designer's discretion.

5 STRENGTH OF MATERIALS

5.1 See Chapter 401.

6 USE OF WOOD

6.1 No part of a Rotorcraft shall be made of wood, unless authorised either in the Rotorcraft Specification or by the Rotorcraft Project Director in writing after application has been made with evidence that it will be satisfactory in tropical and sea-going conditions. (See also Chapter 407 Para. 15).

7 LOCKING OF THREADED FASTENER

7.1 The standard of locking of a threaded fastener shall be determined by the grade of the application of the fastener within the joint or assembly.

7.2 Threaded fasteners in Grade A applications shall have Standard A locking. Acceptable means of locking to Standard A are given in Leaflet 400/4 Para 2. Additionally, those nut and bolt fasteners in Grade A applications at Paras 2.2.1(ii) and 2.2.1(iii) shall be provided with a secondary means of retention such that, once the fastener is placed in position, the secondary retaining device becomes automatically effective in preventing it from dropping out of position even though the usual retaining device may have been omitted. This secondary device should be automatic in operation and should not depend upon maintenance personnel remembering to carry out a separate action such as the bending of locking tabs or the fitting of locking wire. Secondary means of retention which depend upon friction or springs are usually acceptable.

7.3 Threaded fasteners in Grade B applications shall have Standard A or B locking. Acceptable means of locking to Standard B are given in Leaflet 400/4 Para 3.

7.4 Locking wire shall not be used in pure shear or where movement of the joint could result in rotation of the fastener.

7.5 Centre-popping shall not be used.

7.6 Peening shall not be used to lock:

(i) titanium fasteners or titanium alloy fasteners,

(ii) high tensile strength (over 1100 MPa specification minimum) steel fasteners,

(iii) bolts in joints which have to be dismantled regularly in service,

7.7 Locking adhesives shall not be used in tapped holes and other locations where the mating parts could be damaged or are not readily replaceable.

7.8 In all cases where the method of locking, whether by Standard A or B, does not demand more, the end of the bolt or stud shall protrude beyond the nut by a dimension equal to at least 1.5 thread pitches.

7.9 Where the preferred method of locking conflicts with an operational requirement (eg, where quick release of parts is required), or where value engineering assessment clearly indicates an advantage, other methods may be used provided that it can be shown to the satisfaction of the Rotorcraft Project Director, by tests, that they are satisfactory for the environment they will have to withstand.

7.10 Any damage to protective treatment applied to the structure caused by locking shall be adequately repaired.

8 USE OF LEAD SEALS ON LOCKING WIRE

8.1 The use of lead seals on locking wire is prohibited on Rotorcraft and Rotorcraft equipment, except where permitted by AP 100B-01 Order 6470*. Where equipment is supplied with lead seals they shall be removed, unless they form part of the equipment's warranty. When the lead seals form part of the equipment's warranty the Rotorcraft designer/manufacturer shall inform the Rotorcraft Project Director who will decide whether the lead seals are to be retained, bearing in mind the possible danger to flight safety.

* Order No. 6470 is not available for general circulation, but is concerned with special circumstances relating to Rotorcraft armament systems.

9 USE OF COLD FORGED STEEL BOLTS

9.1 Where cold forged steel bolts are used in Grade A applications and are:

(i) likely to be subjected to any conditions conducive to fatigue failure, or

(ii) used in positions where no relaxation of the initially applied torque is permissible,

the bolts shall be to Specification DTD 5162 or to such other specification which has been prepared by the contractor.

10 USE OF MAGNESIUM ALLOYS (See Leaflet 400/2)

10.1 Magnesium alloy skins or parts made from magnesium alloy sheet shall not be used in military aeroplane structures. Magnesium based alloys in cast or wrought forms may be acceptable subject to appropriate precautions (See Leaflet 400/2).

10.2 Where the Aeroplane Design Authority considers it necessary to use magnesium alloy to meet the draft specification for the aeroplane this shall be made known during the pre-contract negotiations, so that the implications can be assessed. Where magnesium alloy is permitted to be used the aeroplane specification will identify the parts, and conditions for use including protective treatment.

10.3 Magnesium alloy parts shall not be used in places where they are liable to continuous exposure of corrosive fluids, for example:

(i) for enclosed systems where water may collect, such as fuel systems (See Chapter 702).

(ii) for structure in areas where water may collect, such as bilge areas, gaps in control surfaces or folding joints in aeroplanes.

(iii) for toilet, washroom and galley compartments and other underfloor supporting structure.

11 USE OF CADMIUM AND ASBESTOS

11.1 Cadmium and asbestos shall not be included in the composition of materials, products or equipment unless:

(i) it is essential for the satisfactory performance of the materials, products or equipment.

(ii) the use of any alternative material would be equally or more hazardous.

11.2 If the above conditions apply, cadmium and asbestos shall only be used when the Design Authority has obtained prior written agreement for the use of the material from the (Rotorcraft) Project Director.

11.3 Any material, product or equipment which contains asbestos shall be marked in accordance with the symbol and wording of Packaging Form FPKG 770, reproduced in DEF STAN 81-41 Part 6.

12 CONTROLLED TIGHTENING OF BOLTS (See Leaflet 400/3)

12.1 OPERATIONAL REQUIREMENTS

12.1.1 The application of controlled tightening to those bolts in joints which may require dismantling during servicing shall be limited to those joints where the loss of tightness could result in unacceptable degradation of structural integrity, or failure of the part to perform its function.

12.1.2 The bolt elongation technique (See Leaflet 400/3) shall not be used in such applications.

12.1.3 When the torque loading technique is to be used, adequate access shall be available for the use of standard service tools.

12.2 SAFETY REQUIREMENTS (See Leaflet 400/3)

12.2.1 Consideration shall be given to controlling the preload of bolts in the following applications:

(i) heavily loaded tension joints,

(ii) heavily loaded shear joints,

(iii) joints with a group of bolts sharing the load,

(iv) joints subject to thermal strain (see para 12.2.4),

(v) joints where the manner of carrying load in the joint is affected by the preload of a bolt,

(vi) joints subjected to cyclic loading.

12.2.2 The drawing shall show the type of preloading to be used, the necessary data to achieve the correct pretension and the lubricant required.

12.2.3 The lubricant shall be compatible with any seals or non-metallic materials in the vicinity.

12.2.4 Where a joint is subject to sufficient heat to affect the torque values on reassembly of threaded items this shall be identified on the drawings, together with instructions on:

(i) whether items can be re-used if the appropriate run-down maximum and minimum torque values and the final tightening torque value can be achieved when the used items are re-assembled.

(ii) whether the items are to be replaced on re-assembly.

(iii) the extent to which reconditioning, re-coating/re-lubrication or re-plating (where appropriate) is permitted.

13 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE (See Chapter 112)

13.1 Cockpit furnishings, paints, finishings and plastic materials having good repairability and resistance to the nuclear, biological and chemical effects of Defined and Specified Threats shall be used in preference to any alternatives wherever possible.

13.2 Preference should be given to structural materials which are easily repairable.

13.3 The response of structural adhesives to nuclear, biological and chemical effects shall be considered before they are incorporated in the design.

LEAFLET 400/0

GENERAL DETAIL DESIGN

REFERENCE PAGE

<u>Defence Guides</u>

DG-4	Defence Guide to the use of unified screw threads

<u>British Standards</u>

1580	Unified screw threads
A125 to A168	Stiffnuts (UNF threads) for aircraft
SP90	Corrosion-resisting steel split cotter pins for aircraft

SBAC Technical Specifications

120	AGS 2000 series stiffnuts (including clinch nuts) (BA and BSF threads) for aircraft.

<u>Defence Standards</u>

05-34	Marking of Service material
53-1	Use of Unified screw threads
53-9	Bolts, machine screws, and nuts for aerospace vehicles
53-88	Washers, lock, key (tab) and spring tension (crinkle), (inch series) for aerospace applications
53-93	Pins, quick release (ball type)
95-7	Wire for locking purposes and wire locking practice for aerospace use

LEAFLET 400/1

GENERAL DETAIL DESIGN

GRADING OF ROTORCRAFT PARTS AND ASSEMBLIES

1 INTRODUCTION

1.1 This Leaflet explains the objectives of the system of grading of Rotorcraft parts and assemblies, and gives acceptable interpretations of the requirements given in Chapter 400.

2 OBJECTIVES OF THE GRADING SYSTEM

2.1 The main purpose of grading Rotorcraft parts and assemblies is to draw attention to the consequences of their failure and to ensure that appropriate care is taken in the choice of materials, in the manufacturing processes used, and in the inspection applied to them.

2.2 Requirements for design and inspection, particularly those applicable to materials and manufacturing processes, are more stringent for Grade A parts than for Grade B parts. Where a part or assembly is not of Grade A importance, the designer is permitted much wider discretion in the use of materials and in the degree of inspection required provided that other requirements (e.g. fire resistance and maintainability) are met.

2.3 With the exception of Vital Parts, and provided that the grading requirements are met, there is no need for the grade to be identified on the drawing. However, the drawing may be used for this purpose where it is convenient for the Rotorcraft Design Authority to do so.

3 APPLICATION OF THE GRADING REQUIREMENTS

3.1 GENERAL

3.1.1 The designation of a part as Grade A need not necessarily signify that the whole of that part is of special importance. Grade A may be used to indicate that there is some designated area of the part which is of particular importance and in such cases the area must be clearly indicated on the drawing. Grade A may also be used where it is necessary to apply Grade A standards of process control or inspection to parts which would otherwise be Grade B.

3.1.2 Where a number of similar parts is used in one rotorcraft, special attention should be given to the grading of each part individually. If any such part needs to be Grade A then it should be ensured by design that a Grade B part cannot be fitted in its place. Alternatively, at the designer's discretion all such parts may be made Grade A. It follows that the application of each standard part used must be graded and the part selected accordingly. However it is clearly impossible to achieve complete non-interchangeability of fasteners and this is an acknowledged risk.

3.1.3 Consideration must be given to grading a part or assembly Grade A if there is a probability that its failure will result in the malfunction or failure of another Grade A part or assembly. However, if the failure of a part or assembly would only cause minor damage to a Grade A part, or if there was only an acceptably low probability of it causing the malfunction or failure of a Grade A part then the part would, from this aspect, be regarded as Grade B.

3.1.4 In addition to grading for design and inspection purposes, all parts have to be considered for inclusion in the List of Identifiable Parts required by DEF-STAN 05-123. It will usually be found to be convenient to identify and grade parts at the same time. There is no positive requirement that all identifiable parts must be Grade A because the two systems have different objectives. The objectives of the grading system are stated above. 'the objective of the Identifiable Parts List is, as stated in DEF-STAN 05-123, Chapter 244, "to enable such parts to be related back to the batches of materials from which they are made."

3.2 STRUCTURAL COLLAPSE

3.2.1 Where parts of the structure support controls, systems, or items of equipment, the parts should be graded not only as structure but also according to the effect their failure would have on the controls etc.

3.3 LOSS OF CONTROL

3.3.1 Consideration should be given to any changes in aerodynamic characteristics which might be caused by structural deformation, or from disturbed airflow over the Rotorcraft after a failure or partial failure has occurred. The possibility of control reversal, flutter, or buffet due to any decrease in stiffness after the failure of a part, is of particular importance.

3.4 FAILURE OF MOTIVE POWER

3.4.1 Any component whose malfunction or failure could result in a significant power loss should be Grade A.

3.5 SYSTEMS AND EQUIPMENT

3.5.1 For the purposes of the requirements the following systems are regarded as essential:

(i) Those on which the safety of the rotorcraft and aircrew depends.

(ii) Those required for the emergency escape of the occupants.

(iii) Those required for the performance of operational missions.

3.5.2 The extent to which the grading system is applied to equipment in one of these systems should be determined by discussion between the Rotorcraft designer and the equipment designer. All mountings and attachments both on the airframe and on the equipment will need to be considered.

3.5.3 When designing multiple systems, the determination of the probability of failure should be based on the system as a whole. Each unit system may be Grade B provided that, after failure of one unit system, the remaining system(s) can function acceptably until the failure is rectified.

3.6 INCAPACITATING INJURY TO ANY OCCUPANT

3.6.1 In normal flight this should be interpreted to mean an injury which would significantly impair the efficiency of any occupant and hence:

(i) hazard safe flight,

(ii) cause a mission to be aborted,

(iii) prevent rapid evacuation in the event of a subsequent crash-landing or ditching.

3.6.2 In a crash-landing or ditching, the degree of injury will be greater and all local structure and equipment should be designed to minimise injuries which would significantly impair the chances of survival.

3.7 UNACCEPTABLE UNSERVICEABILITY OR MAINTAINABILITY

3.7.1 In deciding the grade of a part or assembly the designer should consider the ease with which the part could be replaced if it failed in service.

3.8 'VITAL' GRADE A PARTS

3.8.1 Parts of Rotorcraft, especially in the rotor and transmission systems, which are impractical to duplicate and are subject to fatigue loading, can be affected considerably by small defects and deviations from the drawing and therefore need extra special care and inspection during manufacture and storage. Unless high safety factors show that failure will be extremely remote, these parts should be treated as VITAL.

3.8.2 Procedures for the Quality Control of VITAL parts shall ensure that sufficient information is included on the drawings and that all manufacturing and inspection sequences are sufficiently controlled to maintain quality.

3.8.3 The VITAL procedures may also be applied to other Grade A parts.

LEAFLET 400/2

GENERAL DETAIL DESIGN

PRECAUTIONS TO BE TAKEN IN THE USE OF MAGNESIUM ALLOY PARTS

1 INTRODUCTION

1.1 This leaflet defines the precautions to be taken during design, manufacture and installation of magnesium alloy parts to prevent corrosion during service and storage. Para 4 exemplifies types of usage to be avoided since these are unlikely to be accepted under the provisions of Chapter 400.

1.2 The requirements for the protection of magnesium alloys are given in Chapter 409.

2 MANUFACTURING

2.1 It is strongly recommended that there should be close and early consultation between the Design Authority and the material supplier with regard to surface treatment and application of protective treatments in accordance with Chapter 409.

3 INSTALLATION PRECAUTIONS

3.1 Magnesium alloys are especially susceptible to galvanic corrosion (Leaflet 409/3). Direct contacts between magnesium alloy and carbon fibre reinforced composites are not permitted. Where magnesium components are in direct contact with other metals it is particularly important that the precautions against galvanic attack given in Chapter 409 are rigorously applied.

3.2 The design and assembly of the parts should be such that water collection in pockets is prevented (See Chapter 409).

3.3. Parts should be readily accessible for visual inspection, for approved non-destructive tests, and for treatment or replacement in accordance with the requirements of Chapter 409.

4 LIMITATIONS TO USE OF MAGNESIUM ALLOY

4.1 The use of magnesium alloy parts should be avoided in places where they are liable to intermittent contamination by corrosive fluids, for example:

(i) for wheel wells, and for other parts which may be subject to the wash emanating from the main or nose wheels during take-off and landing on runways affected by snow, slush or salt laden water,

(ii) for parts of ship-borne Rotorcraft exposed to salt spray and other corrosive agents, such as funnel gases,

(iii) for parts of Rotorcraft exposed to salt, fog and spray through operating in a marine environment.

4.2 Magnesium alloy should not be used for parts whose protective coatings may be penetrated to expose the bare metal during normal operation, ground handling or servicing, for example:

(i) parts subject to wear during normal operation, eg sliding parts in control systems,

(ii) parts regularly subject to abrasion eg the effects of heavy rain, hail or sand,

(iii) parts which are liable to be damaged during servicing eg access panels, and

(iv) parts which may be damaged by personnel or ground equipment during servicing, eg walkways, fitting points for ladders.

4.3 However, when magnesium alloy parts are to be used special care and attention should be given to the selection of the alloy, protective scheme and inspectability.

LEAFLET 400/3

GENERAL DETAIL DESIGN

CONTROLLED TIGHTENING OF BOLTS

1 INTRODUCTION

1.1 In many applications, tightening of bolts can be sufficiently accurately determined by the judgement of a skilled operator. However, in other applications, controlled tightening is required to improve the effectiveness of the joint, and to ensure bearings rotate on a lubricated face not on the bolt which would promote rapid wear.

1.2 This leaflet discusses the various applications and means of achieving tightening of parallel shanked bolts.

1.3 When it is proposed to apply controlled tightening of bolts in joints which may require dismantling during maintenance, the proposal should be discussed with the Rotorcraft Project Director so that details on standard service tools and trade skills can be made available and so that the necessary servicing arrangements can be made.

2 APPLICATIONS OF CONTROLLED TIGHTENING

2.1 BOLTS IN TENSION

2.1.1 The fatigue endurance of a bolt under cyclic axial load may be considerably increased by controlled tightening (see Ref 1). Without it, an applied load is transmitted directly to the bolt, whereas with it, some of the applied load is absorbed by a relaxation of the compressive forces in the fitting. Although the mean stress is slightly increased, the alternating stress is reduced.

2.2 BOLTS IN SHEAR

2.2.1 An effective means of increasing the fatigue endurance of a shear joint is to clamp the parts tightly together so that the alternating load is transferred by friction at the mating surfaces. For maximum effectiveness, the pre-load needs to be sufficient to prevent slipping so that fretting is reduced and no alternating load is applied to the bolt or to the material round the hole.

2.3 BOLT GROUPS

2.3.1 Where a joint comprises a number of bolts, controlled tightening ensures that all the bolts in the group are sufficiently and uniformly tight thus assisting in a favourable load distribution in the structure.

2.4 JOINTS IN SENSITIVE MATERIALS

2.4.1 The allowable pre-load in bolts in sensitive materials (e.g. transparent panels) is generally much less than that which would normally be applied. It is therefore important to limit it accurately in these circumstances.

2.5 JOINTS SUBJECT TO THERMAL STRAIN

2.5.1 When an assembly, in which the bolt material differs from the joint material, is subjected to a range of temperatures, it is possible for either the bolt or the joint to be overstressed, or for the bolt to lose all its pre-load. Controlled tightening of the bolts is almost certainly necessary to avoid these problems.

2.5.2 On joints operated at elevated temperatures, where plating, e.g. silver, or coating of threaded fasteners is used to prevent seizure of the joints, re-assembly of the fasteners may give incorrect run-down torque values which may lead to inadequate or excessive final torques at the joint. Minimum and maximum run-down torque values should therefore be identified on the drawing.

2.6 PRODUCTION AID

2.6.1 During the production of a rotorcraft there may be some advantage in controlled tightening all bolts as a means of protecting the structure from slack or over-tightened bolts.

3 METHOD OF ACHIEVING A GIVEN PRE-LOAD (see Refs 2 and 6)

3.1 BOLT ELONGATION

3.1.1 Probably the most accurate assessment of pre-load is obtained by measuring the bolt elongation during tightening. The application of this method is limited because of the difficulty of measuring the bolt elongation in many cases and the time taken to determine the load in each bolt.

3.2 PRE-LOAD INDICATING WASHERS

3.2.1 PLI washers are useful where a high pre-load (e.g. in excess of 80% and the 0.1% proof stress) is required as they have an accuracy of ±15%.

3.2.2 PLI washers are also useful where access for a torque spanner is not practicable. However, PLI washers have a number of disadvantages as follows:

(i) It is not possible to check for correct assembly after the nut is tightened.

(ii) The assembly indicates that the minimum torque has been applied but does not prevent a maximum torque being exceeded.

(iii) Ordinary washers could be substituted inadvertently on re-assembly after any servicing of the joints.

3.3 TORQUE LOADING

3.3.1 Although this is probably the most popular method of determining the pre-load in a bolt, its accuracy may be as poor as ±30%, even though the torque applied may be accurate to within ±1%. Consequently, this method is not recommended where stresses greater than 80% of the 0.1% proof stress are required.

3.3.2 The threads of all torque-loaded bolts must be lubricated to ensure the reliability of the relationship between applied torque and pre-load. The lubricant specified which may be wet or dry must be compatible with any sealing or non-metallic materials in the immediate vicinity.

3.3.3 Theoretical estimates and empirical formulae (Refs 4 and 7) have been derived for the relationship between the applied torque and the pre-load but neither approach gives reliable data for practical use, and it is recommended that the relationship be established by test.

3.3.4 The torque load is usually applied to the nut by means of a torque spanner. Providing that the nut is co-axial with the drive of the torque spanner, the torque applied is accurate to within ±1%. However, large errors (see Ref 5 for a full analysis) can arise when extension adaptors are used to reach into awkward places. It is essential in this case to calibrate the torque spanner with the appropriate adaptor in place and with the force applied to the spanner in the same direction as on the rotorcraft.

3.3.5 On large diameter threads with high pre-load it is necessary to apply pure torque to achieve a consistent pre-load v torque relationship. A high degree of thread cleanliness is also required.

3.4 TURN OF NUT

3.4.1 This method calls for the nut to be just tightened by hand and then turned a further specified amount. The major disadvantage of this method is the difficulty of determining when the nut is just tight, especially if there is any resilience in the joint. The method is not often used and no experimental results are available.

4 LOSS OF PRE-LOAD

4.1 With pre-loaded bolts, the tension reduces significantly with time as the nut and bolt become embedded in the abutments, and because of other factors such as creep and the continued extrusion of sealing and jointing compounds. Thus any joint which relies on controlled tightening to achieve a safe life should either be periodically tightened, or should be initially pre-loaded sufficiently to ensure that the tension does not fall below a safe value.

REFERENCES

No	Author(s)	Title, etc
1	Walker P	Fatigue of a Nut and a Bolt RAE Structures Report 238 June 1958
2	Fisher W A P Cross R H Norris G M	Pre-tensioning as a Means of Preventing Fatigue in Bolts RAE Structures Report 84 July 1950

3	Stang A H	The Tensile Forces in Tightened Bolts Product Engineering - February 1951
4	Johnson K L	Tightening Torque Versus Bolt Tension Relationships ASME Paper 69-DE-48 May 1969
5	Sharman J M	The Principle and Practice of Torque Loading The Production Engineer February 1961
6	Riches D M	Report on Vibration Testing of Threaded Fasteners BAC Ltd GW Division Report L74/20/RES/FAS/MS268
7	ESDU	Engineering Sciences Data Item 72022 'Tension in steel bolts resulting from tightening torque (tentative)'

LEAFLET 400/4

GENERAL DETAIL DESIGN

LOCKING OF THREADED FASTENERS

1 INTRODUCTION

1.1 This leaflet gives some background information and acceptable means of complying with the requirements of Chapter 400, para 7.

1.2 A threaded fastener in the context of these requirements includes nuts, bolts, screws, studs, turnbuckles, thread inserts and threaded pipe couplings.

2 STANDARD A LOCKING

2.1 Standard A locking (often called positive locking) is any method of locking which has been shown by service experience or by representative tests to hold the essential components of the fastener in that relationship to each other, and to the local structure, which is essential to the proper functioning of the fastener, according to its purpose, throughout the life of the joint; and to do this in all combinations of environmental conditions, loading actions, and modes of vibration, likely to be encountered in service.

2.2 The following are acceptable methods of locking nuts to Standard A:

2.2.1 Preferred Methods

(i) Standard approved split cotter pins, or clips, or locking pins, in castellated or slotted nuts.

(ii) Locking wire in accordance with Def Stan 95-7 through the nut corner or bolt head provided that the nut and bolt are locked together or each is locked separately. As locking wire is comparatively weak in shear it must not be used with castellated or slotted nuts, or where the fastener is subject to rotation from any cause including flexure of the structure.

(iii) Approved self-locking nuts (stiff-nuts) with the appropriate grade of approved locking adhesive.

(iv) Approved self-locking nuts (stiff nuts) or approved screwlocking thread-inserts without locking adhesive, on bolts in Grade A joints provided that there are a sufficient number of bolts in the joint to satisfy the damage-tolerance requirements, (see Leaflet 400/1 para 3.1.4) and provided that any loose or detached part could cause no damage.

2.2.2 Other Method(s)

Peening (but not centre-punching or centre-popping) when there is no planned maintenance which would necessitate the parts secured having to be removed or dismantled. Adequate precautions are to be taken, particularly with regard to dollying and the control of impact, to ensure that there is no damage to the structure. A recommended method of peening is shown in Fig. 1. The bolt projection is specified in Chapter 400 para 7.8 and the peening should be carried down to the nut to prevent it slackening. Peening is only to be used in circumstances when preferred methods cannot be used. The agreement of the Rotorcraft Project Director shall be obtained.

NOTES:

1 Adequate head support is essential particularly where harder modem materials are involved, however, material must be sufficiently ductile to obtain deformation without cracking.

2 Special tooling or jigs should be provided as appropriate in order to achieve consistency.

3 The applicable torque loading should be applied before and after peening. If any rotation occurs the process must be repeated.

4 Peening is not to be used on set bolts.

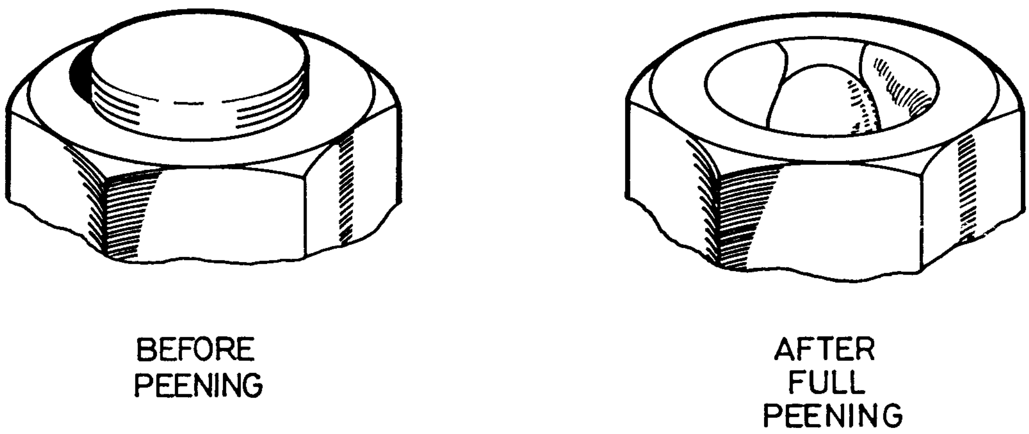

FIG.1 RECOMMENDED METHOD OF PEENING

2.3 The following are acceptable methods of locking threaded fasteners, other than by nuts, to Standard A:

(i) Screws and bolts may be locked directly or indirectly via the structure, to members containing the tapped holes with which they engage, using the appropriate methods described in para 2.2.

(ii) Studs may be locked by any of the methods described in para 2.2 which are appropriate provided that either they:

(a) are locked independently of the nut,

or (b) have each stud locked to its nuts by a split cotter pin and the nut further locked to the component; in the case of a number of studs the nuts may be locked together in groups,

or (c) have a lock nut or jam nut and a structural nut on each stud with the nuts wire locked to each other and to the component or wire locked together in groups.

3 STANDARD B LOCKING

3.1 The following methods are acceptable for locking threaded fasteners to Standard B:

(i) Approved self-locking nuts (stiff-nuts).

(ii) Double nuts, locked together.

(iii) Single or double turn coil spring washers.

(iv) Tab washers provided the bolt is also locked.

(v) A single nut locked by an approved locking adhesive.

(vi) Approved screw-locking thread-inserts.

4 THREAD INSERTS

4.1 Thread inserts should not be used in basic design if an acceptable alternative is available. The use of thread inserts in service will be governed by repair schemes in the maintenance manual. Screw-locking thread-inserts should not be used if they could cause critical fatigue loading or galvanic corrosion.

5 SWARF

5.1 Consideration should be given in all joints to the possible hazards which may arise from swarf. In joints where stiff-nuts are permitted, those which have non-metallic friction elements should be used where swarf may cause a hazard.

6 ENVIRONMENTAL EFFECTS

6.1 Care should be taken to select the correct materials where environmental conditions may affect the efficiency of locking.

7 PIPE COUPLINGS

7.1 In order to comply with the requirements, Grade A pipe couplings will normally have Standard A locking. Other methods may be used provided that it can be shown by tests that the design of the coupling is satisfactory for the environment it will have to withstand. Wire locking, should be avoided if a suitable proven alternative is available.

7.2 The need for Standard A locking for pipe couplings in airborne ground equipment having a Grade A application should also be considered.

8 FURTHER INFORMATION

8.1 Further information and guidance on various methods of locking threaded fasteners are given in DEF STAN 53-32.

8.2 DEF STAN 53-90 lists currently approved self-locking nuts (stiff-nuts).

CHAPTER 401

DESIGN DATA FOR METALLIC MATERIALS

1. Chapter 200 states that ‘B’ allowable values shall be derived for all Grade A structural details. To calculate these values it is necessary to refer to ‘B’ allowable data for structural materials. This chapter provides information on acceptable data for metallic materials.

2. Acceptable properties for use in the design of metallic details are as follows :

(i) As stated in the Metallic Materials Data Handbook (DEF STAN 00-932) obtainable from the ESDU International plc 27 Corsham Street, London N1 6UA.

or

(ii) As obtained or derived from test data when not included in DEF STAN 00-932.

or

(iii) As obtained by using approved (see chapter 400) processes and control specifications to maintain a mechanical property which is more advantageous than that included in DEF STAN 00-932.

3. Data obtained as in para 2(ii) and 2(iii) above shall be derived by the same methods as are used to establish the data in DEF STAN 00-932 and specified therein.

4. Adequate allowance shall be made, where applicable, for the effects of all manufacturing processes and environmental conditions on material strength and for all tolerances, ranges, variations, or limitations stated or referenced in DEF STAN 00-932.

5. Data in DEF STAN 00-932 are given for materials to the specification issue or date given. It is the Chief Designer’s responsibility to ensure that the data used are relevant to the specification issue or date defined on or through the drawing.

6. When using data given in DEF STAN 00-932 under para 2(i) above the following requirements apply:

6.1 For the design and airworthiness acceptance of all parts the data given on white paper shall be used. Data on grey paper shall not be used.

6.2 Where no ‘B’ values are available and only ‘S’ values are given on white paper then ‘S’ values are acceptable.

6.3 Data given on grey paper may be used for preliminary project work only.

7. The use of ‘B’ values shall be associated with a procedure to monitor the appropriate specification mechanical properties to ensure and demonstrate that the values are being maintained.

Note: The definitions of A, B and S values are to be found in DEF STAN 00-932.

CHAPTER 402

PROCESSES AND WORKING OF MATERIALS

1 JOINTING PROCESSES

1.1 This chapter covers jointing processes in general but excludes any application using fasteners.

1.2 In assessing the suitability of a method for making consistently sound and durable joints, account shall be taken of the following points:

(i) the material to be joined and the position and ease of accessibility of the joint,

(ii) the structural importance of the joint,

(iii) the extent to which the uniformity and quality of the joint can be guaranteed by control of the method of fabrication,

(iv) the extent to which the soundness of a joint after fabrication may be judged by inspection methods,

(v) the accuracy with which the static, impact and, where necessary, fatigue properties of the joint may be predicted from evidence available on similar joints,

(vi) the possibility of reduction in strength of the joint with time due to corrosion, deterioration of the jointing medium or similar causes,

(vii) the need to maintain RF screening properties and prevent electrical discontinuities across the joints.

1.3 The strengths used for design purposes shall be based upon the results of tests on representative specimens. Where adequate and relevant strength data are available from past experience, these may be used.

1.4 All joints made by a metallic jointing process shall be graded in the following manner:

(i) Grade A: A joint the failure or leakage of which would result in any of those consequences of failure which define a Grade A part. All such joints are to be subject to basic inspection techniques, augmented with ultrasonic, radiographic and dye penetrant or magnetic examination as appropriate. Drawings for Grade A parts shall define the additional inspection required.

(ii) Grade B: All other joints. These to be subject to basic inspection techniques and such other tests as may be required.

1.5 Leaflet 402/1 gives advice and recommendations on the design, inspection and strength of joints made by fusion and friction welding, and by diffusion bonding.

1.6 Leaflet 402/2 gives advice on processes and control of adhesive bonding of structural joints. Recommended design practice for adhesive bonding of structural joints is given in Leaflet 402/3.

1.7 Leaflet 402/6 gives advice and recommendations on the use of brazing and soldering.

2 STRENGTH AFTER PROCESSING

2.1 DETAIL DRAWINGS

2.1.1 Where the process adopted for manufacture may affect the strength of a part, sufficient information shall be given on the detail drawings or in documents related to the drawing to ensure that the part is fabricated by the process or sequence of processes necessary to provide the required strength.

2.1.2 Where minimum strength properties for welded joints are required, these shall be quoted in the drawings.

3 DE-MAGNETISING

3.1 See Chapter 713 Para 8.1 and Leaflet 713/3.

4 FLAW DETECTION

4.1 The need for a flaw detection test on each part should be considered and the drawings endorsed accordingly. The technique to be employed in conducting such tests shall be agreed, where necessary, between the designer and the manufacturer.

5 FORGINGS

5.1 On the drawings of Grade A parts, the direction of grain required shall be indicated clearly in a manner which will ensure that it is brought to the notice of the person responsible for deciding the forging technique to be adopted.

5.2 At least one specimen from each source of supply of every Grade A forging shall be strength tested in the critical design case. The load demonstrated must be in accordance with Chapter 200, Table 1. A note to this effect shall be included in the drawings of the forging. This test may, however, be waived either:

(i) when the forgings show similarity in design to forgings already approved by test from the same source of supply. (It is important that the initial billet size and ultimate ruling sections are similar to those previously tested and that the strength of the forging can be estimated reliably from existing test results), or

(ii) when the calculated strength of the forging is not less than 1.5 times the fully factored load in the critical design case.

In order that critical highly stressed unmachined areas shall be given the extra degree of inspection which they need to guarantee the integrity of the forging, the drawing shall be marked, normally by cross-hatching, to indicate any areas, not to be machined, which will be critically highly stressed and shall describe the finish to be given to such areas prior to inspection by the forger. Unless otherwise arranged between the forger and the rotorcraft designer, inspection shall be carried out after solution treatment or full heat treatment and subsequent chromic acid anodising.

Note: In no instance is it visualised that a forging should be assessed as critically highly stressed all over, and it does not follow that all forgings will contain surface areas meriting this special treatment.

5.3 Each Grade A steel forging shall incorporate, whenever practicable, one or more projections which, after heat treatment of the forging, can be used as test piece(s) to detect whether the forging has been overheated. The location(s) and dimensions of these test pieces shall be decided in consultations with the supplier of the forging.

6 ADHESIVE BONDING

6.1 Information on approved procedures for adherend surface preparation, application and bonding of adhesives, process control and production facilities is contained in Leaflet 402/2.

6.2 Selection of an adhesive and surface preparation of components shall take account of all relevant factors including the effects of natural environments, in particular warm wet conditions, and aerospace fluids so as to maintain an adequate joint strength of the component throughout its specified life.

6.3 Selection of adhesive and material to be bonded shall take account of the effect of pretreatment and bonding temperature cycle on the strength, fatigue and corrosion resistance of the structural material used.

6.4 All designs shall take account of the complex stress distribution occurring within a loaded adhesively bonded joint and that the strength of a joint cannot generally be computed directly from the strengths obtained from standard single lap shear specimens.

6.5 Structural tests shall be undertaken for all Grade A components unless it can be demonstrated by reference to existing test data for some sufficiently similar approved and satisfactory structural application that the proposed new application will be satisfactory.

6.6 Attention shall be given to the use of corrosion protection coatings on metal honeycomb core (See Chapter 407).

7 SEALANTS AND SEALING (See also Leaflet 402/7)

7.1 The type of sealant shall be selected with due regard to the static and dynamic requirements and environment of the sealed joint. In addition the possible need for subsequent inspection involving both sealant removal and re-sealing shall be considered. Environmental factors include, but are not limited to, temperature range, the presence of liquids and gases, whether by design or by contamination from adjacent systems and other materials with which the sealant may come into contact. it should be noted that some sealants soften in the presence of liquids and/or their vapours and it is important that due allowance is made for this in the design especially where the sealant plays an integral part in the behaviour of the joints in fatigue. Overcoating with a suitable barrier coat/sealant may lessen this effect.

7.2 In designs where a preformed gasket is not appropriate it may be possible to utilise a low adhesion sealant to form an in place gasket.

7.3 All designs shall be such as to take into account the possible requirements for interfayed joints, i.e. those where further operations may require the use of fasteners through the sealed joint either within the work life of the sealant or after it has cured. Where the fatigue behaviour of a joint is critical, the joint shall be fully assembled and fastened within the work life of the sealant being used.

7.4 The design shall ensure that the appropriate pretreatment including cleaning and degreasing and, where necessary, the use of primers, is carried out on the surface which will form the sealed joints.

7.5 Where leak resistance is a prime requirement a sufficient number of fasteners to ensure adequate sealing, over and above that required to provide the necessary strength shall be used.

7.6 In many cases the sealant acts not only as a means of containment of a fluid but also prevents moisture ingress into the joint thus resisting corrosion and in some instances is used only to inhibit corrosion. In any case the requirements for corrosion prevention and control described in Chapter 407 shall prevail.

7.7 Due consideration shall be given to the position of concentrated load attachments and their resulting local strain. As far as possible these shall be designed to be outside the boundaries of the tank or other vessel. Where this is not possible the design shall make provision to enable access, in service, to permit inspection of the joint and rectification if necessary.

LEAFLET 402/0

PROCESSES AND WORKING OF MATERIALS

REFERENCE PAGE

Note: See also list of references in Leaflets 402/1, 2, 4

MOD Specifications

DTD 775	Adhesive suitable for joining metals.
DTD 861	Adhesive for metal (low pressure type).
DTD 5577	Heat stable structural adhesives.

Defence Standards

03-2	Cleaning and preparation of metal surfaces.
05-21	Quality control system requirements for industry.
03-24 .. .. .. (or prEN 2101)	Chromic acid anodising of aluminium and aluminium alloys

British Standards

BS 5350	Methods of test for adhesives
Part A	Adherend Preparation
Part B	Adhesive Properties
Part C	Joints
Part D	Environmental tests
Part E	Sampling and analytical test data

Contractors Reports

AL/Mat 3358	Study of the fatigue characteristics of adhesive bonded joints, Parts 1 and 2. British Aircraft Corporation (CA) Ltd Contract KS43a/334/CB43(a)2 1974.
AL/Mat 3300	Factors affecting the strength of adhesive bonded joints in aluminium alloys. British Aircraft Corporation (CA) Ltd Contract K43A/183/43A2 1974.
SIRA Report	Design data sheets for structural adhesive bonding. SIRA Report 1975 Contract K/A83A/691.
HSA-MSM-R-GEN-0345 ..	The environmental durability of adhesive bonds made between CFRP composite and Titanium. Final report 1798. British Aerospace Aircraft Group, Manchester. Contract K/LR32B/2250.

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AL/Mat3853	Durability of adhesive bonded aluminium alloy joints. British Aerospace, Aircraft Group Contract A/93b/1033 1983
AL/Mat 3917A	Factors affecting the durability of aluminium alloy adhesive joints. British Aerospace, Military Aircraft Division Contract A/94b/1883 1988
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RAE Technical Memoranda

Mat 175	The effects of outdoor exposure on stressed and unstressed adhesive bonded metal to metal joints. Trial 2, Part 2. Four year summary. 1973.
Mat 184	The effects of twelve months' outdoor exposure in a hot wet environment of Defence Standard 03-2/1 acid etched aluminium alloy double overlap joints bonded with an epoxy-polyamide adhesive and protected with polysulphide sealant. 1974.

RAE Technical Memoranda (contd)

Mat 277	.. .. ..	The fatigue and tensile properties of Redux joints between aluminium alloy DTD 646 sheets at temperatures -60°C room temperature and +70°C.
Mat 348	.. .. ..	A comparison of phosphoric acid anodise bonding pretreatment for aluminium with chromic-sulphuric acid pickle and chromic acid anodise, 1980.
Mat 349	.. .. ..	Evaluation of the wedge cleavage test for assessment of durability of adhesive bonded joints, 1980.
Mat/Str 1038	.. .. ..	An evaluation of the Boeing wedge test for the assessment of CFRP surface pretreatment.

RAE Technical Reports

70081	.. .. ..	Effect of outdoor exposure on stressed and unstressed adhesive bonded metal to metal joints. Trial 1. Part 1. Two year summary. 1970.
72100	.. .. ..	Effect of outdoor exposure on stressed and unstressed adhesive bonded metal to metal joints. Trial 2. Two year summary. 1972.
73016	.. .. ..	Effect of outdoor exposure on stressed and unstressed adhesive bonded metal to metal joints. Trial 1. Part 2. Four year summary. 1973.
82029	.. .. ..	The effect of extended exposure to hot-humid conditions and subsequent drying on adhesive bonded CFRP-CFRP joints, 1982
82102	.. .. ..	A comparison of the strengths of metal-metal and metal-CRFP adhesive bonded joints at various test temperatures, 1982.
85093	.. .. ..	Effect of surface pretreatment and alloy type on the durability of adhesive bonded titanium alloy joints, 1985.
89001	.. .. ..	Effect of extended hot-humid exposure on adhesive bonded CFRP joints. Part 1: 175°C cured matrix.

RAE Translations

886	.. .. ..	Contraction and internal stresses in bonded joints. March 1960.
1999	.. .. ..	Environmental effects on the elastic-plastic properties of adhesives in bonded metal joints, 1979.
2038	.. .. ..	The diffusion of water vapour in humid air into the adhesive layer of bonded metal joints, 1980.

American Relevant Documents

ASTM-D-1780-72	..	..	Recommended practice for conducting creep tests of metal-to-metal adhesives (RT-ambient).
ASTM-D-1781-76	..	..	Climbing drum peel test for adhesives.
ASTM-D-1876-72	..	..	Peel resistance of adhesives (T-peel test).
ASTM-D-2919-84	..	..	Determining durability of adhesive joints stressed in shear by tension loading.
ASTM-D-3163-73	..	..	Recommended practice for determining the strength of adhesives in shear by tension loading (rigid plastics).
ASTM-D-3165-73	..	..	Strength properties of adhesives in shear by tension loading of laminated assemblies.
ASTM-D-3166-73	..	..	Fatigue properties of adhesives in shear by tension loading (metal/metal).
ASTM-D-3167-76	..	..	Floating roller peel resistance of adhesives.
ASTM-D-3433-75	..	..	Fracture strength of cleavage of adhesives in bonded joints (G_1).
ASTM-D-3528-76	..	..	Strength properties of double lap shear adhesive joints by tension loading.
ASTM-D-3762-79	..	..	Adhesive bonded surface durability of aluminium (wedge test).
ASTM-D-3983-81	..	..	Recommended practice for measuring strength and shear modulus of non-rigid adhesives by the thick adherend tensile lap specimen.
ASTM-D-1002-72	..	..	Strength properties of adhesives in shear by tension loading (metal to metal). (Lap shear).
ASTM-D-1151-84	..	..	Effect of moisture and temperature on adhesive bonds.
ASTM-D-1183-70	..	..	Tentative - Resistance of adhesives to cyclic ageing conditions.
ASTM-D-1828-70	..	..	Tentative - Atmospheric exposure of adhesive bonded joints and structures.
ASTM-D-2093-84	..	..	Preparation of plastics prior to adhesive bonding.
ASTM-D-2295-72	..	..	Tentative - Strength properties of adhesives in shear by tension loading at elevated temperatures (metal to metal).
ASTM-D-2651-79	..	..	Preparation of metal surfaces for adhesive bonding.
ASTM-E229-70	..	..	Shear strength and shear modulus of structural adhesives.
MM-A-132	..	..	Adhesives, heat resistant, airframe structural, metal to metal.
Mil-A-9067C		..	Adhesive bonding, process and inspection requirements for.
Mil-A-25463A	..	..	Adhesive, film form, metallic structural sandwich construction.
Mil-HDBK-691B	..	..	Adhesive bonding.
Mil-A-83377A	..	..	Adhesive bonding (structural) for aerospace systems, requirements for.
Mil-HDBK-725	..	..	Adhesives. A guide to their properties and uses as described by Federal and Military Specifications.

American Relevant Documents (contd)

McDonnell-Douglas	Primary Adhesively Bonded Structure Technology (PABST), Progress Reports and Technical Bulletins.
Federal Aviation Agency ..	Quality Control Digest No 5, Bonding Inspection.
SAE ARP 1524 (1978) ..	Surface preparation and priming of aluminium alloy parts for high durability adhesive bonding (Phosphoric acid anodising).

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EN 2243	Test methods for adhesives
Part 01	Single lap shear test.
Part 02	Peel test, metal-to-metal
Part 03	Peel test, metal-to-honeycomb core.
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LEAFLET 402/1

PROCESSES AND WORKING OF MATERIALS

FUSION WELDING, FRICTION WELDING AND DIFFUSION BONDING

1 INTRODUCTION

1.1 This leaflet gives data and recommendations for the design of structures made by fusion and friction welding and diffusion bonding.

1.2 Data and recommendations for brazing and soldering are given in Leaflet 402/6.

1.3 If a significant departure from the recommendations of this Leaflet is proposed for a Grade A structure, the proposal should be discussed with the Rotorcraft Project Director.

2 GENERAL NOTES ON THE DESIGN OF WELDED JOINTS

2.1 In order to take full advantage of the strength of the parent metal, it is often advantageous to increase the cross sectional areas of the parts near the joints. When it is practical the joint should be designed to be in the position of least stress and by careful selection of parent metal, welding process, filler metal and sometimes heat treatment, it may be possible to attain parent metal static properties in the weld.

2.2 As a weld normally acts as a stress raiser, both as a result of the change in section that usually occurs at the weld and the change in metallurgical structure, this can reduce markedly the local fatigue strength. Hence, where fatigue strength is important, particular attention should be paid to weld geometry, and quality of weld and underbead. For optimum fatigue performance it may be desirable, where practicable, to machine both top and bottom surfaces of the weld, and preferably both surfaces should be subject to a surface working process, (eg shot peening). If the weld is to be machined after welding, allowance should be made in the initial profile of the basic material for material subsequently machined away.

2.3 Where a joint is made by a single weld from one side, every attempt should be made to ensure full penetration. Special attention should be paid to inspection of the weld root. Where full penetration of the weld is not feasible, the joint should comprise two welds, one from each side, to avoid failure from local bending on either side. Significant tensile stresses in the root of a partial penetration weld should be avoided in view of the severe stress raiser thus formed.

2.4 The design of welded joints should take into consideration proper access for the specified welding process.

2.5 Welding methods which involve the use of a flux should be avoided where possible, particularly where it is not possible to remove the flux completely after welding, (eg welded end fitting which closes the end completely).

2.6 The welded assembly should be adequately protected from corrosion. For this reason, lap joints which are not welded around all edges should be avoided, or special attention paid to their method of sealing.

2.7 The design and construction of components should permit all welds to be examined in accordance with Para 3 as applicable after final welding, heat treatment, and during service. Components which are of such complex configuration that these requirements cannot be met should be examined by NDT at suitable stages of manufacture. Alternative methods such as the use of test pieces fully representative of the component and which can be subject to metallurgical examination may be used.

2.8 Where practicable, the parts to be joined should be of the same thickness and they should be maintained in good alignment.

2.9 It is possible to reheat-treat heat-treatable alloys after welding to obtain maximum strength (eg harden and temper steels or solution-treat and age aluminium alloys), but heat treatments involving quenching may lead to severe distortion in complicated assemblies. It is, therefore, recommended that such reheat treatments should be restricted to simple shapes and where maximum strength is really necessary, or where the parts can be adequately jigged.

2.10 Welded assemblies made from materials with a tensile strength greater than approximately 750 MPa should be subjected to a post-weld stress relieving treatment. Stress relieving of lower strength material can also be advantageous to prevent cold cracking, to optimise heat affected zone properties and where maximum dimensional stability is required. It should be noted that some general engineering structural steels suffer temper embrittlement, and even cracking, if stress relieved.

2.11 The local application of heat should only be permitted for the purpose of preheating or stress relief, and then only under controlled conditions.

2.12 Welding of high strength carbon and low alloy steels with a carbon equivalent greater than 0.26% should not be undertaken without consideration of pre and post-welding procedures that may be necessary to reduce the susceptibility to cracking. The carbon equivalent (CE) should be calculated using the formula:

$$CE = \%C + \%Mn/6 + (\%Cr + \%Mo + \%V)/5 + (\%Ni + \%Cu)/15$$

This is a simplified version of the Dearden and O'Neill formula.

2.13 Different welding processes give widely varying rates of heat input (See Table 1). High rates of heating can give less distortion. High rates of heating can also result in narrow heat-affected zones so that a smaller volume of a component made from age or work hardened material is softened. Nevertheless local high intensity heating can give high hardness in fusion and heat-affected zones (eg, in titanium alloys and in carbon and low alloy steels), resulting from rapid cooling produced by the large volume of adjacent unheated material.

2.14 The materials in the BS Aerospace and DTD series which are normally accepted as weldable are listed in Table 2, together with the recommended filler rods. Generally welds are more prone to cracking, when high strength filler rods are used and where operational stress in the weld is low it may be possible to use a lower strength filler rod with advantage.

2.15 The ductile-brittle transition temperature of steels is often increased in the heat affected zone and may increase the possibility of fracture in the operating temperature range.

3 INSPECTION OF WELDED JOINTS

3.1 GRADING OF WELDS

3.1.1 For quality control purposes all welded joints must be graded A or B as given in Chapter 400 Para 2 in relation to the grading, of parts.

3.1.2 Further advice on the definitions of Grade A and Grade B is given in Leaflet 400/1.

3.1.3 Grade A welds must be subjected to the basic inspection techniques described in Para 3.2 augmented by one or more of the following (See Para 3.3):

(i) Radiographic Examination.

(ii) Penetrant Inspection.

(iii) Magnetic Particle Inspection.

3.1.4 Grade B welds shall be subjected to the Basic Inspection Techniques of Para 3.2 and such other tests as may be required by the Aircraft Design Authority.

3.2 BASIC INSPECTION TECHNIQUES

3.2.1 The basic inspection techniques include the following:

(i) the verification of correct application of the specified preparatory and welding techniques by destructive and non-destructive inspection of representative test specimens at prescribed intervals during weld production.

(ii) the visual examination of each weld produced, including fillet size, toe length, smoothness etc.

3.3 SUPPLEMENTARY INSPECTION PROCESSES

3.3.1 A number of further inspection processes are available, namely:

(i) Proof-loading - This is a suitable method for items like control rods, drive shafts, ducting etc.

(ii) Pressure testing of appropriate welded components.

(iii) Magnetic Particle Inspection - Applicable to steels other than austenitic stainless steels.

(iv) Penetrant Inspection - Applicable to all parts not covered by (ii). It may also be applied to parts made from magnetic materials as a duplicate inspection, for example, to find surface cracks in areas that are difficult to check magnetically, such as multiple tube joints.

(v) Radiographic Procedures, including the preparation of detailed techniques for each item and the scrutiny of radiographs.

(vi) Processes such as ultrasonic and eddy current testing may also be developed for the examination of welds.

(vii) For electron beam welding, consideration should be given to the use of scribed lines on each side of the fraying edge parallel to the joint on both the top and underside of the weld, to determine if the weld is offset with respect to the joint.

4 INFORMATION TO BE QUOTED ON DRAWINGS

4.1 The following is a suggested list of information to be quoted on the detail drawings or in documents related to the drawing as required by Chapter 402:

(i) the Grade of the weld (See Para 3. 1).

(ii) the specification and condition of the material prior to welding.

(iii) details of joint preparation and the location (and sometimes the size) of the weld.

(iv) details of any stress relieving or other heat treatment to be applied after welding (See Para 2.10).

(v) the method(s) of inspection to be applied (See Para 3) and the acceptance standards to be achieved, including any inter-stage inspection.

(vi) any routine control tests to be applied during production and the properties to be achieved.

(vii) the welding process to be used and the welding parameters.

(viii) details of the filler rod or electrode, if used.

(ix) details of any special weld precleaning treatment.

(x) if preheating is to be used, the required temperature and method.

(xi) any special methods of cooling after welding.

(xii) cleaning process, where necessary, for the removal of flux.

4.2 Standard methods of indicating welding requirements on drawings are described in BS 499.

5 NOTES ON PARTICULAR WELDING PROCESSES

5.1 OXY-ACETYLENE WELDING

5.1.1 Recommended for use on low alloy steels not more than 1.6 mm thick, but difficult below 0.33 mm thick. Should not be used for material greater than 3.2 mm thick.

5.1.2 Can be used for stainless steel and aluminium and magnesium alloys with the use of a flux, but an alternative fluxless method eg TIG welding (See Para 5.3) is recommended.

5.1.3 Not suitable for reactive metals e.g. titanium.

5.2 METAL ARC WELDING (STICK ELECTRODE)

5.2.1 Recommended method for welding low alloy steels greater than 2.5mm thick, as penetration is good. Should not be used for material less than 1.2 mm thick.

5.2.2 May also be used for stainless steel. If the thickness of stainless steel is greater than 6.5 mm then manual metal arc welding is preferable to TIG welding. Nevertheless, in many cases TIG welding will be preferable to manual metal arc welding since there is no slag to remove between runs or after completion of welding.

5.2.3 It is necessary to remove the flux between each run and after completion of welding.

5.3 TUNGSTEN ARC INERT GAS SHIELDED (TIG) WELDING

5.3.1 Suitable for all metals normally accepted as weldable and for a wide range of thicknesses eg 0.25 mm to 12.5 mm with multi-run techniques. For aluminium and magnesium alloys, the thinner gauges of this range may require special welding techniques. The process can also be used below 0.25 mm but the preferred process is then micro-plasma, (See Para 5.7).

5.3.2 Recommended method for manual welding of stainless steel, aluminium, magnesium, titanium, nickel and copper base alloys. When welding titanium and its alloys by the TIG process, special precautions are always advisable. These precautions involve the use of backing gas, trailing and leading gas shields, or ideally, conducting all welding in a chamber filled with inert gas (usually argon).

5.3.3 May be used for welding low alloy steels, but particular attention needs to be given to precleaning, eg by shot blasting, and it is normally necessary to re-shot blast between runs to avoid porosity. When such precautions are applied, high quality welds may be obtained and the method is recommended for critical welds in low alloy steels.

5.3.4 The shroud of inert gas prevents contamination of the weld from the atmosphere and greatly reduces and can eliminate scaling.

5.3.5 Some welding torches are bulky, and thus the method is only suitable where the design can be arranged to provide adequate access.

5.4 TIG SPOT WELDING OR PUDDLE WELDING

5.4.1 This is an adaptation of TIG welding (See Para 5.3) which produces a fused "spot" weld for joining sheets. A top part in the thickness range 0.4 mm to 1.6 mm can be welded to an under-component of like or greater thickness. More than two sheets can be welded together.

5.4.2 Useful for "spot" welding where only one side of the joint is accessible. Care should be taken to prevent interface oxidation when applying this weld.

5.4.3 Suitable for stainless steel and titanium alloys and certain alloy steels. When welding titanium alloys, special precautions may be necessary to protect the underside of the weld with an inert gas to prevent contamination.

5.4.4 Joints made by this process have relatively poor fatigue characteristics and are prone to corrosion problems. Weld nuggets should be inspected for the presence of centre cracking.

5.4.5 Not suitable for aluminium alloys.

5. 5 METAL ARC INERT GAS SHIELDING (MIG) WELDING

5.5.1 Suitable for similar applications as TIG welding but enables faster rates of welding. The torches are more cumbersome than TIG welding torches.

5.5.2 MIG welding is an alternative to manual metal arc welding since both processes are generally used on relatively thick sections ie greater than 3 mm.

5.5.3 Most suitable for machine welding of relatively long runs, but is not suitable for materials 1.6 mm (0.063in) and thinner.

5.6 $C0_2$ WELDING

5.6.1 $C0_2$ is widely used as a shielding gas for gas metal-arc welding of low carbon and low alloy steel.

5.6.2 CO_2 is an oxidising gas so the electrodes used must contain larger amounts of deoxidisers such as Si and Al than are required for argon-base shielding gases.

5.6.3 The short circuit or dip transfer mode of CO_2 welding gives the process a greater degree of application than MIG welding because of the ability to carry out positional work. The spray transfer of MIG welding is essentially non-positional ie flat or down-hand.

5 .7 PLASMA ARC WELDING

5.7.1 This is a development of TIG welding (see para 5.3) and produces a very high heating intensity which produces deep penetration welds eg full penetration in thicknesses up to 6.3 mm can be achieved in a single run. It is also particularly suitable for very thin gauges eg 0.025 mm where the process is known as micro-plasma.

5.7.2 Suitable for specialised applications for all weldable materials except aluminium alloys. Although it is possible to weld aluminium and aluminium alloys using, this method, it is only accomplished with difficulty and is not recommended.

5.8 VACUUM AND PARTIAL VACUUM ELECTRON BEAM WELDING

5.8.1 Electron beam welding is suitable for all metals, but special techniques may be necessary for those containing low melting point/high vapour pressure elements, eg zinc. Some alloys usually considered "unweldable" by other techniques can be welded. In these cases, the properties of the welded structure should be subsequently evaluated by test.

5.8.2 Welds may have very narrow width and are generally narrow in relation to depth of penetration; the heat affected zones are also very narrow. It is possible to obtain very deep penetration, eg 50 mm in one run, and hence a wide range of thickness, eg 0.025 mm to 50mm can be welded.

5.8.3 It is possible for some special applications, to weld together different base metals, eg Monel to soft magnetic iron, although in some cases a shim or plated coating may be used to provide a fusion zone which is metallurgically compatible with the dissimilar base metals.

5.8.4 Recommended for special applications requiring very high quality welds and/or minimum distortion, and where access to the joint is difficult for other methods of welding.

5.8.5 As welding takes place in a vacuum, weld contamination is negligible.

5.8.6 Assemblies for welding have to be capable of being located and mechanically traversed in a vacuum chamber under the electron beam.

5.8.7 Close dimensional tolerances of detail parts are required to ensure a good fit at the weld point; normally gaps of less than 0.075 mm are required.

5.8.8 Where possible, ferromagnetic components, should be demagnetised prior to welding.

5.9 NON-VACUUM ELECTRON BEAM WELDING

5.9.1 Similar characteristics to vacuum electron beam welding (see Para 5.8) but the process can be carried out at ambient pressure, eg in argon shroud or even in air and so the weld is more likely to be contaminated. The resultant weld is likely to be wider than when made by vacuum beam welding.

5.9.2 The weld position may be up to 25.4 mm from the electron gun.

5.9.3 The thickness of material which can be welded out of vacuum is considerably less than can be welded with the same power under vacuum.

5.10 FRICTION OR INERTIA WELDING

5.10.1 Joints are characterised by a narrow zone of upset heat affected structure, and the absence of fused material.

5.10.2 The process is suitable for all metals which can be welded by fusion techniques. Some alloys considered unweldable by fusion techniques can be welded by this process. In these cases the properties of the joints should be fully evaluated by test.

5.10.3 It is possible to join different base metals together.

5.10.4 The design of the assembly should be such that access is provided for removal of the upset flash by machining. Removal of the upset flash prior to post-weld heat treatment minimises the risk of cracking during heat treatment.

5.10.5 Local thickening may occur during welding. Thus a limited amount of post-weld machining may be required in addition to flash removal. With good tooling any mismatch should not exceed 0.125 mm.

5.10.6 The maximum cross sectional area which can be joined is limited by the tonnage capacity of available machines.

5. 11 DIFFUSION BONDING

5.11.1 Diffusion bonding is a process used to produce high integrity joints, particularly in titanium but also in dissimilar materials.

5.11.2 It is a process by which materials can be joined in the solid state by atomic diffusion across the joint interface.

5.11.3 Where such a process is required, specific test work to evaluate the process/part will be necessary.

5.11.4 For certain alloys (eg Ti-6A1-4V), diffusion bonding can be combined with a superplastic forming technique.

6 STATIC STRENGTH OF WELDING JOINTS

6.1 The static strength of a welded joint depends primarily on two factors, namely, the strength of the weld metal and the strength of the heat affected zone of the parent metal adjacent to the weld. With undressed welds in material of 1.6mm and thinner the effect of weld reinforcement is such that the weakest part is normally the heat affected zone or parent metal. A drop in strength occurs in the heat affected zone of work hardened or age hardened materials. As the thickness of the materials being joined increases, the strength of the weld metal becomes more important, and above 3 mm is normally the most important factor. Table 3 gives some typical 'as deposited' weld metal strengths.

6.2 Minimum welded strengths for undressed welds are quoted in some tube specifications, but it should be noted that the method of welding is not specified.

6.3 The Metallic Materials Data Handbook (Def Stan 00-932) should be consulted for the allowable stresses in the 'as-welded' condition.

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6	Dinsdale W 0 & Scott M H	An assessment of the weldability, tensile and fatigue strength of Titanium 8% A1-1% Mo-1% V. BWRA Report C129/2/66, June 1966.
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TABLE 1

WELDING PROCESSES v HEATING INTENSITIES

(see para 2.13)

Oxy-acetylene	↑
TIG	
MIG	Increasing extent of
Metal arc	heat affected zone.
Plasma arc	
Electron beam	

TABLE 2

MATERIALS NORMALLY ACCEPTED AS WELDABLE WITH RECOMMENDED FILLER RODS FOR TIG WELDING

Form	Specification	Recommended filler rod
ALUMINIUM ALLOYS		
Sheet, Strip and plate	L113	BS 2901: Pt 4: 1970: 4043A or 5056A
	L115	" " "
Forging	L112	" " "
Bar	L111	" " "
Tube	L114	" " "
	L118	" " "
Casting	2L99	" : 4043A or 4047A
MAGNESIUM ALLOYS		
Bar	2L508	BS2901: Pt 4: 1970: MAG 141
Casting	DTD 5005	" : MAG 8
	DTD 5055	Parent metal
CORROSION AND HEAT RESISTING STEELS		
Sheet and strip	S524	BS 2901: Pt 2: 1970: 347 S96
	S525	" "
	S526	" "
	S527	" "
	S531	" : 313 S96
	S532	Parent metal
	S533	Parent metal
	S536	BS 2901: Pt2: 1970: 308 S92
	S537	" : 316 S16
	S538	*
Bar	S126	BS 2901: Pt 2: 1970: 311 S94
	S129	" : 347 S96
	S130	" "
	S143	Parent metal
	S144	Parent metal
	S151	*
Tube	T66	BS 2901: Pt 2: 1970: 347 S96
	T67	" "
	T68	" "
	T69	" "
	T72	" "
	T73	" "
	T74	: 308 S92
	T75	: 316 S92
Casting	HC101	*
	HC102	*
	HC104	BS 2901: Pt 2: 1970: 347 S96
	HC105	" : 318 S96
	HC106	*

TABLE 2 (Continued)

Form	Specification	Recommended filler rod
CARBON AND LOW ALLOY STEELS		
Sheet, and Strip	S534	BS 2901: Pt 1: 1970: A32
	S535	" "
Bar	S142	" "
	DTD 5082	" "
Tube	T53	" : A31 or A32
	T60	" : "
	T65	" "
NICKEL ALLOYS		
Sheet and Strip	HR203	BS 2901: Pt 5: 1970: NA 35
	HR206	Parent metal
Bar	HR5	BS 2901: Pt 5: 1970: NA 34
	HR10	" : NA 38
Tube	HR403	" : NA 34
	HR404	" : NA 38
Casting	HC205	" : NA 38
	HC206	" : NA 38
OTHER HEAT RESISTING ALLOYS		
Sheet	HR207	BS 2901 : Pt 5: 1970: NA 42
	HR240	Parent metal
Bar	HR40	Parent metal
	HR55	BS 2901 : Pt 5: 1970: NA 42
TITANIUM ALLOYS		
Sheet and strip	TA1	Parent metal
	TA2	"
	TA6	"
	TA10	"
	TA21	"
Plate	TA58	"
Forging	TA13	"
	TA24	"
	TA44	"
	TA55	"
Bar	TA11	Parent metal
	TA22	"
	TA53	"
Tube	DTD 5073	"
Casting	DTD 5363	"

* No standard fillers, use austenitic type + e.g. T308 or matching fillers (choice depends on requirements).

\+ Austenitic type should not be used in cases where thermal fatigue is a possibility.

TABLE 3

TYPICAL "AS DEPOSITED" WELD METAL STRENGTHS

Values to be used only as a guide - see para 6.1

Method of Deposition	Filler Rod		Yield Stress (MPa)	Tensile Strength (MPa)	Elongation on 50 mm %	Izod (J)	Hardness HB
ALUMINIUM & ALUMINIUM ALLOYS							
Gas	BS 1453	G1B	40	78.7	38	17	14
		NG2	88	152.8	6	4	47
		NG6	100.4	216	12	7	58
		NG21	57.2	114.2	7	5	38
MAGNESIUM ALLOYS							
Gas	BS 1453	D1	108	151.3	1.2	1	71
		D2	29.3	101.9	6.8	7	43
STEELS							
Gas	BS 1453	A1	261	386	32	81	121
		A2	352	491	31	74	145
			284	429	22	30	146
		A3	278	438	22	69	150
		A4	364	538	23	28	202
		A5		821	2	9	286
		A6	349	503	19	61	180
TIG	BS 2901	A8 NG	528	671	41	-	-
COPPER & COPPER ALLOYS							
Gas	BS 1453	C1	63	231	45	54	52
		C2	167	449	22	40	117
		C4	221	463	12	27	112
		C5	268	579	16	42	151

Note: Information based on data supplied by British Oxygen Company Limited.

LEAFLET 402/2

PROCESSES AND WORKING OF MATERIALS

ADHESIVE BONDING OF STRUCTURAL PARTS - PROCESSES AND CONTROL

1 INTRODUCTION

1.1 The purpose of this Leaflet is to give, for the information of designers, a summary of the considerations and recommendations affecting the use of adhesives for the bonding of structural materials. It is not a complete process document but should be taken into consideration when preparing a Process Schedule; it does not override any relevant mandatory document.

2 SURFACE PREPARATION

2.1 GENERAL

2.1.1 Silicone and fluorocarbon compositions, e.g., aerosols, greases, oils, release agents and rubbers should not be kept or used in work areas where any part of any adhesive bonding process is carried out. Any contamination by these materials of items to be adhesively bonded is almost impossible to remove and complete removal cannot be assured. It may be necessary, however, for certain materials of the above type to be used in specific applications, and in such cases prior approval should be obtained from the Design Authority.

2.1.2 Specially prepared papers impregnated with hardened silicone resins, e.g., as used for the protection of 'pre-pregs' and film adhesives, are acceptable but it should be verified that silicone transfer cannot take place. These papers should not be used for any purpose other than that for which they were intended and are to be removed immediately from the work area.

2.1.3 Surfaces which have been prepared for adhesive bonding are very sensitive to any contamination and this will have an adverse effect on bond performance. It is essential that care should be taken at all times to avoid contamination by hand, clothing, protective wear, cleaning and wrapping materials, PVC bags, vapours from release agents and dirty working areas. Despite all precautions for protection it should be noted there is a limiting time between surface preparation and application of bonding primer or bonding. Suitable protected prime items may, however, be stored in clean, dry conditions for periods as specified for particular primer systems.

2.2 CLEANING AND DEGREASING OF METAL SURFACES

2.2.1 All aluminium and aluminium alloys, and corrosion resisting steels, should be degreased in accordance with the requirements of DEF STAN 03-2/1 Methods A1 or A2 or B1 or B2.

2.2.2 Titanium CP and titanium alloys should be vapour degreased in accordance with the requirements of DEF STAN 03-2/1 Method A1 using 1.1.1 - trichloroethane.

2.3 ALUMINIUM AND ALUMINIUM ALLOYS

2.3.1 After degreasing, surfaces should be pickled in chrome/sulphuric acid in accordance with DEF STAN 03-2/1 Method O.

2.3.2 When anodising is required the aluminium alloys should, after degreasing, be pickled in accordance with DEF STAN 03-2/1 Method O. This should be followed by chromic anodising, in accordance with DEF 151 Type 2, or by alternative anodising processes approved by the Rotorcraft Project Director.

2.4 CORROSION RESISTING STEELS

2.4.1 Subsequent to cleaning and degreasing, surfaces should be etched in a 30% solution of sulphuric acid or other process approved by the Rotorcraft Project Director.

2.5 TITANIUM CP AND TITANIUM ALLOYS

2.5.1 Information on approved processes for pretreating these materials may be obtained from the Defence Research Agency, Farnborough.

2.6 OTHER METALS

2.6.1 If metals other than aluminium, corrosion resisting steels and titanium CP or titanium alloys are to be bonded special treatments will be necessary. Information may be obtained from the Defence Research Agency, Farnborough.

2.7 FIBRE REINFORCED POLYMERS

2.7.1 Surfaces should be swab degreased with an approved cleaning fluid for that material to remove any parting agent or surface contamination and then carefully abraded, with removal of all resultant dust and debris, to ensure a clean matt surface suitable for bonding.

2.7.2 Surface water content of composites can affect bond strength. Composite components to be bonded should be dried prior to bonding until the total moisture content is less than 0.4 per cent by weight.

2.8 HONEYCOMB

2.8.1 Uncoated metal honeycomb - After machining operations and prior to bonding, metal honeycomb should be degreased. Repeated immersion may be necessary for effective vapour degreasing. Any materials, such as wax or temporary adhesive, used to support the core during machining should be removed by a specified and controlled process prior to final degreasing for bonding.

2.8.2 Non-metallic and resin coated honeycomb - The honeycomb must be clean and dry, and may require oven drying immediately prior to bonding. Any feathered ends formed when the honeycomb is cut should be removed. Cutting swarf and dust should be removed by blowing with clean dry air or by vacuum cleaning.

3 APPLICATION OF PRIMER AND/OR ADHESIVE

3.1 It is strongly recommended that primer or liquid adhesive be applied or bonding accomplished within four hours of prebond treatment. All items awaiting priming or bonding should be kept in a clean area. Where a film adhesive is used without a primer and assembly cannot be completed within four hours the process should be demonstrated to be adequate to the satisfaction of the Rotorcraft Project Director.

3.2 Consideration should be given to the possibility that with certain adhesives, water absorption prior to bonding may have an adverse effect on the properties of adhesive bonded components.

4 BONDING

4.1 The fit of component details should be such that a solid glueline free of gaps and voids and of a substantially uniform thickness is achieved. Actual glueline thickness achieved will depend on the particular adhesive used, the type of component (e.g., honeycomb or plate to plate), the bonding pressure and the area of plate to plate joints. Flow of adhesive away from the central region of large area plate to plate bonds will be limited and the problem more apparent in autoclave processes. Development components should be used to study fit and flow of adhesive.

5 PROCESS CONTROL

5.1 A complete Process Schedule should contain requirements regarding materials, preparation of surfaces, application of the adhesive, bonding and subsequent treatment tests required, including non-destructive inspection techniques and the keeping of records and tests values and, for Grade A parts, be approved in accordance with DEF STAN 05-123, Chapter 107.

5.2 Coupon test pieces will normally be bonded with the work piece. The methods of preparation and testing should be in accordance with Specification DTD 5577 or other approved MOD testing schedule. The tests required are normally overlap shear, but peel tests may also be specified. When the adhesion quality of plate to honeycomb joints is measured by a peel test method due allowance should be made for the work done in bending the metal skin. The peeling torque will change with variation in the stiffness of the metal skin even though adhesion may not have changed. Tests should be carried out in accordance with DTD 5577 or to other Process Schedules approved by the Rotorcraft Project Director. Where hot/wet durability is required, a wedge test (eg ASTM-D-3762-79) may be used to assess joint performance.

5.3 It is essential that full records are kept of all stages in the process. The records should show for each component:

(i) details of cleaning and pretreatment (including time of immersion and withdrawal, and temperature of the acid pickling bath),

(ii) time of pretreatment and application of the adhesive or primer,

(iii) time between application of the adhesive and the bonding process,

(iv) the complete time, temperature and pressure history of the cure cycle,

(v) results obtained on test pieces processed with the components,

(vi) any non-destructive inspection results obtained on the component.

Cumulative elapsed times that adhesives are without refrigerated storage and, for two part adhesives, the date and time of mixing the components should also be recorded.

5.4 The tests of Para 5.2 should be supplemented by periodic tests on components or on specimens cut from components. Records of all tests should be kept, preferably in chart form, showing the comparison between results obtained on the coupon test pieces and on components or specimens cut from components. The use of statistical quality control methods to control the process is recommended (see also FAA Quality Control Digest No 5, Bonding Inspection).

LEAFLET 402/3

PROCESSES AND WORKING OF MATERIALS

ADHESIVE BONDING OF STRUCTURAL PARTS - RECOMMENDED DESIGN PRACTICE

1 INTRODUCTION

1.1 This leaflet gives recommendations on the design and strength of structures with approved adhesives.

2 GENERAL DESIGN CONSIDERATIONS

2.1 The design requirements for the application of any form of bonding process to Grade A structures are given in Chapter 402, para 1.

2.2 There are two main types of bond used in adhesive bonded structures, namely plate to plate and plate to honeycomb, and a bonded structure may be effected by using either one or a combination of both types of joint.
2.2.1 Typical applications for bonded joints include the attachment of stringers to skins, stiffeners to rib webs, shear plates, reinforcing plates bonded around cutouts and edges to reduce stress concentration and to permit cut countersunk riveting.

2.2.2 Typical examples of adhesive bonded sandwich structures are for bulkheads, flooring, doors and control surfaces. Construction of the components provides stiff lightweight structures where tensile and compression loads are carried by the facing skins and resistance to acoustic fatigue, bending and buckling is afforded by the bonded honeycomb.

2.3 Detailed consideration is recommended at the design stage in matching the requirements of joint manufacture with the requirements of the joint and component design. When designing sandwich structures, consideration should be given to the bonding operations which must be effected with suitable bonding pressures to prevent collapse of the core. Since the integrity of the structure is dependent on the bonding of the honeycomb cell walls to the skins with which they are in contact, the adhesive should have adequate filleting and gap-filling properties. The components dimensions have to be closely controlled and due allowance made to accommodate the thickness of the adhesive layer. Foaming adhesives may be used in bonding and consolidating honeycomb structures and it should be noted that the expansion properties of the adhesives enable them to fill relatively large gaps. Adequate precautions should be taken to seal sandwich structures against the ingress of water to minimise deterioration.

2.4 Due account should be taken of differing thermal properties when bonding dissimilar materials or sheets of different thickness.

2.5 The substitution of adhesive bonding for riveting, welding or other jointing procedures without a re-design of the structure is unlikely to result in a satisfactory design.

2.6 The designer should satisfy himself that the design values used take into account degradation due to environmental ageing. Exposed edges of adhesive layers should not lie unprotected in contact with water/fluids in undrained parts of the structure.

2.7 In the design of honeycomb structures, due allowance should be made for damage tolerance in the presence of internal pressure and acoustic pulses.

2.8 As peeling and cleavage stresses are unevenly distributed through a bonded joint, they can under working conditions cause premature failure, and anti-peel devices should be incorporated in likely cleavage situations. Likewise the strength of an adhesive in tension, (e.g. butt joint), normal to the plane of the joint, is very sensitive to the quality of the bond, uniformity of loading and level of stress concentrations. Adhesive bonded joints should therefore be designed to minimise stress concentrations in shear, cleavage, peel and tension stresses.

2.9 The fatigue characteristics of bonded joints relative to their static strength will be influenced by joint design and the adhesive used.

2.10 Design should not include the bonding of previously mechanically fastened or welded sub-assemblies which may allow entrapment of pretreatment solutions.

2.11 The adhesive is normally electrically insulating, and it will therefore be necessary to provide electrical connections between the parts to comply with the requirement of Chapter 708, para 3.1.2.

2.12 The load distribution of sandwich structure to the main structure is normally achieved by conventional methods of riveting or bolting through reinforced edge members or other reinforced areas. These joints must comply with the requirements of Chapter 402, para 1.

2.13 As aluminium honeycomb core material is itself a bonded item, it is important to verify that the curing cycle/cycles applied when bonding the structure does not permanently degrade the honeycomb nodebond adhesive and thereby reduce the structural efficiency of the honeycomb structure.

3 CHOICE OF ADHESIVES

3.1 It should be noted that certain structural adhesives are intolerant of multistage cures and this aspect should be checked with the manufacturer and Approving Authority.

4 STRENGTH CONSIDERATIONS

4.1 In order to predict the failing strength of a bonded joint with an adhesive loaded in shear, it is advisable to perform a detailed load diffusion calculation taking into account the shear modulus of the adhesive and the stiffness of the components being joined. This calculation is to find the peak shears which occur at the extremes of the joint because they determine the ultimate strength. There are no simple rules to estimate these peak shears relative to the average stress in the joint. As a general guide to the allowable design values for these peak shears the results of single lap shear specimens, such as are used for process control purposes, can be used directly since the average failing shear on a short overlap specimen is almost equal to the peak shear, especially if there is some plasticity in the adhesive. In the case of large area plate to plate bonds, where ventilation holes cannot be provided, inwards migration of the adhesive can result in a thick glueline. Account should be taken of the reduction in strength and stiffness of the thicker layer.

4.2 Care should be taken in structures designed to buckle that such buckling does not lead to peel failure.

4.3 Soft tack rivets should not be regarded as contributing to the static strength of the joint.

5 TEMPERATURE CONSIDERATIONS

5.1 Adhesive-joints are temperature/time sensitive. The maximum temperature which the joint is likely to achieve from all sources of heat, including kinetic heating is therefore to be determined and due allowance made in strength calculations or in the interpretation of test data for the associated reduction in bond strength incurred for the time at which the structure is at that temperature.

5.2 When practicable, strength data is to be obtained from structural tests under the most adverse conditions of load and temperature/time that will occur in operation.

5.3 When sandwich structures are designed hermetically sealed to exclude moisture then the effect of change of pressure in the structure with change of temperature and altitude should be allowed for in the strength calculations of the component.

5.4 The coefficient of thermal expansion of the adhesive is greater than that of metal parts of the assembly but may be the same as, or similar to that of non-metallic parts.

6 HANDLING AND MANUFACTURING CONSIDERATIONS

6.1 Control of the adhesive glueline thickness is achieved by controlling the dimensional tolerances of the constituent parts of the assembly being bonded, the tools heat-up rate and bonding pressures applied during the curing cycle.

6.2 Unsupported edges of adhesive bonded joints are particularly susceptible to damage, and the use of tack rivets to prevent damage at such edges is recommended even when subsequent assembly procedures would provide adequate protection. Location by one or more tack rivets may be necessary to ensure adequate positioning during the curing process. The tack rivets should be loose in order to avoid variable glue line thickness. It should be noted that rivets can provide a means of water ingress into the bonded area, and that with brittle adhesives, riveting after the adhesive bond has cured, may result in fracture damage to the adhesive. These effects may adversely affect the fatigue life and durability of the joint and, therefore, appropriate protection should be given.

6.3 As a bonding process may involve hot curing, the jigged assembly requires to be of a size and shape that will permit curing of the adhesive under the specified conditions of temperature and pressure. The component should be suitably jigged so that the bonding surfaces can be of an intimate fit with the adhesive before application of the bonding pressure.

6.4 It is normal to bond edge reinforcements at the same time as the core to skin bond is effected in a honeycomb structure. If the edge attachments or further skins are secured as a second operation, the exposed bond at the skin to core edge should be protected until the operation has been completed. Where multi-stage curing is carried out, adequate support and protection should be given to previously bonded joints and subsequently bonded faces.

6.5 Adhesive bonded sandwich should never be manipulated after bonding, and it is only possible to manipulate bonded plates under closely controlled conditions which have previously been demonstrated to be satisfactory.

LEAFLET 402/4

PROCESSES AND WORKING OF MATERIALS

THE EFFECT OF SURFACE FINISHING AND PROTECTIVE TREATMENTS ON FATIGUE PROPERTIES

1 INTRODUCTION

1.1 The severe reduction of fatigue properties caused by stress concentrations associated with notches, sharp radii and poor surface finish is generally known and appreciated. Less well known, however, are the effects of various surface and protective treatments applied to finished machined parts. Some of these treatments impart fatigue properties considerably lower than those of the polished machined test pieces by which materials are normally evaluated. Although the effects of surface treatments are of a second order compared with those of holes, small radii and poor machining, they become increasingly important as the basic fatigue design is improved.

1.2 Fatigue properties are affected by surface treatment in many ways. In particular, those which introduce residual surface stresses modify the fatigue performance under some conditions of loading, compressive stresses being beneficial, and tension stresses deleterious. Treatments which roughen the surface tend to reduce fatigue strength. Some chemical and electrochemical treatments cause absorption of hydrogen by the metal which may have a harmful effect.

1.3 The following information, based, it should be emphasised, in all cases on plain unnotched test specimens, is intended as a help to designers in considering the effects of post-machining treatments. The importance of the effects is usually smaller on notched specimens, but data are meagre. The changes in fatigue properties quoted in this Leaflet are the changes in alternating stress required to produce failure at a number of cycles of the order of 10 million.

1.4 Many of the treatments which reduce fatigue properties are very desirable for protection against corrosion, and give much enhanced fatigue properties under corrosive conditions. The data given in this Leaflet do not imply a recommendation to omit protective treatments.

2 ALUMINIUM ALLOYS

2.1 EXTRUSIONS AND FORGINGS

2.1.1 The fatigue properties of "as extruded" and "as forged" surfaces are up to 50% lower than those of polished machined test specimens.[1] Chemical pickling has little deleterious effect on these surfaces, the deleterious effect of roughening being apparently balanced by the benefit arising from removal of the original surface, but chemical pickling by many solutions, (eg, caustic soda), reduces the fatigue properties of machined surfaces.

2.2 SHEET

2.2.1 The fatigue strength of clad high strength sheet alloys may be up to 40% less than that of unclad sheet of the same total thickness.[2] This reduction is much greater than can be explained merely by the reduced thickness of the high strength core material, and is usually ascribed to a special influence exerted by the low strength surface cladding.

2.2.2 The removal of the "as rolled" surface from unclad thick sheet, by chemical means, causes some reduction in fatigue properties.[3]

2.3 ANODISING

2.3.1 Normal anodising has little effect on fatigue properties except at holes, where a reduction may be caused. Hard anodic films 0.025 mm and more thick, however, reduce fatigue properties of Wohler type test pieces of aluminium copper alloy by up to one half.[4] This effect can be reduced to a 10% reduction by a sealing treatment in boiling 5% potassium dichromate solution.[5] The sealing treatment, however, causes a 20% reduction in the abrasion resistance of the coating.

2.4 ABRASIVE BLASTING

2.4.1 Under reversed bending fatigue, the compressive surface forces induced have a good effect, and any roughening has a bad effect. Mild blasting treatments which leave a satiny finish have been found to have a nett beneficial effect. Heavy abrasion such as grit blasting would probably have an effect depending on the sharpness of the grit; the overall effect with very sharp grit being probably deleterious.

2.4.2 Under axial fatigue, compressive surface forces give little benefit, except at radii and notches. Roughening from abrasive is detrimental.

3 MAGNESIUM ALLOYS

3.1 The normal chromate treatments have little effect on fatigue properties.[6] The thick, hard film deposited by the H.A.E. process and the DOW 17 anodic process, however, reduces the fatigue strength, an effect analogous to that of hard anodic films on aluminium alloys. The observed reduction can be up to one third,[7] but this will vary with alloy type and should be verified by the design authority.

4 STEELS

4.1 Electropolishing treatments have been shown to reduce the fatigue properties of conventional machined Wohler type test pieces considerably.[8, 9] Non-electrolytic treatments which dissolve the surface layers of the metal would probably have the same effect. The fatigue properties can be restored by shot-peening or nitriding or partially restored by light abrasive blasting.[10] Results on stainless steels suggest that the effect of electropolishing on fatigue properties of sheet is not large.

4.2 Cadmium and zinc plating have little effect on the fatigue properties of medium strength carbon steels.[11] Tests on high tensile steels, however, show that the fatigue properties of heat treated alloy steels of conventional high strength types having an ultimate tensile strength greater than 1235.5 MPa can be lowered by the standard anodic pickling and cadmium plating treatment even when followed by baking.[12] The harmful effect can be avoided by plating the cadmium on to a grit blasted surface without any interposed chemical cleaning treatment.[13]

4.3 Hard chromium plating causes a reduction in the fatigue properties of steel, the reduction tending to be greater on stronger steels. On a low alloy steel heat treated to an ultimate tensile strength of 1235.5 MPa, a 40% reduction in limiting fatigue strength is caused by a deposit of 0.025 mm chromium. The harmful effect is increased by a post plating heat treatment to 150°C. to 200°C., but is removed by heating to 450°C. to 500°C. Deleterious effects can be avoided by prior shot-peening.[14]

4.4 Nickel plating can also cause an appreciable reduction in fatigue properties. The effect varies with the type of plating solution used, and is related at least in part to the magnitude and sign of stresses in the plated layer.[14]

4.5 The beneficial effect of suitably controlled shot-peening on the fatigue properties of high tensile steels is well known. Grit blasting has also been found to have a similar beneficial effect, but the subsequent aluminium spraying for which it is a pre-treatment, reduces the fatigue properties to those of "as machined" metal.

4.6 Approved prebond cleaning and etch processes for metals may have been shown to have adverse effects on fatigue properties.

4.7 Surfaces of steel components should be checked for grinding burns before any protective treatment is applied.

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LEAFLET 402/5

PROCESSES AND WORKING OF MATERIALS

COATING OF METALS WITH PLASTICS MATERIALS

1 INTRODUCTION

1.1 This Leaflet draws attention to the possible loss of strength which may result from coating metal parts with polyvinyl chloride, polythene, nylon or other plastics materials.

2 METHOD OF APPLICATION OF COATING

2.1 The coatings are applied by spraying or, more frequently, by dipping in a fluidized bed of the powdered material, after heating the part concerned to a temperature above the melting point of the coating material. This temperature will depend not only on the coating material but also on the thickness of the coating required and on the heat capacity of the part. In practice, the temperatures may range from about 100°C up to 400°C.

3 EFFECT OF COATING ON STRENGTH OF MATERIALS

3.1 The temperatures involved in the application of the coating may cause a loss in mechanical properties in some metals. High strength aluminium alloys, are at greatest risk, as they are commonly given a final heat-treatment in the range 130°C to 190°C. High strength steels can also be affected, depending on the tempering temperature which can be as low as 200°C. Titanium alloys however are normally unaffected by temperatures below 400°C.

3.2 Hence it is essential for the designer to ensure that the temperature employed in any coating process should not be such as to cause an unacceptable deterioration in the properties of the material. He should also ensure, as required by Chapter 402, para 8, that the maximum temperature to which each part nay be subjected is stated on the drawing or order.

LEAFLET 402/6

PROCESSES AND WORKING OF MATERIALS

BRAZING AND SOLDERING

1 INTRODUCTION

1.1 This leaflet gives data and recommendations on the use of brazing and soldering.

1.2 If a significant departure from the recommendations of this leaflet is proposed for a Grade A part, the proposal should be discussed with the Rotorcraft Project Director.

2 BRAZING GENERAL

2.1 The term brazing is used to describe the joining of metals by means of a filler material without melting the parent metal. The filler when molten is normally drawn by capillary action into the space between closely adjacent surfaces of the parts to be joined. The composition of the filler material is not necessarily similar to that of the parent metal and the melting point is normally higher than 500°C, but lower than that of the parent metal.

2.2 In the case of bronze welding (braze welding) a fillet is produced by a technique similar to fusion welding using a filler material with a lower melting point than the parent metal but without using capillary action or intentionally melting the parent metal.

2.3 The heating to achieve this may be attained by any means, providing the temperature and, in some cases, the atmosphere surrounding the joint can be adequately controlled. Available processes are listed in Para 6.

3 EFFECT OF BRAZING ON MATERIAL PROPERTIES

3.1 The temperatures required for brazing, are such that heat treated and work hardened materials normally suffer reduction in mechanical properties and the designer should make due allowance for such reductions in strength. These changes may be localised when torch or induction brazing is used, but affect all the material after salt bath or furnace brazing.

3.2 The melting temperature of the brazing filler material sometimes precludes subsequent full heat treatment to restore the properties although this is possible in certain special cases such as the copper brazing of steel. Post-braze precipitation treatments of some aluminium and nickel base alloys may be used to restore the properties of the parent metal.

3.3 Tests made to determine the effects of copper brazing and subsequent heat treatment upon the impact, tensile and fatigue properties of a number of heat treated steels, weldable low alloy steels and plain carbon steels showed that the impact resistance of almost all the steels was reduced by the brazing cycle. The ultimate tensile and proof stresses of heat treated low alloy steels and cold worked steels are also reduced by the brazing cycle. Some form of post-brazing treatment to restore properties is essential for heat treated low alloy steels and desirable for plain carbon steels. The properties of cold worked materials cannot be restored after brazing.

3.4 High nickel alloys, corrosion resistant steels and plain carbon and carbon-manganese steels may be subjected to intergranular penetration by the filler alloy if brazed in a state of stress (See Paras 6.2, 7.2, 7.3. 2 and 7.5.2.)

4 DESIGN AND STRENGTH OF JOINTS

4.1 The strength and efficiency of brazed joints depend on a number of factors, including the design of the joint, the method of applying the process, the composition of the materials to be brazed, the use of correct brazing materials and fluxes, and the competency of the operator. Primarily the strength of the joint depends on the area of the braze which unites the surfaces of the parts forming the joint. There is an optimum overlap in a lap shear joint above which the strength will barely increase. This is because the load is carried mainly by the extremities of the lap, the centre carrying little or no load. In practice the optimum length of overlap is 3 to 4 times the thickness of the thinner component of the joint.

4.2 To a lesser extent the joint strength is also governed by the thickness of the braze. Recommended joint clearances are listed in Table 1. The fit of parts to be joined is of vital importance, the gap between the joint faces is critical and must be obtained without strain at room temperature and at the brazing temperature. The liquid filler material is drawn through the gap by capillary attraction and the extent of this action is governed by the gap clearance. Non-uniform joint gaps allow different rates of progress of the advancing liquid metal front through the gap with the subsequent risk of flux or gas entrapment on solidification resulting in low integrity joints. An adequate gap is necessary to allow sufficient flux, when required, to be present to complete its action without chemical exhaustion.

4.3 Joints for assembly by brazing shall be designed so far as possible with the plane of the brazing material in shear. The exception is bronze welding where provision must be made on the joint design for direct application of the brazing filler material. Such joints should not be designed in shear but as fillet or butt joints as for fusion welding.

4.4 So far as possible, joints should be positioned so that the edges of the joint are not subjected to tension or bending.

4.5 Allowance must be made in the design for application of the filler material. Assemblies that are to be brazed by a process requiring the use of a flux should be designed so that flux residues are easily removed and entrapment of flux is avoided. Blind cavities must be vented to prevent gas entrapment.

4.6 Parts should be designed to be self jigging during the brazing process (see Figs. 1h, k and 1).

4.7 In all cases the temperature of operation must be considered. Strength at elevated temperature depends largely upon the type of filler material used. In general terms the silver brazing filler materials, having the lowest melting temperatures, are suitable for continuous service at temperatures up to about 200°C.

4.8 In appropriate cases where fatigue resistance is important, care should be taken to avoid stress concentrations and marked discontinuities in structure stiffness.

4.9 The dependence of brazed joint strength on the wide range of interacting variables, as described in Paras 4.1 to 4.4, makes it unrealistic to define strength values for brazed joints. However the provision of sound joints of optimum strength can be assisted by careful design based on consideration of the factors outlined above. The final design adopted should be checked by a full scale trial embodying strength tests, if so required, on important joints before going into production. The following points will assist in achieving satisfactory results:

(i) Design the joint in such a way that the filler material will flow through the joint to an edge that can be inspected.

(ii) Ensure that minimum assembly stresses are induced prior to brazing.

(iii) Guidance on recommended joint clearances is given in Table 1. Allowances should be made for changes in the joint gap at room temperature and the brazing temperature. This will depend on the method and the materials used for jigging, but for self-jigging applications on parts manufactured from similar materials, there will be no significant change. When brazing dissimilar materials, allowance should be made for the effects of differing thermal expansion coefficients.

(iv) For critical joints establish a technique of manufacture.

(v) The design of joints should be such as to minimise possible corrosion hazards.

(vi) Some recommended joint forms are shown in Fig. 1.

5 INSPECTION OF BRAZED JOINTS

5.1 GRADING OF JOINTS

5.1.1 For inspection purposes, all joints must be graded in accordance with Chapter 400 Para 2 in relation to the grading of parts and will be inspected as required by Chapter 402.

5.2 BASIC INSPECTION TECHNIQUES

5.2.1 The basic inspection techniques include the following:

(i) Visual examination of each joint produced.

(ii) Verification that adequately controlled techniques and operators of acceptable competency are employed.

5.3 SUPPLEMENTARY INSPECTION TECHNIQUES

5.3.1 A number of supplementary inspection processes are available, namely:

(i) Radiographic inspection, including the preparation of detailed techniques for each item and the scrutiny of radiographs. When radiography is used an acceptable defect level should be established for the joint.

(ii) Dye penetrant or magnetic particle inspection as appropriate. It should be noted that dye penetrant or magnetic particle inspection effectively inhibits the use of a reheating procedure to rectify defective joints. This is due to potential capillary contamination by the fluids these methods entail.

(iii) Processes such as ultrasonic and eddy current testing may also be developed for the examination of brazed joints.

(iv) Proof loading - this is a suitable method for items such as ducting.

(v) Mechanical testing, cut-up or NDT of test pieces.

(vi) Pressure or rig testing, as necessary.

6 BRAZING METHODS

6.1 TORCH BRAZING

6.1.1 Heat is applied by means of a hand torch burning a variety of gas mixtures of oxygen with acetylene, hydrogen, propane or natural gas or air with acetylene, natural gas or propane. The process is inexpensive and is suitable for all types of work. The brazing alloy may be hand fed by the operator or pre-placed as an insert. A flux is generally required.

6.2 BRONZE WELDING

6.2.2 Bronze welding is a special form of torch brazing, using a joint design similar to that for welding. It is a useful process for joining low strength mild steel components in complex configurations with minimum distortion. The process is not recommended for use on Grade A components unless it can be shown by test that the fabricated component will meet the appropriate design requirement. Bronze welding is particularly liable to introduce cracking and intergranular penetration in low alloy steels if welding is attempted in the non-stress relieved condition.

6.3 INDUCTION BRAZING

6.3.1 The heat required is developed within the component by placing it in the magnetic field of a coil carrying, a suitable high frequency alternating current. The heating effect is rapid and by careful design of the induction coil the heat may be closely localised to minimise distortion and oxidation and reduction of the properties of fully heat treated materials. A protective atmosphere may be used if the heating operation is effected within a suitable container. When heating is effected in air, a flux is required. The method is particularly suitable for high speed brazing of ferrous materials in large quantities.

6.4 FURNACE BRAZING

6.4.1 heating is effected within a furnace and a suitable protective atmosphere or vacuum may be used to prevent oxidation. When heating is effected in air, a flux is required.

6.4.2 The method is suitable for large batches of small parts which are self-locating or easily jigged or for parts likely to distort through uneven heating. Furnace brazing of aluminium and aluminium alloys is widely used. Furnace brazing using a controlled atmosphere is specially suitable for mild steels and for steels brazed with copper and requiring heat treatment after brazing.

6.4.3 Vacuum brazing is particularly suitable for components requiring clean, contamination free surfaces. It is possible to combine brazing with heat treatment and it is also used for producing high integrity joints in which the brazing alloy is diffused into the parent metals. Materials containing elements which form stable surface oxides can only be fluxless brazed in vacuum or in a strongly reducing atmosphere. Titanium alloys are often vacuum brazed because protective atmospheres, other than argon, result in contamination and embrittlement.

6.4.4 Filler materials containing alloying elements with low boiling points or high vapour pressures should be avoided. The amount of filler material applied to the joint area must be carefully controlled to reduce the likelihood of filler material spreading into other areas.

6.5 SALT BATH BRAZING

6.5.1 Heating is effected within a salt bath. Whenever possible the constituents of the salt bath are chosen so that they have some fluxing action upon the parent and filler materials. The method is particularly suitable for aluminium and certain of its alloys (See Para 7.4). The design needs to be such that removal of the residual salt can be achieved.

6.6 RESISTANCE BRAZING

6.6.1 The heat required is developed by:

(i) resistance at the joint surface, as in resistance welding,

or

(ii) resistance in carbon electrodes which are in direct contact with the area to be brazed and which conduct heat to the joint faces.

7 NOTES ON BRAZING OF PARTICULAR METALLIC MATERIALS

7.1 GENERAL

(i) The following paras summarise the use and the limitations of the more commonly used brazing alloys in relation to the main aerospace structural materials currently employed namely steels, aluminium alloys, copper and heat resistant alloys.[3]

(ii) Full details of the composition and melting range of brazing filler materials are contained in BS 1845. Brazing alloys which are satisfactory for the materials covered in Para 7 are given in Table 2.

7.2 CORROSION RESISTANT STEELS

7.2.1 It is recommended that all corrosion resistant steels be stress relieved after forming and before brazing to reduce the risk of cracking and intergranular penetration.

7.2.2 For brazing austenitic steels where strength and/or oxidation resistance at temperatures above 250°C are required, palladium and gold bearing materials can be used. Both are suitable for vacuum or controlled atmosphere brazing applications.

7.2.3 Furnace brazing in a dry reducing atmosphere using copper brazing material may be used with austenitic corrosion resistant steels.

7.2.4 Butt joints can be made with the silver brazing materials but lap or shear joints are preferred.

7.2.5 For brazing corrosion resistant steels other than the austenitic types, the brazing cycle and material should be compatible with the heat treatment required by the parent metal.

7.3 PLAIN CARBON AND WELDABLE LOW ALLOY STEELS

7.3.1 Silver brazing materials (silver solders) are suitable for joining:

(i) plain carbon steels,

and

(ii) carbon manganese steels.

The normal brazing methods are torch or induction. Butt joints can be made with these silver brazing materials, but lap or shear joints are preferred.

7.3.2 Brazing brasses have limited use for joining:

(i) plain carbon steels, e.g. S21, S510,

(ii) carbon manganese steels, e.g. S92, S514, S515, T64, T65,

(iii) chromium-molybdenum steels, e.g. S142, S534, T53.

The brazing materials used melt in the temperature range 870°C and 920°C. Furnace brazing may reduce cracking and intergranular penetration of the parent metal because of the more uniform heating and cooling and the annealing effect of the process.

7.3.3 Bronze welding (See Para 6.2) is a suitable process for joining parts in mild steel. Bronze welded parts have a similar static strength to oxy-acetylene welded joints, but the fatigue strength of bronze welded joints is appreciably lower than that of fusion welded joints.[2]

7.3.4 Copper brazed joints produced by furnace brazing have limited application (See Para 3.3).

7.4 ALUMINIUM AND ALUMINIUM ALLOYS

7.4.1 Aluminium brazing is suitable for joining:

(i) pure aluminium: L16, L17, L34, L54, L116,

(ii) aluminium-manganese alloys: L59, L60, L61,

(iii) heat treatable aluminium-magnesium-silicon alloys with a maximum magnesium content of 2 per cent: L111, L113, L114.

Note: Aluminium magnesium alloys L44, L56, L80 and L81 may only be brazed if the actual magnesium content is not greater than 2.25 per cent.

7.4.2 Higher strength aluminium alloys suitable for brazing have recently been developed, eg US alloy 7005.

7.4.3 The advantages of aluminium brazing are that joints can often be made in thinner metal than is possible with welding and it often enables a neater joint to be made with less distortion than results from welding. Parts can also be heat treated after brazing if required. Aluminium alloys are frequently fillet brazed where fillet welds are unsatisfactory such as brackets, especially under 16 SWG (0.064in).

7.4.4 The filler materials used for aluminium brazing are aluminium silicon or aluminium-silicon-copper alloys which have melting points slightly lower than that of the metal being joined, to avoid melting of the parent metal during brazing.

7.4.5 In applications of aluminium alloy brazing where the mass of one component is large relative to another, Preheating of the larger component may be essential.

7.4.6 Parts may be brazed by torch, furnace, or for complex assemblies, by salt bath brazing.

7 5 NICKEL BASE ALLOYS

7.5.1 Nickel alloys should normally be brazed in the annealed condition. Thin sections, (eg honeycomb cores), are liable to grain growth at temperatures in excess of 1100°C.

7.5.2 Most of the high nickel alloys can be readily joined using silver rich filler materials, but may be subject to intergranular penetration by the filler material if brazed in a state of stress. When high melting point silver rich brazing materials are used, all stresses may be relieved during the brazing process, but if low melting point silver rich brazing materials are used on heavily worked components, stress cracking could result if the components are not stress relieved prior to brazing. Nickel alloys can be attacked by sulphur compounds and only sulphur free fluxes and gases should be used.

7.5.3 For joining nickel base heat resisting alloys where strength and/or oxidation resistance at temperatures above 250°C are required, three types of brazing alloy are available:

(i) Nickel base brazing materials.

(ii) Palladium bearing brazing materials.

(iii) Gold bearing brazing materials.

Suitable filler materials are given in Table 2. Both palladium bearing and gold bearing materials are suitable for vacuum or controlled atmosphere brazing applications.

7.6 TITANIUM ALLOYS

7.6.1 Processes for the brazing of titanium alloys are available but where such a process is required, specific test work to evaluate the process and the brazing material selected will be necessary.

8 TECHNIQUE OF MANUFACTURE

8.1 In order to achieve satisfactory joints the followings aspects should be considered:

(i) The required surface finish of the joint prior to brazing and the positioning of the filler metal should be defined.

(ii) Jigs and fixtures should be designed to minimise assembly stresses prior to brazing and stresses arising during the brazing cycle. The design should be such that the correct design joint clearances are maintained to ensure satisfactory flow of the filler material.

(iii) Non-uniform heating and cooling, including that which might arise from dissimilarities in mass, should be avoided to ensure freedom from distortion.

(iv) Distortion can also be caused by stress relief of component parts during the brazing cycle.

9 SOLDERING (For other than electrical systems)

9.1 GENERAL

9.1.1 Solders are alloys applied at temperatures below 400°C for joining such metals as steels and copper alloys. Soldering is NOT normally recommended for Grade A joints (See Leaflet 400/1).

9.1.2 Steel parts having a tensile strength of 770 MPa or over should not be soldered or sweated owing to the susceptibility to intercrystalline penetration by the molten solder.

9.1.3 Aluminium and magnesium alloys, zinc base die-casting alloys, high tensile brasses to BS 2872 CZ 115 and BS 1400 and aluminium bronzes are not suitable for soldering.

9.2 Standard grades of solder are covered by BS 219 which also indicates the type of work for which the various grades are suitable. All of these standard grades start to melt at approximately 185°C, but special solders, which have melting points up to approximately 310°C, are available and these are used when retention of strength at elevated temperatures is required.

9.3 Where it is impracticable to remove flux residues, a resin-based soldering flux to DTD 599 should be used. When any other type of flux is used all residues should be removed. Parts to be soldered should be an easy fit, a gap of 0.08 mm (0.00315in) to 0.3 mm (0.0118in) being suitable.

9.4 For soldering oxygen equipment, the requirements of BS N100 are to be observed, especially restrictions therein applying to the use of resinous fluxes.

9.5 There are restrictions on the use of soldered or brazed parts which may come in contact with explosives or pyrotechnic material and guidance should be sought from the Director PERME.

9.6 Where soldered joints are to be used in components for which creep is a design criterion, the following properties may be used as a guide. Specific tests should be conducted to establish requirements for a given design:

(i) Soldered lap joints in copper, brass and mud steel have "short time" shear strengths of 30 to 45 Newtons per sq mm of the lap area, the precise value depending on the material, the solder and the conditions under which the joints are made. However, solders creep at normal temperature so that lower stresses cause failure if applied for long periods. Tests on lap joints gave maximum stresses for a 500 day life ranging from 1.4 to 3.2 Newtons per sq mm of lap area, the lower values being obtained for joints in steel made with antimony-free solders and the higher values for joints in brass. At a life of 500 days, the shear stress-life curves were nearly parallel to the life axis indicating that slightly lower stresses than those quoted could be applied indefinitely without causing failure.

(ii) The strength of soldered joints falls off rapidly with increasing temperature. In some recent tests it was found that lap joints in copper made with a solder to BS 219 had a "short time" shear strength of 38 MPa but when tested at 100°C, the shear strength was only 23 MPa. The creep strength is also reduced; for instance joints in copper which at room temperature would withstand a shear stress of 2.1 MPa for 500 days would only withstand a stress of 0.7 MPa for the same period at 80°C.

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TABLE 1

RECOMMENDED JOINT CLEARANCES
(AT BRAZING TEMPERATURE)
(for guidance only)

Brazing Alloy	Material being joined	Method	Clearance (mm)
Copper	Low C. Steel	Furnace with reducing atmosphere	0.025 to 0.050
	Corrosion resistant steel		0.025 to 0.075
	Nickel base alloy		0.050 to 0.125
Silver alloy BS 1845 Type AG1	Low C. Steel	Torch	0.050 to 0.125
	Corrosion resistant steel	Torch	0.025 to 0.100
	Copper alloys	Torch	0.050 to 0.15
Aluminium/ Silicon BS 1845 AL1	Aluminium alloys	Torch	0.075 to 0.38

TABLE 2
GUIDE TO BRAZING FILLER MATERIALS

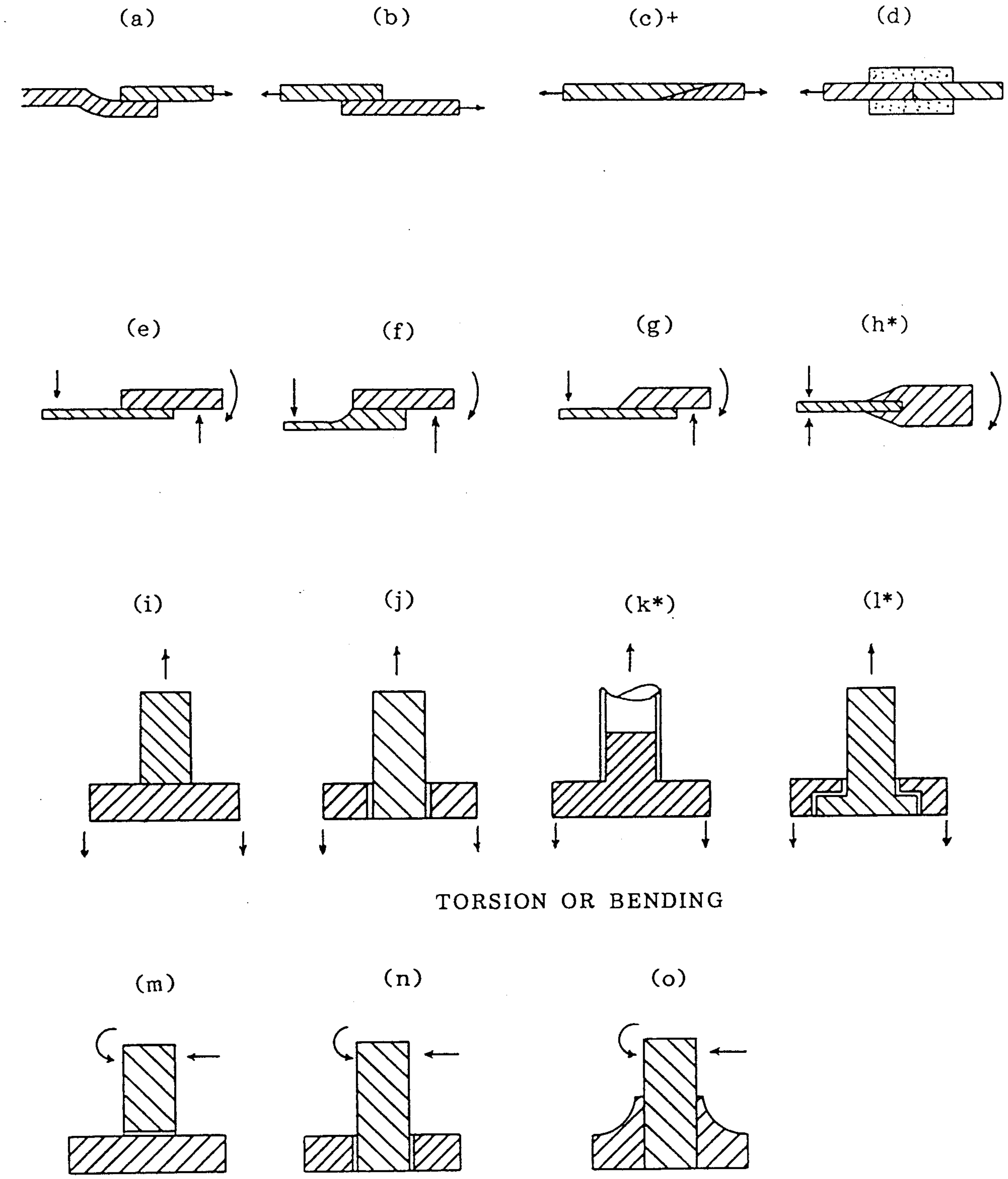
Material	Brazing Filler Material BS 1845 Type	Melting Point or Melting Range °C	Remarks
Corrosion resistant austenitic steels (see para 7.2.5)	Silver Brazing fillers AG 14 AG 15 AG 5 AG 7 V	 630-660 675-735 700-775 780	 Cd free Cd free Cd free Zn and Cd free. Suitable for vacuum or induction brazing. Can be used for service temperature up to 300°C
	Palladium bearing brazing fillers PD 1 PD 14	805-810 1235	Good resistance to oxidation at temperatures up to 500°C. Good resistance to oxidation at temperatures up to 1000°C. Good mechanical strength up to 700° C.
	Gold bearing brazing filler AU 5	 950	 Good strength and resistance to oxidation at temperatures up to 500° C.
Plain carbon and Weldable low alloy steels	Silver brazing fillers AG 14 AG 15	 630-660 675-735	 Cd free Cd free
	Copper-zinc brazing fillers CZ 3 CZ 6 CZ 6A CZ 7 CZ 7A CZ 8	 885-890 875-895 875-895 870-900 870-900 920-980	 Has a higher Zn content to other CZ filler metals listed
	Copper	1083	

TABLE 2 (Continued)

Material	Brazing Filler Material BS 1845 Type	Melting Point or Melting Range °C	Remarks
Aluminium and aluminium alloys	Aluminium brazing fillers AL 1 AL 2	 535-595 565-595	 Low silicon, high copper High silicon, low copper
Nickel base alloys	Silver brazing filler AG 5 AG 7	 700-775 780	 Cd free Cd and Zn free. See para 7.5.2
	Nickel base brazing fillers NK 3 NK 4 NK 5 NK 6 NK 8	 970-1000 980-1040 980-1070 980-1060 1080-1135	
	Palladium bearing brazing fillers PD 1 PD 14	 805-810 1235	
	Gold bearing brazing filler AU 5	950	

FIG 1. JOINT FORMS

NOTE: STRENGTH IMPROVES FROM LEFT TO RIGHT
ARROWS INDICATE DIRECTION OF APPLIED LOAD



+ Joint is difficult to prepare and to align.

* Indicates self jigging

LEAFLET 402/7

PROCESSES AND WORKING OF MATERIALS

SEALANTS AND SEALING

1 INTRODUCTION

1.1 This Leaflet relates to the use of sealants for three main purposes:

(i) Sealing of integral fuel tanks.

(ii) Sealing of pressurised zones such as cockpits and cabins.

(iii) Sealing in high temperature regions such as ductings.

Bearing in mind that these zones have also to meet structural requirements the following aspects should be taken into consideration. In all cases the designer must satisfy himself that the sealant chosen has adequate performance in all the environments to be met during its design life.

2 DEFINITIONS

2.1 SEALANT
Any material which is applied between, or over, joints to prevent passage of a fluid.

2.2 ONE-PART SEALANT
A sealant requiring no mixing of additional components before use. These materials normally attain their final physical form through chemical reaction with moisture vapour in the atmosphere or through the evaporation of contained solvent.

2.3 TWO-PART SEALANT
A sealant which requires the addition of a second component which through chemical reaction causes the material to attain its final physical form.

2.4 BARRIER COATING/BARRIER SEALANT
A material which is applied over a sealant to provide additional resistance to a particular medium/environment.

2.5 OVERCOATING
The application of a further coat of sealant, a second sealant, or a barrier coating over partially or fully cured sealant to provide enhanced environmental resistance.

3 MATERIALS

NOTE: The information below is given as a broad guide only, particularly that relating to working temperature range which may vary considerably within a class depending on the formulation of the sealant and the working environment.

3.1 POLYSULPHIDE SEALANTS

3.1.1 A range of sealants is available with the exact properties of the sealants depending upon the type of curing system used. In general these have a good resistance to fuels and mineral oils and may be expected to perform satisfactorily within the temperature range -50°C to +100°C. However, higher temperature properties may be obtained by selection of the curing system used. These sealants have good weather resistance but may absorb moisture resulting in slight swelling. Certain types can be obtained with insoluble chromates specifically added to inhibit corrosion. Grades of each type are available having differing work lives and rheological properties to assist application. These are suitable for use in fuel tanks, cabin, and dry structure low temperature areas. However, for optimum performance in fuel tanks certain types may require overcoating with a compatible barrier coating.

3.2 FLUOROCARBON SEALANTS

3.2.1 These materials have excellent fuel resistance and weather resistance and may be expected to perform satisfactorily in the temperature range -45°C to +160°C or +200°C for fuel tanks and dry structures respectively. (Warning: Hydrogen Fluoride may be liberated by the sealant when operating at 160°C and above for long periods). Partial precuring and/or overcoating with a barrier sealant may be necessary in fuel zones to overcome leakage caused by shrinkage. A number of grades having different rheological properties is available. A range includes compositions of varying solid content and solvents to assist the control of the open assembly times. Primer systems are essential to obtain maximum adhesion. Special application techniques are available which employ the sealant in the form of semi-cured tape. This reduces the sealant shrinkage on the final assembly application and also ensures that the interfay sealant thickness is consistent. The tapes can be produced in varying widths and thicknesses. They are suitable for use in fuel and ducting systems where heat or chemical resistance is required.

3.3 SILICONE SEALANTS

3.3.1 These materials are not resistant to fuels and oils but have excellent weather and heat/oxidation resistance and may be used within the temperature range -70°C to +250°C continuously, or up to 300°C intermittently. These materials are available as one or two component systems. Single component materials require atmospheric moisture to crosslink and care should be taken to ensure that the bond area and thickness are such that full cure can be obtained throughout the joint. Some of these single component sealants liberate acetic acid vapours during cure and consideration should be given to this potential corrosion risk particularly on copper, copper alloys, magnesium or aluminium alloys in the vicinity, especially if the sealant has been applied to the internal surfaces of a sealed container adjacent to electrical contacts. Whenever possible sealants not liberating acetic acid vapours should be used when in contact with or in proximity to these alloys. For many of these materials primers may be necessary to enhance adhesion. These are used in ductings and fire walls.

WARNING: Particular care should be exercised when silicone polymers are involved as they can act as "release agents" consequently preventing effective adhesive bonding or painting. Contamination from products embodying these materials can occur by both contact and in some cases proximity alone.

3.4 FLUORO-SILICONE SEALANTS

3.4.1 These materials have good resistance to fuels and mineral oils and will operate satisfactorily within the temperature range -60°C to + 175°C but they have poor resistance to some synthetic oils even at temperatures as low as 100°C. They are particularly susceptible to degradation (reversion) when enclosed in a confined space with minimal ventilation.

3.5 NITRILE BARRIER COATINGS

3.5.1 These materials have good resistance to fuels, mineral oil and water. They are low viscosity liquids (20% solids) which can be applied by brushing or fill and drain methods and can be expected to operate satisfactorily within the temperature range -50°C to +100°C. The majority of these materials do not cure but dry by solvent evaporation. Because of this their long term high temperature ageing properties are questionable. For continued satisfactory performance, once wet with fuel they should remain so.

3.6 POLYURETHANE BARRIER COATINGS

3.6.1 These materials have good resistance to fuels and mineral oil. Water resistance is superior to that of nitrile compounds at and slightly above ambient temperatures, but their long term hydrolytic stability at high temperatures is questionable and they may be subject to reversion at any temperature. They can be expected to operate satisfactorily within the range -50°C to 100°C. However, samples of old materials taken from aeroplanes in service have shown that thick sections tend to embrittle. Therefore, care should be taken on application to keep coatings as thin as possible and also to prevent thick sections forming by flow to the lowest point (sumping).

NOTE: There is a potentially very large family of fully elastomeric polyurethane based coatings which do not tend to embrittle and to which the thickness strictures do not apply.

3.7 FLAMMABILITY

3.7.1 All the listed sealants will burn under certain conditions. Some, once ignited, will continue to burn in the presence of sufficient air. Others will only burn if heat is supplied from another source. In many cases the smoke and fumes arising from the burning sealants are asphyxiating and/or toxic to an extent which varies with the nature of the sealant, the temperature of the fire and the degree of combustion. In general, partial combustion arising from lack of air is more likely to produce toxic emissions.

4 SEALING PRACTICE

4.1 GENERAL

4.1.1 It is recommended that to cover anti-corrosion and fluid sealing requirements an interfay sealant should always be used. When sealant is applied as interfay, subsequent assembly and torque tightening of the fasteners should be completed within the work life of the sealant. Where there is sealant shrinkage during cure access should be provided in critical structure positions to allow retightening of the attachments prior to overcoating.

4.1.2 To ensure good sealing and adequate joint behaviour it is necessary that the surface of the joint be properly prepared and primed where required, before application of the sealant. The Design Authority should ensure that the sealants to be used are compatible with any surface treatment or coating on the detail or sub-assemblies. It is recommended that trial assemblies are made and any action taken to correct the fit to ensure that proper mating and sealing can be attained. Where fasteners are inserted during the work-life, adequate sealant should be present to ensure that fasteners are fully wet, assembled in holes etc.

4.1.3 Most types of sealants form a bead from the excess material exuded when the joint is finally assembled. Dependent on the type of interfay sealant used, the bead may be satisfactorily formed into a continuous fillet. In all other cases, the excess bead should be removed and consideration given to the need for the build up of a fillet with additional filleting grade material and whether overcoating is necessary to complete the seal.

4.1.4. In all areas where sealants are used the extent of sealing and the filleting and overcoating requirements should be shown on the appropriate drawing to ensure full inspection is possible. Provision should be made for the internal inspection of the sealant i.e., access panels. Should the design prevent the provision of adequate inspection access panels then access for probe inspection should be provided i.e., removal of bolts or special adaptors with self-sealing caps.

4.2 SEALING OF INTEGRAL FUEL TANKS AND SIMILAR VESSELS

4.2.1 As far as possible highly loaded attachments, with their resulting high local strains, should not be included in the vicinity of the tank/vessel boundary.

4.2.2 Tank edge members should preferably face away from the fuel space to ensure that the edge member skin attachments are external to the fuel. Tank edge members should be free from steps or joggles. Where this is not possible they should be as small as possible.

4.2.3 Where possible the design should allow for the wet assembly of fasteners and where necessary subsequent overcoating.

4.2.4 Consideration should be given to the pitch of fasteners and the extent of overlaps to ensure proper sealing where consideration of the strength requirements alone might otherwise lead to insufficient joint integrity. Blind fasteners should not be used at tank overlaps.

4.2.5 Subsequent to initial tightening, retightening of bolted assemblies should not be carried out unless absolutely necessary.

4.3 AREAS NOT SEALED FOR LIQUID RETENTION

4.3.1 It is often preferable to require interfay and overcoating in these areas also. These areas should be fully sealed to prevent moisture ingress as well as provide the necessary sealing.

5 FORM-IN-PLACE GASKETS AND SEALS

5.1 Where it is required for these form-in-place seals/gaskets to remain attached preferentially to one half of the assembly this may be achieved by the use of a suitable adhesion promoting primer on that part or alternatively a release agent on the other. In some assemblies a combination of both techniques may be required. Where such primers and release agents are used, care should be taken to ensure that they are compatible with the sealant chosen.

5.2 When designing form-in-place gaskets and seals consideration should be given to the effects of compression set and load transmission. Compression set in most sealants will reduce the effectiveness of the seal and load transmission through flexible jointing materials will be poor, i.e., by bolts bending.

NOTE: Where release agents are used consideration should be given to the means of their removal prior to any subsequent finishing operations.

6 PROCESS INSTRUCTIONS

6.1 The process specification detailing the sealing operation should given all details pertaining to surface treatment, primers, mixing and curing instructions and any other information necessary to ensure a properly sealed joint.

LEAFLET 402/9

PROCESSES AND WORKING OF MATERIALS

THE EFFECT OF MACHINING ABUSE ON ALUMINIUM ALLOYS

1 INTRODUCTION

1.1 Most aluminium alloys for airframe construction are heat treatable alloys used in the solution treated and artificially aged condition. The specific heat treatment given is dependent upon the material composition and the desired properties. The artificial ageing treatment is normally carried out at temperatures between 120° and 200° C and may be optimised to give balanced properties for strength, stress or exfoliation corrosion resistance etc.

1.2 Incorrect machining operations can result in the generation of local temperatures of sufficient magnitude to adversely affect the local properties of the material by negating the beneficial effects of the material's prior heat treatment.

1.3 The prime causes of machining abuse are tool wear, lack of coolant, pauses in cutter traverse and inappropriate feeds and speeds.

2 EFFECTS OF MACHINING ABUSE

2.1 Machining abuse takes the local temperature above the ageing temperatures and as the abuse severity increases the material will become progressively overaged, annealed, solution treated and even surface melted, with areas of intermediate effects. Deformation of the surface may occur, and even cracking in very severe cases.

2.2 The strength, fatigue and corrosion resistance may be reduced as the abuse becomes more severe. These local areas have differing metallurgical structures and conductivities and are potential sites for electrolytic corrosion effects, which can be exacerbated by internal residual stresses.

3 DETECTION OF DAMAGE

3.1 Two principle methods for the detection of machining abuse on aluminium are used.

3.2 USE OF ANODISING PROCESS

3.2.1 As part of a corrosion protection scheme most aluminium alloys are subjected to chromic or sulphuric acid anodising after machining. The pre-clean and anodising treatments will reveal areas of machining abuse by a local change in colour or shade of an otherwise uniform anodised surface. The effect is easiest to detect on a flat surface.

3.3 ELECTRICAL CONDUCTIVITY TESTING

3.3.1 Machining abuse changes the metallurgical state of the aluminium alloy and hence its conductivity. A conductivity survey on a machined part can reveal abused areas. In performing a survey, particular attention should be paid to pockets, recesses etc, and any other area there may be reason to suspect.

3.3.2 Local changes in conductivity are more significant than the actual conductivity obtained.

3.3.3 When performing conductivity testing after machining care must be taken not to be misled with “apparent” rather than “real” conductivity changes.

3.3.4 Apparent changes, but in which the material properties are unaffected may be the result of material thickness, probe to edge, hole, or corner distance effects.

4 VULNERABLE AREAS

4.1 Whilst machining abuse can occur anywhere on a component, it is most likely to occur during operations such as machining of deep pockets. In such cases, the material thickness remaining at the bottom of the pocket can be thin and the “heat sink” reduced, thus the formation of “hot spots” can occur with incorrect practice.

4.2 Similarly, incorrect programming for Numerically Controlled (NC) machining can cause the cutter to plunge into the work, travel too quickly, or to dwell too long. This can result in a degradation of the desired material properties.

5 AVOIDING MACHINING ABUSE

5.1 The probability of machining abuse occurring can be reduced by the use of optimum settings for feeds, speeds, and tool cutting angles. Optimum settings and usable tool life can be determined, and NC tapes programmes can be checked. Further, the delivery of adequate quantities of coolant to the tool tip should be ensured.

5.2 The careful inspection, using methods such as those described in Paragraph 3, can establish the absence of machining abuse.

5.3 Regular checking of vulnerable types of components is recommended to verify that good machining practices are established and maintained.

CHAPTER 403

CASTINGS

1 INTRODUCTION

1.1 Castings may be used subject to the compliance with the requirements of this Chapter.

1.2 The Chapter applies to Grade A castings only. There are no special requirements for Grade B castings.

1.3 A radiological technique shall be prepared by an accredited radiologist (see DEF STAN 05-65, Part 1 for definition). This is to be approved by the Aircraft Design Authority and shall be referenced on the casting drawing.

2 RADIOLOGICAL EXAMINATION, MECHANICAL TESTS AND CRACK DETECTION

2.1 All Grade A castings from each source of supply shall be subjected to the radiological examination approved for the part until the suitability of the foundry methods, patterns and alloy have been proved.

2.2 When the production technique has been established the Aircraft Design Authority may authorise a reduction in the radiological examination. Any change in the source of supply, production technique, material or design will necessitate a reversion to 100% radiography as required by Para 2.1.

2.3 In addition, where appropriate, occasional cut-up tests shall be made to check uniformity of the castings and/or the radiological technique.

2.4 If radiological examination should present great difficulty then a cut-up procedure may be acceptable on a suitable sample basis.

2.5 All Grade A castings shall be examined by penetrant flaw detection or magnetic particle flaw detection to a technique approved by the Aircraft Design Authority. Examination shall be after completion of all heat treatment and machining operations.

3 STATIC STRENGTH APPROVAL

3.1 GENERAL

3.1.1 Castings shall normally be approved by test.

3.1.2 Castings may be approved by calculation alone in accordance with Chapter 200 Para 5.1, or by comparison with a casting of similar design and the same conditions of manufacture which has already been tested in the relevant design loading mode and complies with the ultimate strength requirements.

3.2 APPROVAL BY TEST

3.2.1 It shall be shown by test on one or more castings under design loading that the average ultimate strength of the castings tested is not less than the ultimate strengths for the conditions of test, times the factor given in Table 1 of Chapter 200.

3.2.2 Coefficients of variation have been established in the past for a number of specifications and will be found in DEF STAN 00-932. For materials not listed in DEF STAN 00-932, the coefficient of variation shall be established in accordance with the methods described in DEF STAN 00-932 Volume 1 Section 1 Appendix A.

3.2.3 The number of castings tested is at the discretion of the designer who will arrange for random selection from the first 20 castings passed by the founder's inspector. If, owing to limited production in the early stage, 20 castings are not available, a test or tests may be made on a casting or castings from a smaller number; a test or tests from the next 20 passed shall then be made as soon as possible.

3.2.4 Only finished castings shall be used in acceptance tests. Where finishing includes any process of surface restoration or repair which may vary from one casting to another, the castings used for the tests shall represent the weakest type of casting which will subsequently be allowed in production.

3.2.5 When the critical locations and failure modes may be influenced by temperature, the tests should be conducted at the design temperatures. Alternatively, room temperature testing may be acceptable provided that the loading has been suitably factored to allow for the degradation of material properties. The factors to be used may be determined from specimens chosen to represent the behaviour of the critical locations and tested separately at the two temperature conditions.

3.3 APPROVAL BY CALCULATION

3.3.1 Calculations shall be in accordance with the provisions of Chapter 200, using material values allowed in Para 3.4 below.

3.4 ALLOWABLE VALUES OF MATERIAL STRENGTH

3.4.1 The following values may be used subject to the requirements of Chapter 401:

(i) Allowable A values obtained from DEF STAN 00-932 on white paper.

(ii) Allowable A values obtained in accordance with the methods described in DEF STAN 00-932 Volume 1 Section 1 Appendix A. Values obtained in this way shall be associated only with the one specification and the founder supplying the data until data from other founders has been obtained and included.

(iii) Values given in specifications for the acceptance of cut-up tests.

(iv) Values obtained from separately cast test bars called for in the material specifications using the appropriate reduction factors (See Leaflet 403/1 Para 2.3).

4 FATIGUE STRENGTH APPROVAL

4.1 Fatigue strength approval by test and calculation shall be in accordance with Chapter 201.

5 PRODUCTION

5.1 MONITORING

5.1.1 Castings shall be monitored by the founder in accordance with the requirements of the material specification and additionally if required by the designer and stated on the drawing.

5.2 RE-APPROVAL

5.2.1 If the founder, the material specification, the casting process, the casting technique, or the castings,, design is changed in any significant way then the casting shall be re-approved and such repeat tests as may be required by the designer shall be performed.

LEAFLET 403/1

CASTINGS

STATIC STRENGTH APPROVAL OF CASTINGS

1 INTRODUCTION

1.1 This Leaflet explains the procedure for the approval of the static strength of castings.

2 MATERIAL STRENGTH VALUES

2.1 Chapter 401 states requirements which must be complied with when using A, B or S values given in the Metallic Materials Data Handbook (DEF STAN 00-932). The specification may give S values for both cut-up tests and for test bars and these may also be given in DEF STAN 00-932.

2.2 The value given for cut-up tests is a minimum which must be obtained in these tests and may therefore be used directly. From these tests however there should be eventually, sufficient data to provide A and B values and when available these may be used in accordance with the requirements of Chapter 401.

2.3 If the test bar S value is to be used to devise an allowable value for calculation it must be appropriately reduced. In the absence of better data the reduction factors given below should be used.

SAND CASTING		INVESTMENT CASTING	
LIGHT ALLOYS	STEELS	SPECS DTD 666 OR 5072	OTHER STEELS
1.6	1.25	1.15	1.25

2.4 When specification S values only are available and failure in any mode other than tension (eg shear, bending or torsion) may be expected then design ratios f_{so}/ft, b_{10}/t_2 and F_q/ft may be established in accordance with the methods of DEF STAN 00-932.

3 APPROVAL

3.1 CALCULATION

3.1.1 The uncertainties in approval by calculation relate not to the material used, but to the calculation process and, therefore, no special provisions are necessary for castings.

3.1.2 Allowable values for sand castings in aluminium alloys and for steels other than DTD 666 and 5072 may be used for investment castings in the same materials unless the designer prefers to obtain his own by the methods of DEF STAN 00-932. The first two methods of para 5.1 below are acceptable to provide data for the calculation of allowable values but the third is not. Of test bar data it has been found that either the mean of the standard deviation or both often do not represent the casting properties accurately enough for this purpose.

3.2 TEST

3.2.1 The test factor for castings may be obtained from Chapter 200.

3. 3 COMPARISON

3.3.1 Approval by comparison with a casting of similar design, which has previously been approved by test, involves production under the same conditions of manufacture. This may only be achieved by using the same founder.

4 GRADE B CASTINGS

4.1 There are no mandatory requirements for the static strength approval of Grade B castings.

5 MONITORING

5.1 Monitoring of production castings by the founder is commonly done by one of 3 methods:

Method (i) tests on standard test pieces cut from castings,

Method (ii) tests on standard test pieces taken from an attachment to the casting which are processed with the casting throughout,

Method (iii) tests on standard test bars from the same melt as the castings.

The object of this monitoring is to ensure that the material strength properties of the castings are maintained in production.

5.2 Methods (i) and (ii) of para 5.1 above are the preferred methods for monitoring by the designer as required by Chapter 401 when clearing a casting by calculation.

CHAPTER 404

MARKING OF ROTORCRAFT PARTS

1 INTRODUCTION

1.1 This Chapter defines requirements for the marking of rotorcraft parts to make them easily identifiable.

1.2 The requirements of this Chapter shall apply to all rotorcraft parts.

1.3 The design authority shall be responsible for the selection of the types of marking to be used and the method(s) by which such markings are applied to a particular part.

1.4 In addition to the markings required by this Chapter, the contractor shall apply such permanent and/or temporary markings as may be called for by the Aeroplane Specification.

2 DEFINITIONS

2.1 Part: A general term describing any or all of the items which go to make up equipments, general arrangements, assemblies or sub-assemblies, or any combination thereof.

2.2 Assembly: A number of parts or sub-assemblies or any combination thereof, joined together to perform a specific function. The assembly may consist of either:

(i) parts which do not have their elements welded, soldered, riveted or otherwise connected together in a permanent manner and which will not be stocked (though their elements may be), or

(ii) parts which have been fabricated and have their elements permanently connected together. The parts or their elements or both may be held in stock (see DEF STAN 05-10).

2.3 Components: A part or any combination of parts, sub-assemblies and assemblies mounted together, normally capable of independent operation in a variety of situations, and includes those assemblies that are regarded as complete units for storage.

2.4 Identification Marking (see also DEF STAN 05-123 Chapter 311). Markings applied to a part or its package for the purpose of engineering, manufacturing traceability or inspection control, eg:

(i) Nato Stock Number.

(ii) Part number and issue number.

(iii) Material batch code.

(iv) Serial number.

(v) Foundry or forge marks.

(vi) Inspection marks.

(vii) Radiological examination marks.

(viii) Drawing numbers.

(ix) Any other special markings.

2.5 Permanent Marking: Markings which will ensure identification of the part during its normal service life.

2.6 Temporary Marking: Markings which ensure identification of the part during handling, manufacture and storage prior to assembly if practical considerations preclude permanent marking at those stages. Under certain conditions these markings may exhibit the characteristics of permanent markings.

3 GENERAL

3.1 No method of marking shall be used in such a manner or in such a place that it would reduce the strength or the life or affect the performance of the part or the rotorcraft in any other way.

3.2 The method of marking adopted shall not increase the risk of corrosion. In particular, where a plate made from a different material to the component is affixed to it for marking purposes, precautions shall be taken so that no risk of corrosion is introduced.

3.3 Details of identification markings used, the methods by which they are applied and their location shall be stated on the drawing of the part (see DEF STAN 05-10).

3.4 The number of markings on any one part shall be kept to a minimum.

3.5 Parts shall be marked so that they can be easily identified for maintenance purposes when assembled on the rotorcraft.

3.6 Identification symbols which can be mistaken as representing a direction of movement or flow shall not be used.

4 MARKING

4.1 PARTS

4.1.1 Each part shall be marked with:

(i) the detail drawing number for the part or a code number for the part or a code number corresponding thereto; the latter being recommended in all cases where the drawing number consists of more than 5 characters,

(ii) the 13-digit NATO Stock Number (NSN), wherever possible prefixed by the relevant Service Domestic Management Code (DMC). If the space available is inadequate for the full NSN to be used, then the last 7 digits may be shown.

Note: Items which are too small or fragile to carry the required marking, will be identified by the markings required to be put on the packaging by DEF STAN 81-41 Part 6.

4.1.2 Where two or more parts are identical and therefore strictly interchangeable but are used on different types of rotorcraft, the same numbers shall be used. For parts which differ from each other as regards 'hand' (though otherwise identical), different numbers shall be allotted.

4.1.3 The contractor may impose such private markings as necessary to facilitate the assembly of parts, provided that they are not located in a position where they will confuse other identification marks.

4.1.4 Non-standard hexagon headed bolts having a feature significantly different from the basic standard but which is visually indistinguishable, shall have a raised disc on the head in addition to the part number.

4.1.5 The following parts are, however, (subject to Para 1.4) excluded from all forms of marking:

(i) hardened steel springs,

(ii) standard bolts and screws of any diameter not greater than 5.6 mm (7/32 in), nuts of any diameter not greater than 9.5 mm (3/8 in), ordinary and spring washers of any diameter,

(iii) split pins, taper pins and parts similar in respect of size.

4.2 ASSEMBLIES

4.2.1 Elements of assembled parts referred to in para 2.2 (i) shall be marked in accordance with the requirements of para 4.1.

4.2.2 The fabricated parts referred to in para 2.2(ii) shall be marked with the number of the assembly drawing. The elements shall be marked in accordance with the requirements of para 4.1 if these elements are stocked separately.

4.3 COMPONENTS

4.3.1 Components or complete units including those assemblies that are regarded as complete units for service storage shall be marked with the following:

(i) The name and mark or type number of the item.

(ii) The 13 digit NATO Stock Number (NSN) and wherever possible prefixed by the Service Domestic Management Code (DMC).

(iii) The manufacturer's name, or NATO manufacturer's 5-character code (if allotted).

(iv) The serial number (where required). Contractors' symbols shall precede all serial numbers. The symbol/serial number combination for any part shall not have been used previously for any other part or component on the same rotorcraft.

(v) Any additional identification marking required by the contract or particular specification.

4.4 AIRFRAME COMPONENTS

4.4.1 Components such as fuselages, mainplanes, control surfaces, alighting gears etc., shall bear a serial number plate and a modification record plate. Where the fuselage, wing etc., is built in sections to comply with the requirement of Chapter 801 similar plates shall be fixed to each section. The method of fixing the plate shall not increase the risk of corrosion, stress corrosion or fatigue.

4.4.2 The serial number and modification record plates shall be manufactured of approved corrosion-resistant material. Their dimensions and inscriptions shall be in accordance with Figs.1 and 2 where practical and the plates when completed may be protected with clear varnish as necessary.

4.4.3 The serial number plate shall contain the NATO Stock Number and wherever possible prefixed by the Service Domestic Management Code, the contractor's symbol/serial number for the component, drawing number and issue number/letter to which manufactured and the stamp of the approved member of the contractor's quality/inspection department accepting the component. It shall be permanently fixed to the main structure of the component to which it relates and so positioned that it is possible to read the details on the plate without resort to dismantling. It is permissible to provide special removable panels if necessary for examination and such panels need not be of the quickly detachable type.

4.4.4 The modification record plate shall bear the NATO Stock Number and the contractor's symbol/serial number of the component to which it relates and shall be located adjacent to its companion serial number plate but may be detachable for the endorsement of modification numbers provided that, when fitted, it is positively locked in position. When it is not detachable it shall be mounted in a position where stamping of numbers on the plate will not damage the surrounding structure. (See DEF STAN 05-123 for details of the modification recording procedure).

4.5 OIL TANKS

4.5.1 All oil tanks whether they are covered or otherwise shall have the NATO Stock Number, the serial number, drawing number, issue. number and inspector's stamp applied in white colouring by stencil. Such markings shall be reproduced in a similar manner on the outside of the self-sealing or crash-proof cover, where fitted, in a position adjacent to the inspection access panel.

4.6 FUEL TANKS

4.6.1 The marking of fuel tanks shall be in accordance with the requirements of Specification No. DTD 1101.

4.7 REPAIRABLE PARTS

4.7.1 During system development and procurement phases the MOD, in conjunction with the manufacturer and in accordance with the procedures of DEF STAN 05-123, will identify those items perceived to require regular repair in Service or Industry. During initial manufacture the manufacturer shall allocate to each item a 'Repairable Component Tracking Serial Number' (but see para 4.7.2 below) to enable them to be tracked during Repair and throughout their Service life. This Repairable Component Tracking Serial Number shall be confined to 13 alpha-numeric characters in accordance with AECMA Specification 2000M 'International Specification for Materiel Management - Integrated Data Processing for Military Equipment: Appendix 1'.

4.7.2 Where the contractor has allocated a serial number already, for identification or other purposes, this serial number will be utilised by MOD for repair control purposes, and there will be no need to allocate an additional Repairable Component Tracking Serial Number.

4.8 RECONDITIONED PARTS

4.8.1 Components which have been reconditioned shall have a new serial number plate affixed either alongside the existing one or, if this is not practicable, on top of it. The new plate shall show the original markings with the original serial number prefixed by the reconditioning contractor's symbol followed by the letter 'R'. In no circumstances are the old markings to be removed. Where used the Repairable Component Tracking Serial Number is to be retained unaltered.

5 TYRES

5.1 PROTOTYPE TYRES

5.1.1 Each new type of tyre shall be defined by a drawing prepared by the tyre firm and prototype tyres will be known by this drawing number. The tyre drawing will give the dimensions of the tyre and wheel rim, its load deflection characteristics over the permissible range of inflation pressures, and particulars of the tyre construction in general terms.

5.2 PRODUCTION TYRES

5.2.1 For production tyres, the stores reference number shall be added to the drawing. When a tyre design is altered in such a way as to affect its application, the drawing number and stores reference shall be changed.

5.2.2 Tyres intended for the same application will bear the same stores reference number.

5.3 SIZE OF MARKINGS

5.3.1 The markings shall be in accordance with the requirements of Specification No. DTD 1097.

6 TRANSPARENT COMPONENTS

6.1 All markings shall (as signified on the drawing) be in a readily visible position when installed in the rotorcraft, and the lettering shall be large enough for all serial and part numbers to be easily readable.

6.1.1 A preferable standard serial number code would include; the manufacturer's code number, manufacturer's plant number, the date (e.g., 01 June 83), followed by a serial number taken from a series of numbers which start from one each day.

6.2 AS-CAST ACRYLIC SHEET

6.2.1 Transparent components such as windows, windscreens, panels and canopies made from as-cast acrylic sheet to DTD 5592, shall be marked by one of the following methods:

(i) A marked metal label attached to the metal frame.

(ii) An engraved acrylic label attached with an approved cement to the fibre-glass or terylene reinforced edge member.

(iii) Written in a suitable marking ink direct on to the fibre-glass or terylene reinforced edge member, and covered with a thin layer of an approved acrylic cement.

(iv) Written direct on the transparency itself with a suitable marking ink PRIOR TO ANNEALING.

6.3 STRETCHED OR PRESSED ACRYLIC SHEET

6.3.1 Transparent components made from stretched or pressed acrylic sheet shall be marked by one of the following methods:

(i) Any of the methods mentioned in para 6.2 provided subsequent annealing as mentioned in para 6.2.1 (iv) is not used.

(ii) By abrasive blasting through a stencil using 400 mesh white alumina. The depth of marking shall not exceed 0.005 mm.

6.4 GLASS

6.4.1 Transparent components made from glass shall be marked by one of the following methods:

(i) Assemblies may have a small metal part number or serial number label laminated within the panel and near the panel edge, or in the case of an air space design, the part number or serial number label may be attached within the air space with an approved silicone or polysulphide adhesive.

(ii) Single piece thermally toughened glass components may be marked by an approved abrasive blasting process through a suitable stencil.

or alternatively:

(iii) Thermally toughened glass may be marked with a ceramic paste subsequently fired during toughening.

(iv) Chemically toughened glass shall be marked using an air drying ceramic paste or paint.

7 LOCATION OF IDENTIFICATION MARKINGS

7.1 Markings shall not be placed in the following positions:

(i) On bearing surfaces or areas subject to wear.

(ii) On mating surfaces or where bolts etc., may obscure the markings.

(iii) On areas subsequently removed or obscured by mechanical or chemical processing, except for temporary marking.

(iv) In areas where there may be a stress concentration, such as near holes, on fillets, bends, radii, lug faces or edges of parts.

Note: It is permissible to mark stressed areas using the Marking Ink Method.

8 METHODS OF IDENTIFICATION MARKING

8.1 Leaflet 404/1 provides details of recommended methods of identification marking and shall be taken into consideration when selecting methods for the marking of rotorcraft parts.

8.2 PHYSICAL IDENTIFICATION METHODS (e.g., metal die stamping, engraving, vibro-percussion engraving)

8.2.1 Physical identification methods shall not be used as follows:

(i) On steels of maximum specified tensile strength exceeding 1450 MPa.

(ii) In regions of high stress or where fatigue may be critical. (However, physical identification may be used at the designers discretion if all the safeguards contained in this Chapter and Leaflet 404/1 are followed).

(iii) On areas of high grade surface finish.

(iv) On surface hardened or nitrided areas.

(v) On materials thinner than 0.9 mm.

(vi) On welded containers.

(vii) On any surface that has been painted, plated, anodised or conversion coated.

(viii) On hydraulic pipes or on the pressurised walls of ducts.

8.2.2 Sharp cornered borders, sharp angles and cuts shall be avoided and the depth of marking shall be kept to a minimum consistent with legibility.

8.3 MARKING INK

8.3.1 Marking inks, paints and lacquers shall be compatible with the applied paint scheme or the base material or protective treatment, if not painted. The colour of the marking shall be a distinct contrast with the background. For titanium alloys, inks shall be halogen free.

8.4 LABELS

8.4.1 Where transparent labels are used, the paint and adhesive shall be compatible with the part and label.

8.4.2 The method to be used to attach metal labels to the part shall be specified on the drawing. The label shall not foul moving parts or cause damage if it becomes detached.

8.5 Identification by marking ink, transparent plastic labels or metal labels and self-adhesive labels shall not be used to identify parts that will be in contact with powerful oxidants such as hydrogen peroxide, liquid oxygen.

8.6 ELECTROCHEMICAL MARKING

8.6.1 Electrochemical Marking shall not be used on:

(i) anodised aluminium parts,

(ii) magnesium parts,

(iii) non-metallic parts,

(iv) steels of maximum tensile strength exceeding 1550 MPa.

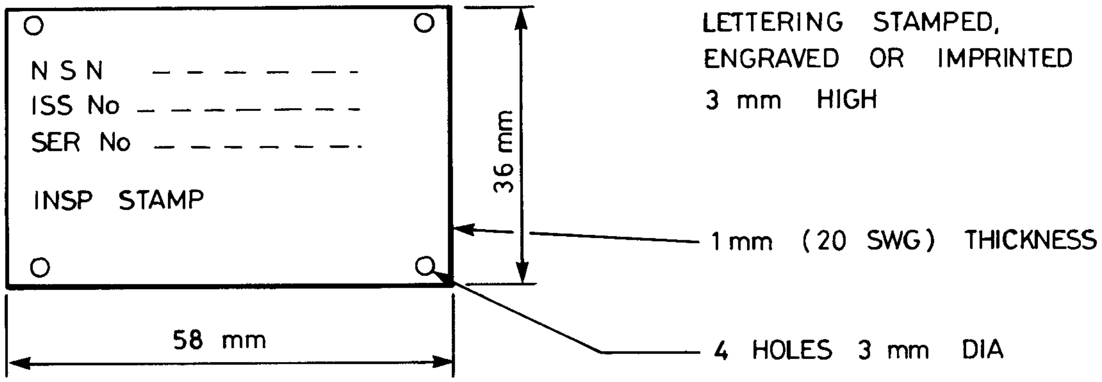


FIG 1 SERIAL PLATE

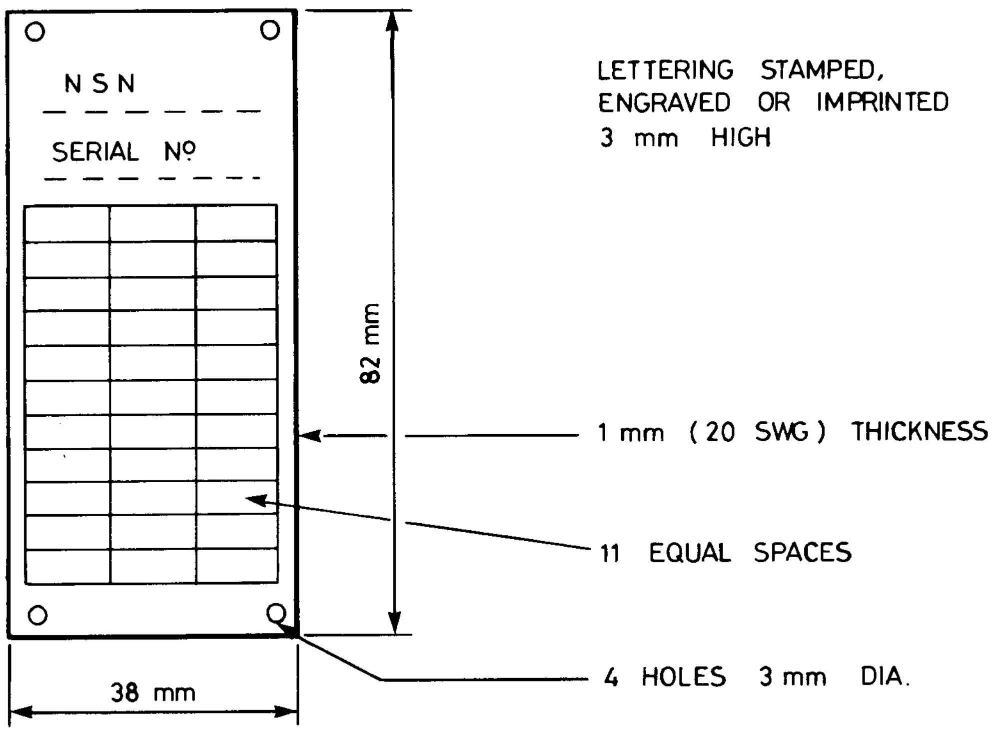


FIG 2 MODIFICATION PLATE

LEAFLET 404/1

MARKING OF ROTORCRAFT PARTS

METHODS FOR THE IDENTIFICATION MARKING OF ROTORCRAFT PARTS

1 INTRODUCTION

1.1 This Leaflet provides details of the methods recommended for the identification marking of rotorcraft parts. Tables 1, 2, 3, 4 and 5 provide a guide for the selection of identification marking methods for various categories of rotorcraft parts.

2 IDENTIFICATION MARKING METHODS (see Chapter 404 para 8)

2.1 METHOD 1: HAND, ROLLED OR MACHINE METAL DIE STAMPING

2.1.1 This method affects the structure of the metal in areas of marking and may reduce fatigue strength of the part. Cold working and notching can cause stress concentrations. When there is a possibility of damage or distortion during stamping, the part should be adequately supported and the minimum force should be applied consistent with the production of legible marks.

2.1.2 This method is not recommended for Grade A parts, as defined in Chapter 400, subject to fatigue conditions.

2.2 METHOD 2: MACHINE ENGRAVING

2.2.1 This method may reduce the fatigue strength of the part.

2.3 METHOD 3: VIBROPERCUSSION ENGRAVING

2.3.1 This method may reduce the fatigue strength of the part. This reduction can be minimised by strictly controlling the marking tool bit radius and the stroke length or pressure. A bit radius of not less than 0.5 mm is recommended.

2.4 METHOD 4: PACK AND LABEL

2.4.1 The part should be contained in a suitable bag or container to which is attached a tag bearing all the required identification marks.

2.5 METHOD 5: EMBOSSED, MOULDED OR IMPRESSED FROM MOULD

2.5.1 Raised marks should be used wherever possible.

2.5.2 When raised marks cannot be used, depressed may be specified by the drawing but the marks should be fully rounded with no sharp angles, corners or border edges.

2.6 METHOD 6: MARKING INK (INCLUDING PAINT AND STENCIL)

2.6.1 Markings by this method are not of a durable nature and due precautions should be taken against their accidental removal or obscuration This method may be used for permanent marking, but a more permanent method is preferred whenever possible.

The markings may be applied:

(i) on parts not requiring protective treatment,

(ii) after protective treatment,

(iii) after painting, except for bleed-through inks which may also be applied prior to painting for certain types of paint schemes.

2.6.2 INK MARKING: The ink can be applied either directly onto the part or alternatively onto a patch of paint of a contrasting colour applied to the part. The marking should be protected by a clear lacquer, or resistant self-adhesive tape.

2.6.3 BLEED-THROUGH INK MARKING: The mark can be applied to the part prior to the application of certain paint schemes.

2.6.4 PAINT WITH STENCIL: Paint markings are applied through a stencil.

2.7 METHOD 7: TRANSPARENT PLASTIC LABEL

2.7.1 Transparent parts can be identified by cementing to them a transparent plastic label which has been engraved and paint filled on its contact surface (see Chapter 404 para 6.2.1).

2.8 METHOD 8: METAL LABEL AND SELF-ADHESIVE LABEL

2.8.1 This method is primarily intended for temporary marking, but may be used for marking more categories than shown in Table 1, at the designer's discretion. Self-adhesive labels may also have permanent applications.

2.8.2 The label may be protected by a compatible clear lacquer or transparent coating.

2.9 METHOD 9: ELECTROCHEMICAL MARKING

2.9.1 For limitations on the use of this method, see Chapter 404 para 8.6.

TABLE 1

GUIDE TO THE SELECTION OF METHODS OF IDENTIFICATION MARKING

CORROSION RESISTING AND NON-CORROSION RESISTING STEELS

FORM OF MATERIAL	THICKNESS	GRADE[1]	METHOD OF MARKING	REMARKS
Forgings, Castings	Below 0.9 mm	A & B	5, 6, 9[2]	
	0.9 mm and thicker	A & B	1, 3, 5, 9[2]	
Machined from bar	Below 0.9 mm	A & B	6, 9[2]	
	0.9 mm and thicker	A & B	1, 3, 9[2]	
Surface Hardened, Nickel or Chromium Plated		A & B	2, 3, 9[2] before treatment or on untreated areas	Marking prior to treatment is preferred
Sheet	Below 0.9 mm	A & B	6, 9[2]	
	0.9 mm and thicker	A & B	1, 3, 6, 9[2]	
Tubes	Below 0.9 mm	A	6, 9[2]	Self-adhesive labels (Method 8) may be used and should normally be covered with a compatible clear lacquer
		B	3, 6, 9[2]	
	0.9 mm and thicker	A & B	6, 9[2]	
Gas/Liquid Pipes		A & B	6, 9[2]	Self-adhesive labels (Method 8) may be used and should normally be covered with a compatible clear lacquer.

Notes: 1 Parts are graded in accordance with Chapter 400 para 2.

2 See Chapter 404 para 8.6 for limitations on the use of Method 9.

TABLE 2

GUIDE TO THE SELECTION OF METHODS OF IDENTIFICATION MARKING

ALUMINIUM AND ALUMINIUM ALLOYS

FORM OF MATERIAL	THICKNESS	GRADE	METHOD OF MARKING	REMARKS
Forgings, Castings	Below 0.9 mm	A & B	5, 6, 9	
	0.9 mm and thicker	A & B	1, 3, 5, 9	
Machined from bar	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A & B	1, 3, 9	
Sheet	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A & B	1, 3, 6, 9	
Structural and Control Tubes	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A & B	3, 6, 9	
Gas/Liquid Pipes		A & B	6, 9	

MAGNESIUM AND MAGNESIUM ALLOYS

May be marked using similar methods to those recommended for aluminium alloys except that Method 9 must not be specified.

TABLE 3

GUIDE TO THE SELECTION OF METHODS OF IDENTIFICATION MARKING

TITANIUM AND TITANIUM ALLOYS

FORM OF MATERIAL	THICKNESS	GRADE	METHOD OF MARKING	REMARKS
Forgings, Castings	Below 0.9 mm	A	5, 6, 9	
		B	5, 6, 9	
	0.9 mm and thicker	A	2, 5, 9	
Machined from bar	Below 0.9 mm	A	6, 9	
		B	6, 9	
	0.9 mm and thicker	A & B	2, 9	
Sheet	Below 0.9 mm	A	6, 9	
		B	6, 9	
	0.9 mm and thicker	A & B	6, 9	
Tubes and pipes	Below 0.9 mm	A	6, 9	
		B	6, 9	
	0.9 mm and thicker	A & B	6, 9	

TABLE 4

GUIDE TO THE SELECTION OF METHODS OF IDENTIFICATION MARKING

COPPER AND COPPER ALLOYS

FORM OF MATERIAL	THICKNESS	GRADE	METHOD OF MARKING	REMARKS
Forgings, Castings	Below 0.9 mm	A & B	5, 6, 9	
	0.9 mm and thicker	A & B	1, 3, 5, 9	
Machined from bar	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A & B	1, 3, 9	
Sheet	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A & B	1, 3, 9	
Tubes and pipes	Below 0.9 mm	A & B	6, 9	
	0.9 mm and thicker	A	6, 9	
		B	3, 6, 9	
Oilite		A & B	2, 3, 4	

TABLE 5

GUIDE TO THE SELECTION OF METHODS OF IDENTIFICATION MARKING

NON-METALLICS

MATERIAL	METHOD OF MARKING	REMARKS
Transparent Materials (e.g. Glass, acrylic sheet)	See Chapter 404 para 6	
Laminated Plastics	4, 6	
Moulded Plastics	4, 5, 6, 7	
Wood, Leather, Felt, Fabric, Canvas, Cork, Fibre, Glass Fibre	4, 6	Method 6 is not suitable for felt
Rubber	4, 5, 6	

CHAPTER 405

EXFOLIATION CORROSION OF ALUMINIUM ALLOYS

1 The aim shall be that all aluminium alloy parts of the rotorcraft are so designed that when the rotorcraft is maintained according to the servicing schedule, there shall be no unacceptable loss of airworthiness as a result of exfoliation corrosion.

2 Necessary precautions shall be taken in the choice of alloy, heat treatment, degree of working, grain structure, manufacturing methods, protective treatments, etc, to minimize the chance of exfoliation corrosion occurring. Leaflet 405/1 gives advice on means of complying with this para.

3 Wrought aluminium alloys categorised D in Table I of Leaflet 405/1, because of their high susceptibility to exfoliation corrosion, shall not be used without the prior approval of the Rotorcraft Project Director. In the case of aluminium alloys not included in the Table, application shall be made to the Rotorcraft Project Director for Categorisation.

4 Aluminium alloys categorised C in Table 1 of Leaflet 405/1, which have occasionally suffered exfoliation corrosion in service, shall be used only after consultation with materials specialists. Advice on testing procedures to establish the susceptibility of aluminium alloy components is given in Leaflet 405/1.

LEAFLET 405/1

EXFOLIATION CORROSION OF ALUMINIUM ALLOYS PRECAUTIONS TO BE TAKEN TO AVOID EXFOLIATION CORROSION OF ALUMINIUM ALLOYS

1 INTRODUCTION

1.1 In certain heat treated forms, aluminium-copper-magnesium (-silicon) and aluminium-zinc-magnesium (-copper) alloys are susceptible to layer or exfoliation corrosion. This form of corrosion usually initiates on the surface of the alloy, often in areas where end grain is exposed (such as fastener holes) and especially where end grain is exposed by machining, such as at changes of section. After an induction period, delamination of the alloys commences; the corrosive attack is usually intergranular and follows an anodic path parallel to the original surface of the alloy. Once initiated, the rate of attack often increases markedly. Thus major exfoliation damage can appear in an aircraft structure after several years during which time corrosion had not been detected at the usual inspection intervals. The first signs of corrosion may be local lifting or blistering of the paint scheme, or in the form of pits with the layer corrosion spreading laterally from the base of the pits. Exfoliation corrosion can severely reduce the load-carrying capacity of a structure.

2 FACTORS AFFECTING THE SUSCEPTIBILITY OF ALUMINIUM ALLOYS TO EXFOLIATION CORROSION

2.1 Susceptibility to exfoliation corrosion can be markedly affected by the choice of heat treatment and working of the alloy. In general, susceptibility can be increased by slow quench rates an increase in the directionality of the grain structure, and underageing heat treatments. For example, the Al-Cu-Mg alloy 2024 is susceptible to exfoliation when naturally aged, it is most susceptible when rolled into thick sheet or thin plate of from 4 to 20 mm section, and susceptibility increases with heating at 100°C to 150°C for times too short to achieve peak hardness. The behaviour of Al-Zn-Mg-Cu alloys is similar except that they are very susceptible when aged to peak hardness, and can be very susceptible in thin sheet form.

3 ALUMINIUM ALLOYS RESISTANT TO EXFOLIATION CORROSION

3.1 Precipitation heat treatments to peak strength after cold working (T8 tempers) are suitable for making most Al-Cu-Mg (-Si) alloys resistant to exfoliation; other 2000 series alloys, such as 2618, are fairly resistant even in the T6 temper, while slight overageing of 2014 alloys is beneficial when done at the upper end of the allowed heat treatment range (180°C). Overageing of Al-Zn-Mg-Cu alloys can render them immune to exfoliation corrosion. In the T76 temper, the alloys are resistant to exfoliation corrosion while in the T73 condition they should be immune.

4 TESTS FOR EXFOLIATION RESISTANCE OF ALUMINIUM ALLOYS

4.1 When it is necessary to demonstrate the resistance of an alloy or component to exfoliation corrosion then the unprotected component should be submitted to an accelerated exfoliation test. The intermittent acidified salt fog test[1] and the SWAAT test[2] both use the salt fog cabinet procedures which are recommended for assessing the susceptibility to exfoliation corrosion of most types of aluminium alloys included Al-Cu-Mg (-Si) and Al-Zn-Mg-Cu alloys, and useful guide for assessing test results is given in the ASTM Standard[3]

4.2 It may not be possible to perform an accelerated test on a component to assess the exfoliation resistance of all of the surfaces. In such cases a suitable portion (or portions) of the component should be exposed to the test environment, and the portion (or portions) should be selected so that the surfaces exposed are those in which the grain flow is most favourable for exfoliation to occur.

REFERENCES

No	Author(s)	Title, etc
1	B W Lifka D O Sprowls	An improved exfoliation test for aluminium alloys. Corrosion (NACE) 22 pp 7-15 (1966).
2	H B Romans	An accelerated laboratory test to determine the exfoliation corrosion resistance of aluminium alloys. Materials Research and Standards pp 31-34, p68 (November, 1969).
3	-	ASTM standard method of test for exfoliation corrosion susceptibility in 7XXX series copper containing aluminium alloys. ASTM Standard Method of Test, G34-72, pl (1973).

TABLE 1

RELATIVE SUSCEPTIBILITIES TO EXFOLIATION CORROSION OF VARIOUS WROUGHT ALUMINIUM ALLOYS

Alloy Type and Temper		Sheet ⩾3.2mm section / Sheet < 3.2 mm			Plate		Tube		Extruded Bar and Section		Forgings	
2014	T3,T4	BS L156) BS L158) BS L163) BS L164)	B/C	D			BS L105	D	BS L102	D	BS L103	D
	T6	BS L157) BS L159) BS L165) BS L167)	B/C	B/C	DTD 5040 BS L93	C C	BS 3L63	C	BS L168 BS 3L87	C C	BS 2L77	B/C
2618	T6	DTD 5070	B	B	2618-T651	B			DTD 5014 2618-T6, T62	B/C B/C	DTD 717 DTD 731 DTD 745	C B B
	T71										DTD 5084	B
2219	T6						2219-T62	B/C	2219-T62	B/C		
	T8				2219-T851	B	2219-T851	B/C	2219-T8510	B/C		
2024 (2124)	T3,T4	BS L109 BS L110	B/C B/C	D D	DTD 5100 BS 2L97	D D						
	T8			0	2024-T851	B						
6082	0	DTD 346	A/B	A/B								
	T6	BS L113	A	A	BS L115	A/B	BS L114	A/B	BS L111	A/B	BS L112	A
6061	T6						BS L117 BS L118	A A				
7010	T76				DTD 5120	B						
	T736				DTD 5130	A/B					DTD M239	B
7050	T76				7050-T7651	B						
	T736				7050-T73651	A/B					7050-T736	B
7075 (7175) (7475)	T6	BS 2L88	D	D	DTD 5110 BS 2L95	D D			DTD 5074 DTD 5124	D D		
	T73				7075-T7351	A			BS L160	A	BS L161 BS L162	A A
7XXX	T6								DTD 5114	D	DTD 5024	D
	T7										DTD 5104	C

NOTE 1: The susceptibility of 2000 series (Aluminium-copper) alloys are markedly affected by grain structure, which is dependent on the amount of working and the quench rate experienced by the alloy. In general, the thinner section materials will be categorized B and susceptibility will increase with section to category C: this is indicated by B/C ratings.

NOTE 2: The susceptibility of forgings is very dependent on the degree of working. Those categorised D may occasionally be used quite safely and the use of one of the test procedures given in para 4 is recommended to assist the Aeroplane Project Director in assessing the case for using the alloy.

CLASSIFICATION OF SUSCEPTIBILITY

A Immune to exfoliation corrosion.

B Resistant to exfoliation corrosion, although mild exfoliation may be induced under the most extreme conditions.

C Susceptibility to exfoliation corrosion. Under some conditions exfoliation can occur in service. Alloys in this category should only be used after discussion with materials specialists.

D Very susceptible to exfoliation corrosion. These alloys shall not be used without the prior approval of the Aeroplane Project Director (see Chapter 405, para 3).

CHAPTER 406

STRESS CORROSION CRACKING

1 INTRODUCTION

1.1 The aim shall be that all parts of the rotorcraft are so designed, protected, assembled, drained and vented that when it is maintained in accordance with the Servicing Schedule, there will be no unacceptable loss of airworthiness as a result of spontaneous, progressive or delayed cracking induced by stress corrosion and/or associated hydrogen embrittlement.

2 GENERAL REQUIREMENTS

2.1 The designer shall take the necessary precautions by controlling such aspects as the strength of material, heat treatment, grain direction, stress concentrators, manufacturing methods, surface condition, residual and assembly stress, and protective treatment to minimise the chance of failure due to such cracking. (Leaflet 406/1 gives advice on acceptable means of compliance with this para).

2.2 Because of their high susceptibility to stress corrosion cracking, those materials (aluminium alloys and steels) categorized D in Table 1, 2 or 3 of Leaflet 406/1 shall not be used without the approval of the Rotorcraft Project Director. When it is proposed to use an aluminium alloy or steel which is not included in the tables, application shall be made to the Rotorcraft Project Director for categorization. (The materials considered in Leaflet 406/1 have been based on the SBAC's current rationalized list).

2.3 Materials categorized C in Tables 1, 2 and 3 of Leaflet 406/1 have occasionally been subject to stress corrosion failure in service and shall therefore only be used after consultation with materials specialists, preferably within the contractors organisation or alternatively within MOD. Parts made from these materials shall be designed and manufactured observing all of the precautions listed in para 2.

LEAFLET 406/1

STRESS CORROSION CRACKING

1 DEFINITION

1.1 Stress corrosion cracking was defined by Dix[1] as "spontaneous failure by cracking of a metal under the combined action of high stress and corrosion". Champion[2] has expanded this definition and commented as follows:

> "Susceptibility of a metal to stress corrosion implies a greater deterioration in the mechanical properties of the material through the simultaneous action of a static stress and exposure to a corrosive environment than would occur by the separate but additive action of those agencies".

2 CHARACTERISTICS OF STRESS CORROSION CRACKING

2.1 Sustained tensile stress is a prerequisite for stress corrosion cracking and may be due to service loads, fabrication or assembly techniques, heat treatment or the wedging action of corrosion products. The maximum safe tensile stress level depends upon material susceptibility in the given environment. It is, therefore, always desirable to reduce the residual and assembly stresses to the minimum since it is possible that a combination of residual, assembly and service stresses can reach a level high enough to cause rapid stress corrosion cracking.

2.2 Surface flaws or corrosion pits act as stress concentrators and can thus initiate stress corrosion cracking of a susceptible alloy (see para 2.7).

2.3 With some more susceptible alloys, really hostile environments are not necessary; e.g., at appropriate stress levels moist air (RH 60%) may cause stress corrosion cracking.

2.4 Generally only a few specific environments are effective in causing stress corrosion cracking in a given alloy.

2.5 Stress corrosion cracking can occur at elevated temperatures at which no liquid water is present, e.g., hot salt stress corrosion in titanium alloys (see para 5.2.6).

2.6 The environment which causes stress corrosion cracking is often one that leads to only very local corrosive attack; environments that cause general corrosion do not necessarily cause stress corrosion cracking. There are however exceptions in which neither a pre-existing flaw nor a corrosion pit is necessary for initiating stress corrosion cracking if the environment has the critical composition for the alloy. For example, although a titanium alloy may not undergo stress corrosion cracking in salt water except from a pre-existing crack or flaw, in methanol, stress corrosion cracking initiates readily at a smooth surface of the same alloy.

2.7 In the absence of any significant surface defect, there appears to be a threshold stress (σ_{scc}) for each alloy below which stress corrosion cracking does not occur. There also appears to be a threshold value of plane-strain stress intensity (K_{Iscc}) below which a crack will not propagate at a significant rate by stress corrosion. Both σ_{scc} and K_{Iscc} vary with grain direction; for example, in aluminium alloys susceptibility is far greater when stressed in the short transverse direction.

2.8 Grinding may promote stress corrosion cracking by modifying the surface structure of the metal and by introducing tensile stresses into the surface.

2.9 Stress corrosion cracking usually initiates from the surface of an alloy, but holes can expose highly stressed interior material and thereby precipitate stress corrosion failures.

2.10 Stress corrosion cracking may initiate in unexpected locations and propagate within a part for an appreciable distance (e.g., along a flash line plane) instead of breaking surface quickly. Designers should therefore beware of the danger of hidden cracks occurring, with the resultant unforewarned reduction in static strength.

3 ALUMINIUM ALLOYS

3.1 GENERAL

3.1.1 Severe stress corrosion cracking problems are possible with some aluminium alloys unless the susceptibility of the alloy is reduced by metallurgical means and by the use of appropriate heat treatments; the application of protective schemes cannot alone be relied upon to prevent stress corrosion cracking.

3.1.2 By means of good design (see references) it is possible to make safe use of susceptible alloys, while even alloys of low susceptibility may suffer stress corrosion cracking if the design permits excessive residual and assembly stresses in the short transverse direction. Components should be so designed that defects and other local stress concentrators are minimised: stress corrosion crack growth is possible if the stress intensity exceeds K_{Iscc} even though the loads present would be below σ_{scc} on a defect-free component. It should be noted that the categorization of aluminium alloys (Table 1) takes into account not only σ_{scc} and K_{Iscc} values but also the variation of stress corrosion crack propagation rates with stress intensity for the various alloys.

3.1.3 Machining of components in the fully heat treated condition may expose material in which residual tensile stresses exist. These residual stresses can be minimised by the use of extruded stock or plate which has been controlled stretched, or by the use of cold compression on forgings. Some rough machining operations can be carried out before heat treatment and the consequent reduction in residual stress can be beneficial with regard to stress corrosion susceptibility.

3.2 HEAT TREATMENTS AND STRESS RELIEVING

3.2.1 Cold water quenching of forgings, extruded sections, plate and castings can induce residual stresses of up to 80% of the alloy's proof stress. With plate, extrusions and sheet, controlled stretching can be used to effectively relieve the stresses, and certain forgings can sometimes be stress relieved by a cold compression process. However, many forgings cannot be cold compressed and various special quenching techniques can be used to minimise the level of any residual stresses. These techniques include the use of mixtures of polyalkylene glycols in water as quenchants which result in more uniform cooling rates and hence reduce the level of internal stress.

3.2.2 The conditions at the parting or flash line in forgings can lead to stress corrosion cracking in susceptible alloys. It is recommended that flash line geometry should be discussed with the forger to enable a technique to be defined for each forging which will allow deleterious grain flow to be removed during subsequent machining. Attention to flash line effects is particularly important for hollow cylindrical components such as jacks, where a relatively thin wall is close to the original outer surface of the forging. The use of hand forgings or back extrusions should be considered in such cases.

3.2.3 Even after optimisation of quenching and stress relieving operations, the susceptibility of aluminium alloys is very dependent on subsequent ageing treatments. In general, 2000 series (aluminium copper-magnesium) alloys are susceptible to stress corrosion cracking when aged at room temperature, but by the correct choice of temperature (in the region of 175-190°C) artificial ageing for a few hours can greatly reduce their susceptibility. Compared with naturally aged material, artificial ageing leads to increased tensile strength, but reduced ductility and resistance to fatigue. The 7000 series (aluminium-zinc-magnesium-copper) alloys are very susceptible to stress corrosion cracking in the short-transverse direction when aged to peak strength (T6 temper), but subsequent overageing treatments can lead to progressive improvements in stress corrosion resistance with ageing time (e.g., T76, T736, T73 tempers). There is a progressive reduction in tensile strength on overageing, but toughness is generally improved.

3.3 FORMING TO SHAPE AND OTHER MANIPULATIONS

3.3.1 Appreciable manipulation (other than controlled stretching or compressing) of any form of the material (unless done in the annealed condition or immediately after quenching from solution treatment) will often cause very high residual stresses which may initiate stress corrosion cracking. With the exception of thinner clad sheet, all manipulation (or correction of distortion) should be carefully controlled to minimise residual tension stresses.

3.4 FABRICATION STRESSES

3.4.1 Many situations that give rise to severe residual tension stresses, or to tension stresses in the transverse grain direction, can be avoided at the design stage. Stresses caused by mismatch during assembly can be alleviated by shimming which must be carefully controlled. It may be possible to provide a surface compressive layer by an approved peening process, which may retard or prevent propagation of small cracks by stress corrosion.

Note: The heat treatment, manipulation, fabrication and sustained service loading stresses may combine to produce stresses exceeding the proof stress of the material.

4 STEELS

4.1 GENERAL

4.1.1 It is usually accepted that in the absence of surface defects or other stress concentrators, low alloy steels of below 1450 MPa (93.5 tonf/in^2) maximum specified tensile strength, are not susceptible to stress corrosion cracking in the commonly encountered rotorcraft service environments and that steels over this strength may fail in service when subjected to sustained tension loadings (e.g., landing gear pressure cylinders) in industrial or marine environments. However, some steels of less than 1450 MPa maximum specified tensile strength are readily susceptible to stress corrosion cracking in specific aqueous environments.

4.1.2 Steels which appear to be resistant to stress corrosion cracking by laboratory evaluations using smooth test pieces may indeed suffer stress corrosion cracking from a pre-existing crack or defect (e.g., in a screw thread root), under sustained tensile loading (see para 2.7). This can be determined by testing precracked bend specimens (Brown tests) in a 3½% NaCl environment and the threshold stress intensity factor K_{Iscc} gives a measure of the safe operating stress intensities at a given crack tip under these conditions. It is emphasised that K_{Iscc} generally decreases as tensile strength increases; e.g., increasing the maximum tensile strength from 850 to 1800 MPa (55 to 120 tonf/in^2) can reduce the K_{Iscc} value by a factor of 3.

4.1.3 It is not easy to establish whether failures are due to initial hydrogen embrittlement arising from pretreatment or protective treatment, or to stress corrosion cracking in service which itself is probably by a hydrogen embrittlement mechanism. As tensile strength is increased from 1450 to 2000 MPa (93.5 to 130 tonf/in^2) susceptibility to stress corrosion cracking increases rapidly.

4.1.4 In contrast to aluminium alloys where initiation and propagation are of equal significance, the period of crack initiation in steels is usually greater by a factor of 10^5 than the time for crack propagation. It is therefore important to ensure that the protective treatment scheme is not broken and that any component made from a susceptible steel contains no pre-existing cracks or defects. The early development of a fatigue crack could also be catastrophic. Failures in the higher strength steels (i.e. over 1550 MPa (100 tonf/in^2) tensile strength), exposed to any wet environment, could be produced in a few hours at stresses as low as one half the tensile strength. However, with adequate corrosion protection, certain steels can perform satisfactorily at strength levels as high as 2000 MPa max tensile strength (e.g., DTD 5212).

4.1.5 In general, steels over 1450 MPa (93.5 tonf/in^2) maximum specified tensile strength will require consideration of stress corrosion hazards, and steels over 1800 Mpa (120 tonf/in^2) maximum specified tensile strength, very special consideration. As with aluminium alloys, successful use may be made of the higher strength materials providing the design is good and care taken at every stage of manufacture. Close liaison with structure and materials specialists is essential.

4.1.6 Corrosion resisting steels can suffer severe corrosion and stress corrosion in crevice situations. This effect should be carefully considered; for example when corrosion resisting steel pipes are used, corrosion can occur beneath identification sleeves, clamps and earthing straps if chloride contamination occurs (either from the environment or from the breakdown of contacting materials).

4.2 HEAT TREATMENT AND STRESS RELIEVING

4.2.1 With most steels, high residual internal stresses can be introduced by quenching from the hardening temperature but these may be relieved by subsequent tempering. Steels with higher tempering temperatures are usually more completely stress relieved.

4.2.2 The use of a stress relieving heat treatment (see DEF STAN 03-4) should always be considered if residual tensile stresses have been introduced. This consideration should take account of the magnitude and distribution of the expected residual stresses and the category for susceptibility to stress corrosion cracking indicated in Table 3.

4.3 FABRICATION AND ASSEMBLY STRESSES

4.3.1 As with other materials these should be minimised with the higher strength steels.

4.4 PROTECTION

4.4.1 Protective schemes, such as sacrificial metallic coatings (cadmium, zinc, or aluminium) or phosphating followed by an approved paint scheme, can prevent the initiation of stress corrosion cracking. It is important that any specified de-embrittlement post treatment is carried out to prevent hydrogen embrittlement which might be induced by plating or acidic treatments involved in the overall protection scheme.

5 TITANIUM ALLOYS

5.1 GENERAL

5.1.1 Titanium and its alloys are very resistant to corrosion in most media including marine environments. The metal is usually immune to pitting corrosion and as pitting is normally the initiator of stress corrosion cracking it follows that titanium alloys without crack-like defects are not generally susceptible to stress corrosion cracking in industrial or marine environments.

5.1.2 Certain titanium alloys are susceptible to stress corrosion or sustained load cracking if a crack or defect is present. The threshold stress intensity factor for stress corrosion cracking (K_{Iscc}) can be as low as 25% of the fracture toughness (K_{Ic}). The designer should seek appropriate specialist advice and take due account of these effects.

5.2 SUSCEPTIBILITY IN VARIOUS SPECIAL ENVIRONMENTS

5.2.1 Titanium alloys can be embrittled by contact with certain environments. Those presently recognised are described in the subsequent paragraphs, but others may exist.

5.2.2 CHLORINATED HYDROCARBONS

5.2.2.1 Commercially pure titanium is not susceptible but certain titanium alloys (particularly welded) can be susceptible to stress corrosion when exposed to chlorinated hydrocarbons (e.g., during degreasing or paint stripping). The requirements of DEF STAN 03-2 must be followed.

5.2.3 FLUORINATED SEALANTS

5.2.3.1 Certain sealants when in contact with titanium alloys may cause stress corrosion cracking (see Leaflet 407/2).

5.2.4 CADMIUM

5.2.4.1 There is evidence that cadmium can penetrate the surface of titanium alloys and embrittle them even at temperatures as low as ambient (see Leaflet 407/1).

5.2.5 SILVER

5.2.5.1 Titanium alloys may be embrittled by penetration when in contact with silver or silver containing alloys at temperatures above 300°C.

5.2.6 METHANOL

5.2.6.1 Stress corrosion cracking is possible in certain methanolic solutions. The water content of the methanol is important: cracking will not take place if 2% or more of water is present. Stress corrosion cracking has never been detected in alcohols higher in the homologous series than methanol.

5.2.7 HOT SALT STRESS CORROSION CRACKING

5.2.7.1 Most alloys can be susceptible if they are heated to over 250°C in contact with dry salt residues. In service this environment will normally occur only on or near engine parts, but during fabrication there is also a danger of stress corrosion from this source.

5.2.8 PHOSPHATE ESTER FLUIDS AND PHOSPHORIC ACID

5.2.8.1 Phosphate ester hydraulic fluids (such as Skydrol 500) can cause severe corrosion and embrittlement of titanium alloys operating at temperatures over 120°C, probably because of decomposition of the fluid to phosphoric acid.

5.3 GENERAL PRECAUTIONS

5.3.1 Care in machining (particularly grinding), fabricating and assembly is required, and surface conditioning by controlled peening is generally helpful (see Ref 6).

REFERENCES

No	Author(s)	Title, etc
1	-	Stress Corrosion Cracking in Aircraft Structural Materials AGARD Conference Proceedings No 18. April 1967
2	-	Symposium on the Engineering Practice to avoid Stress Corrosion Cracking AGARD Conference No 53. February 1970
3	-	Specialists Meeting on Stress Corrosion Test Methods AGARD Conference No 98. October 1971
4	E H Spuhler C L Burton	Avoiding Stress Corrosion Cracking in High Strength Aluminium Alloy Structures Alcoa Green Letter
5	D O Sprowls R H Brown	Resistance of Wrought High Strength Aluminium Alloys to Stress Corrosion Alcoa Technical Paper No 17.
6	-	The Aerospace Structural Metals Handbook. US Department of Defence, Belfour Stulen Inc.

TABLE 1

RELATIVE SUSCEPTIBILITIES TO STRESS CORROSION CRACKING OF VARIOUS ALUMINIUM ALLOYS (IN SHORT TRANSVERSE GRAIN DIRECTION EXCEPT FOR SHEET AND TUBE)

Alloy Type and Tempers		Sheet (See Note 1)		Plate		Tube		Extruded Bar and Sections		Forgings	
2014	T3,T4	BS L108 BS L156 BS L158 BS L163 BS L164	A B B A A			BS L105 (See Note 2)	B/C	BS L102 (see Note 3)	B/C	BS L103 DTD 150	C C
	T6	BS L157 BS L159 BS L165 BS L167	B B A A	DTD 5040 BS L93 (See Note 4)	B/C B/C	BS 3L63 (See Note 2)	B/C	BS L168 BS L87 (See Note 4)	B B/C	BS 2L77 (See Note 4)	B/C
2618	T6	DTD 5070	A	2618-T651 (See Note 5)	A			DTD 5014 2618-T6, T62 (See Note 5)	A B/C	DTD 717 DTD 731 (See Note 5) DTD 745	C A/B B
	T71									DTD 5084	A
2219	T6					2219-T62	B	2219-T62	B		
	T8			2219-T851	B	2219-T851	A	2219-T8510	A		
2024 (2124)	T3,T4	BS L109 BS L110	A A	DTD 5100 BS 2L97	C C						
	T8			2024-T851	B						
6082	T6	BS L113	A	BS L115	A	BS L114	A	BS L111	A	BS L112	A
6061	T6					BS L117 BS L118	A A				
7010	T76			DTD 5120	B			7010-T7651	B		
	T736			DTD 5130	A			7010-T73651	A	DTD M239	A/B
7050	T76			7050-T7651	B						
	T736			7050-T73651	A					7050-T736	A/B
7075 (7175) (7475)	T6	BS 2L88	B	DTD 5110 BS 2L95	D D			DTD 5124	D		
	T73			7075-T7351	A			BS L160	A	BS L161 BS L162	A A
7XXX	T6							DTD 5114	D	DTD 5024	D
	T76									DTD 5104	C

NOTE 1: 2014 and 2024 sheet alloys categorized A are clad.
If the cladding is extensively removed the alloys should be categorized B.

NOTE 2: 2014 alloy tubing can suffer intergranular stress corrosion cracking at high hoop stresses. The normal categorization of C may be modified to B if a stress relieving operation is carried out after solutionizing and quenching.

NOTE 3: BS L102 will be categorized B for thin section material (less than 3 mm as extruded) but otherwise will be categorized C.

NOTE 4: 2014-T6 alloys may be categorized B or C according to the ageing temperature employed; material aged at 160-170°C will be categorized C whilst material aged at over 180°C will be categorized B.

NOTE 5: 2618-T6, T62 alloy extrusions will be categorized B for thin section material and categorized C for thick section material. DTD 731 forgings will be categorized A if they are cold compressed before precipitation treatment, but only B if not. Plate, extrusions, and forgings in 2618 will be far more susceptible to stress corrosion than indicated in the Table if the precipitation treatments used are less than those required in the approved specifications.

NOTE 6: Alloys well rated with respect to stress corrosion may still be susceptible to other forms of corrosion e.g. exfoliation (see Chapter 405).

NOTE 7: Apart from sheet and tube the above Table considers stress corrosion susceptibility in the short transverse grain direction (or parting line of forgings) only. It is based on data from stress corrosion tests performed both in natural environments (marine and industrial atmospheres) and by alternate immersion in 3½% neutral NaCl (ASTM G 44-75). The susceptibility is considerably reduced in the long transverse grain direction and is still further reduced in the longitudinal grain direction. Trouble in the long transverse direction is unlikely unless design is poor, and the possibility of trouble in the longitudinal direction can be disregarded. Where the sustained short transverse grain tensile stresses are known to be very low or alternatively where sheet or plate is thin and no short transverse loadings are possible, stress corrosion problems may be less severe than indicated by the ratings in the above Table.

NOTE 8: Where the international alloy and temper designations are quoted in the above Table material specifications approved to Chapter 400 are required.

CLASSIFICATION OF SUSCEPTIBILITY

A Very resistant to stress corrosion cracking.

B Resistant to stress corrosion cracking although failures may occur under extreme conditions.

C Susceptible to stress corrosion cracking. Stress corrosion cracking of these alloys can be expected unless the appropriate precautions are taken at the design stage. Alloys in this category should only be used after discussion with materials specialists.

D Very susceptible to stress corrosion cracking. These alloys should not be used where sustained tensile stresses are applied in the short transverse direction. When possible stresses relieving treatments should be used to reduce quenched-in residual stresses. The use of alloys in category D is restricted and is only permitted with the approval of the Rotorcraft Project Director (see Chapter 406, para 3).

TABLE 2

RELATIVE SUSCEPTIBILITIES TO STRESS CORROSION

CRACKING OF VARIOUS CAST ALUMINIUM ALLOYS

Alloy Type and Temper		UK Specification	Susceptibility
242.0	T6	BS 4L35	C
295.0 B295.0	T6 T6	BS2L92	C C
355.0 C355.0	T6 T6	BS3L78	B B
-	T6	BS 3L51	C
356.0 A356.0	T6 T6	BS 2L99	A A
A357.0	T6	-	A
K0.1) Avior B)	T6 T7	-	D C
-	T6	BS L119 (RR 350)	C
-	T4	BS L154	A/B
-	T6	BS L155	A/B

CLASSIFICATION OF SUSCEPTIBILITY

A Very resistant to stress corrosion cracking.

B Resistant to stress corrosion cracking although failures may occur under extreme conditions.

C Susceptible to stress corrosion cracking. Stress corrosion cracking of these alloys can be expected unless appropriate precautions are taken at the design stage. Alloys in this category should only be used after discussions with materials specialists.

D Very susceptible to stress corrosion cracking. These alloys shall not be used without the prior approval of the Rotorcraft Project Director (see Chapter 406, para 3).

TABLE 3
RELATIVE SUSCEPTIBILITY TO STRESS CORROSION CRACKING OF VARIOUS STEELS ETC.

	MAXIMUM SPECIFIED TENSILE STRENGTH IN RANGE (MPa)		BARS AND FORGINGS		ONLY FOR BOLTS (ETC)		SHEET AND PLATE		TUBE		CASTINGS		SPRINGS	
	OVER	NOT OVER												
Non-Corrosion Resisting Mild Steels & Low/Medium Alloy Steels	-	1450	BS.S.91 BS.S.95 BS.S.98 BS.S.99 BS.S.131 BS.S.139 BS.S.140 BS.S.142 BS.S.153 BS.S.154 DTD.5082	A A B B A A A A B A A	BS.S.147 BS.S.149 BS.S.158	B B B	BS.S.534 BS.S.535	A B	BS.T.53 BS.T.60 BS.T.65	A B A	BS.HC.3 BS.HC.4 BS.HC.7 BS.HC.8 BS.HC.402 BS.HC.403 BS.HC.404	A A B C A A A	BS.S.201 BS.S.202 BS.S.203 BS.S.204	B B B B
	1450	1550	-	-	-	-	-	-	-	-	-	-		
	1550	1800	BS.S.134 BS.S.138	C C	- -	- -	- -	- -	- -	- -	- -	- -		
	1800	-	BS.S.135 BS.S.136 BS.S.146 BS.S.155 (300 M Var)	B B D D	DTD.5222	D	-	-	-	-	-	-		
Maraging Steels	1800	-	DTD.5212	C	-	-	(New Spec)	C	-	-	BS.HC.401	C	-	
Nitriding/Carburising Steels	-	1450	BS.S.15 BS.S.106 BS.S.133 BS.S.157	A A A A	-	-	-	-	-	-	BS.HC.5 BS.HC.6	A B		
	1450	1550	BS.S.132 BS.S.156 (S82 Type)	B B	-	-	-	-	-	-	-	-	-	
Precipitation Hardening Steels	-	1450	BS.S.143 BS.S.144	B B	BS.HR.650	A	BS.S.533	B	-	-	BS.HC.101	B	-	
	1450	1550	BS.S.145	C	-	-	BS.S.532	B	-	-	BS.HC.102 BS.HC.106	C C	-	
	1550	1800	-	-	-	-	-	-	-	-	-	-	-	
	1800	-	-	-	-	-	-	-	-	-	-	-	DTD.5086	C*
Other Corrosion and/or Heat Resisting Steels	-	1450	BS.S.62 BS.S.80 BS.S.126 BS.S.129 BS.S.130 BS.S.137 BS.S.150 BS.S.151 BS.S.152 DTD.5066	A B A A A B B B B A	DTD.5076	B	BS.S.524 BS.S.525 BS.S.526 BS.S.527 BS.S.530 BS.S.531 BS.S.536 BS.S.537 BS.S.538	A A A A B B A A B	BS.T.66 BS.T.67 BS.T.68 BS.T.69 BS.T.72 BS.T.73 BS.T.74 BS.T.75 (21.6 9 Type)	A A A A A A A A A	BS.HC.103 BS.HC.104 BS.HC.105 M.190	A A A B	BS.S.205	A
	1450	-	-	-	-	-	-	-	-	-	-	-	-	-
	1800	-	440	C	-	-	-	-	-	-	-	-	DTD.326	D*
Nickel Alloys	-	1450	These forms are not covered by this Table										BS.HR.502	A
	1800	-											BS.HR.501	A

NOTE 1 Maximum specified tensile strengths for spring steels BS.S.201/5 vary widely with wire dia (refer to material specifications for details). Other materials are grouped according to type of steel and to strength ranges related to strengths arising in para 4 of this Leaflet.

NOTE 2 Many steels, including some low strength steels, are susceptible to stress corrosion cracking in hot caustic and nitrate solutions.

NOTE 3 Steels, especially springs, immersed in hydraulic fluid or oil may be so protected, and even category D materials may be safely used.

NOTE 4 Electropolishing can improve the resistance of the spring steels marked * to category B, but this treatment should not be applied to light gauge springs.

NOTE 5 Spring steel BS.S.205 and nickel alloys HR501 and HR502 are recommended for use in high humidity and other corrosive environments.

NOTE 6 Non-sacrificial metallic coatings can adversely affect the stress corrosion performance of steels and such coatings (eg electroplated silver) should be avoided.

CLASSIFICATION OF SUSCEPTIBILITY

A Very resistant to stress corrosion cracking in commonly encountered environments.

B Resistant to stress corrosion cracking. When pre-existing cracks or defects are present, failures may occur under sustained tension stresses in wet environments. This group of alloys can often be used without many stress corrosion design limitations. The normal protective treatments applied to stop rusting also give good protection against stress corrosion to non-corrosion-resisting steels in this category.

C Susceptible to stress corrosion cracking. Stress corrosion cracking of these steels can be expected unless the appropriate precautions are taken at the design stage. Steels in this category should only be used after discussion with structures and materials engineers.

D Very susceptible to stress corrosion cracking. The use of steels in Category D is restricted and is only permitted with the approval of the Rotorcraft Project Director (see Chapter 406, para 3).

CHAPTER 407

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

1 INTRODUCTION

1.1 This chapter contains the requirements for the protection of Rotorcraft and their equipment against corrosion and environmental deterioration.

2 BASIC OPERATIONAL REQUIREMENTS

2.1 The aim shall be that all parts of the rotorcraft, both inside and outside, shall be so designed, protected, drained and vented that, when the Rotorcraft is maintained in accordance with the Servicing Schedule, there will be no unacceptable loss of airworthiness as a result of weathering, corrosion, abrasion, unavoidable mechanical damage to protective treatment during normal maintenance or other causes.

2.2 The Rotorcraft should therefore be able to withstand satisfactorily:

(i) the effects of standing for long periods in the open in all weathers, including the corrosive effects of contamination by an industrial atmosphere,

(ii) the corrosive effects of operating from salt-laden runways and of flying low over the sea, and

(iii) on Naval Rotorcraft, the corrosive effects of salt spray and funnel gases, when stowed in the open or operating from the deck of a ship.

Notes: 1 Requirements relating to the extremes of temperature and humidity that may be experienced on the Ground and in flight are given in Chapter 101.

2 Requirements on operational, camouflage and identification markings are given in Chapter 103.

3 EXCLUSION OF CONTAMINATING LIQUIDS

3.1 GENERAL

3.1.1 All static joints shall be sealed to prevent the ingress of liquids. Unless otherwise excepted in para 23.4, sealing shall be effected by wet assembly with a suitable sealant, or jointing compound (see para 23.3).

3.1.2 As a supplement to the sealant within the lap joints, the application of sealing or caulking compounds to exposed edges of the lap to prevent the ingress of liquids shall be considered.

3.1.3 Particular care shall be taken to prevent the wetting of equipment and heat and sound proofing material.

3.2 RAIN AND AIRBORNE SPRAY (See also Chapter 101 Para 5 and Chapter 1013)

3.2.1 Particular care shall be taken to prevent water leaking into, or being driven into, any part of the rotorcraft either on the ground or in flight. All windows, doors, panels, canopies etc., shall be provided with sealing arrangements such that the entry of water is prevented when these items are correctly closed. Where sealing is not practicable, precautions shall be taken to ensure that any water that does gain access to the interior of the rotorcraft does not constitute a direct or indirect danger.

3. 3 OTHER FLUIDS

3.3.1 The design of the rotorcraft shall be such as to minimise the possibility that any fluid - which may leak from a system, enter the rotorcraft from the outside, condense on cold surfaces (eg, pressure cabin walls) or may be spilt within the rotorcraft - will cause corrosion. The following fluids shall be taken into account:

(i) Water, cleaning fluid, ground and airborne de-icing fluids, fuel, oil, hydraulic fluid, battery electrolytes, and fluids associated with galley and toilets.

(ii) Slush (including grit, salt or other runway de-icing chemicals) which will impinge on, or possibly enter, parts of the rotorcraft during taxying, take-off or landing on precipitation-covered runways.

3.3.2 The use of corrosive fluids should be avoided in rotorcraft systems (eg, in toilets). Adequate access, to the satisfaction of the Rotorcraft Project Director, shall be provided for the maintenance of all joints and unions in pipe lines.

3.3.3 Particular attention shall be given to the design of those compartments where liquids are likely to be spilt. Sealed floors with suitable drainage shall be provided in battery areas, galleys and toilets (See also Leaflet 407/5). Urinals, sick basins, lead-away pipes, and collector boxes shall be manufactured from urine-resistant material.

3.3.4 Heat and sound proofing materials shall not cause corrosion either by direct contact or as a result of leachants if wetted. Precautions shall be taken against the possibility of corrosion caused by materials acting, as a wick. Suitable provision shall be made so that it is practicable to inspect for any signs of corrosion due to these causes.

3.4 DRAINAGE AND VENTING

3.4.1 All compartments in the structure, including those in control surfaces, cockpits and cabins, shall be completely sealed or adequately drained and vented, both on the ground and in flight. The design of each drainage system shall be such that it will still be effective with any one drain or drain-hole blocked. Pockets in the structure which cannot be drained shall be filled with inert, non-absorbent, caulking or filling material. Honeycomb structure is dealt with in Para 10.2.6.

3.4.2 Particular attention shall be given to the location and size of internal drain-holes to minimise the possibility of fluids being trapped in the structure and being prevented from running to a drain. Care shall be taken to avoid undrained pockets especially where liquids could freeze and so jam, mechanisms.

3.4.3 In pressurised compartments, automatic drains, that open when pressure is released, are preferred.

3.4.4 The position of drain-holes shall be shown on component drawings and a complete key shall be given in the Rotorcraft Servicing Manual.

4 ACCESS FOR EXAMINATION

4.1 The aim shall be to provide access to every part of the structure to facilitate visual examination for corrosion or deterioration at a reasonable manhour cost. Although visual aids (eg, endoprobes) may be used to meet this requirement, it is desirable to be able to examine the majority of the structure without visual aids. Ideally, it should be possible to gain access simply by removing, access panels.

4.2 The Contractor shall demonstrate compliance with the requirement of para 4.1 to the satisfaction of the Rotorcraft Project Director.

5 DEHUMIDIFICATION OF AIRCRAFT STRUCTURE AND SYSTEMS

5.1 Dehumidification of aircraft whilst on the ground, to reduce metal corrosion and improve the reliability of electrical and avionic systems, may be a requirement for aircraft. See Leaflet 407/7.

5.2 The Rotorcraft Specification will identify;

(i) Whether dehumidification is required on individual aircraft, or whether area dehumidification is to be used. If dehumidification is required on individual aircraft it must be provided for in the aircraft design.

(ii) For on-aircraft dehumification, to what extent standard Service equipment is to be used.

(iii) Whether specific air flow and air change rates are required.

5.3 Dehumidification is not to be assumed as a means of preventing corrosion to meet the structural life requirement or to meet the reliability requirement of the Rotorcraft Specification.

6 DESIGNATION OF THE REQUIREMENTS FOR THE PROTECTION OF PARTS AND ASSEMBLIES

6.1 The requirements for the protection of parts and assemblies used in the construction of rotorcraft are given in the subsequent paragraphs of this chapter.

6.2 The materials and processes used shall be to specifications approved for aircraft use. A list of specifications is given in Table 1. Approved proprietary materials and processes are given specification DTD 900. Other materials and processes may be used in accordance with DEF STAN 05-123.

6.3 The treatments appropriate to a detail item shall be indicated on the drawings, of the item and any to be applied during and after assembly, on the appropriate assembly drawing. This may be by direct reference to the appropriate material or specification number or may be by coded reference to such information, which has been set down as a comprehensive schedule included in the master drawings.

6.4 Treatment requirements shall be sufficiently specific that they permit only those processing options that are acceptable, (eg, 'anodise to DEF- 151, Type 2, chromic acid').

7 TREATMENT OF METAL PARTS, GENERAL

7.1 GENERAL REQUIREMENTS

7.1.1 All metal parts shall be given an appropriate protective treatment as set out in subsequent paragraphs unless otherwise excepted (See Para 7.7). In the case of fasteners see Para 7.4.2.

7.2 CLEANING

7.2.1 All metal parts shall be cleaned in accordance with DEF STAN 03 -2.

7.3 SURFACE TREATMENT

7.3.1 All metal parts shall be given a surface treatment or plating treatment as specified in the appropriate paras 8 to 13 below. Components shall not be treated in the assembled condition unless danger of corrosion by trapped treatment chemicals can be completely avoided and there is no danger of removal or degradation of sealants, jointing compounds etc.

7. 4 PAINTING

7.4.1 Where called for in the appropriate paras 8 to 13 below, metal parts shall be painted. Painting shall be in accordance with DEF STAN 03-7 and with the rotorcraft specification.

Note: In the chapter, the meanings of the terms 'paint', 'primer', 'etch primer', 'finish' (in relation to painting), 'finishing coat' etc., are as defined in BS 2015 'Glossary of paint terms'.

7.4.2 Fasteners shall not be painted prior to use, but shall where required, be painted after assembly (See Para 7.6.3).

7.5 STANDARD PAINT SCHEMES FOR ALL ROTORCRAFT (excluding areas made of doped fabric)

7.5.1 General: Etch primer is regarded as a pretreatment, not as a full part of the paint scheme itself. It is not acceptable where high resistance to aggressive fluids (eg phosphate-ester hydraulic fluids, castor oil and synthetic lubricants) is required.

7.5.2 Exterior surfaces: All surfaces of a rotorcraft that are exposed to the external environment either permanently, as part of the exterior construction, or transitorily to a significant extent, and are made of metals required to be painted in accordance with the requirements of this chapter, or that are to be painted for the sake of appearance or to meet operational requirements, shall be given a full paint scheme of at least primer and finish to give, when dry, a minimum film thickness of 50 µm.

Note: External surfaces include wheel-wells, undercarriages, wing-folds, air brakes, wing flaps, fairings or other areas which may be occasionally exposed to the external environment or into which water may be driven or collect.

7.5.3 Interior surfaces (heavy duty): Interior surfaces subject to heavy condensation, aggressive hydraulic fluids, or other potentially corrosive fluids noted in Paras 3.2 and 3.3, shall be coated with a full scheme of at least primer and finish, or two full coats of primer, resistant to these fluids. In all cases interior finishing schemes for heavy duty protection shall have, when dry, a minimum film thickness of 40µm.

7.5.4 Interior surfaces (normal): Interior surfaces, not subject to heavy condensation or severe contamination with aggressive fluids or special requirements for colour, shall be coated with epoxy primer to DTD 5567 unless otherwise specified, to give a minimum film thickness when dry of 20µm.

7.5.5 Interior surfaces of integral fuel tanks: This is dealt with in detail in Para 14.2.

7.5.6 Other paint schemes: Steel parts forming part of mechanisms may be painted with stoving enamel (DTD 56, BS X 31). Other paint schemes and other organic coating schemes, eg, plastic coatings and anti-fretting coatings, may be used for the purposes for which they are approved.

7. 6 PAINTING IN RELATION TO ASSEMBLY (excluding the specific cases covered by Para 7.7) (for wet assembly see Para 23 and for touch-up after assembly see Para 24)

7.6.1 Components shall be given at least a coat of primer and preferably painted to the requirements of Para 7.5 before assembly.

7.6.2 Components which have only been primed before assembly, and which require a finishing coat after assembly, should preferably be primed again overall.

7.6.3 After assembly, where required, all fasteners shall be painted according to Para 7.5, (See also Paras 10.2.1 and 10.2.2) and all damaged areas shall be painted according to Para 24.2.

7.7 EXCEPTIONS AND SPECIAL CASES

7.7.1 Bushes: Where bushes are inserted into aluminium or magnesium alloys, especially those exposed to the atmosphere, (eg, in undercarriage assemblies), particular care shall be taken in the application of protective surface treatments and in the prevention of ingress of water to the mating surfaces. Where a plating treatment is required to prevent galvanic corrosion, eg, the cadmium or tin plating of copper-alloy bushes (See Para 12.2.1), the maximum thickness of plate shall be applied consistent with satisfactory assembly. Bushes shall be wet-assembled (See Para 23) even though the greater part of the jointing compound is exuded from the fit. Bushes with flanges shall be similarly wet-assembled so that a fillet is formed around the flange (See Para 23.2.1). As a supplement to sealing within the joint, the application of sealing or caulking, compounds along the lap shall be considered.

7.7.2 Metallic Shims: When shims are used during assembly, special attention shall be paid to wet assembly. Both sides of the shim shall be coated with sufficient wet assembly material so that a fillet is formed covering the edge of the shim. As a supplement to sealing within the joint, the application of sealing or caulking compounds along the edges of the lap shall be considered.

7.7.3 Lubricated parts: Surfaces that run in a maintained environment of oil or grease generally require no protective treatment (See also Paras 8.5.5 and 10.2.5). Magnesium alloy parts shall however, be protected to the requirements of DTD 911.

7.7.4 Tubes and pipes: Tubes and pipes carrying operating fluids need not be painted internally. If they are painted, the paint scheme chosen shall be resistant to the fluid and shall be suitable for continuous immersion in the fluid.

7.7.5 Contact with fabric dope: To prevent softening of the finishing coat by dope solvents, all surfaces liable to come into contact with fabric during doping operations shall be treated so as to be compatible with dope. This requirement does not apply to surfaces to which fabric is attached by an adhesive.

7.7.6 Rubbing surfaces: These may require coatings or treatments to prevent galling or wear. In making a choice, the contractor shall take into account their adequacy in protecting against corrosion in the environment in which the parts will operate, and their effects upon fatigue life (See also Paras 8.6.3 and 10.2.5).

7.7.7 Electrical bonding: To achieve satisfactory electrical bonding to the rotorcraft structure, it is sometimes necessary to remove, or to omit locally, any surface protection. At such bonding points, precautions shall be taken to prevent the ingress of any corrodent or operating fluid by suitable use of jointing compounds, sealants and caulking materials, followed where possible by painting.

7.7.8 Holes machined after painting: Except in the case of magnesium alloys, holes and countersinks machined in parts which have been painted need not be given any protective treatment prior to wet assembly (See Para 23) with fasteners (See Paras 23.2.2 and 23.2.3), or insertion of bushes (See Para 7.7.1). After assembly the parts shall be further protected according to Para 7.6.3 or Para 7.7.1.

8 TREATMENT OF NON-CORROSION-RESISTING STEELS

8.1 These requirements apply to all parts, including fasteners, made of steel containing less than 9% chromium.

8.2 GENERAL REQUIREMENTS

8.2.1 Cleaning: All steels shall be cleaned (DEF STAN 03-2) and, unless otherwise excepted, given one of the treatments in Para 8.2.2 or Para 8.2.3.

8.2.2 Standard treatment for all parts for use below 235°C: Cadmium plating (DEF STAN 03-19) followed by painting. Prior to painting the cadmium shall not be either passivated (DEF-130) or etch primed DEF STAN 80-15 and proprietary alternatives covered by approved specification for paint schemes) if etch primer is acceptable, (See Para 7.5.1). Fasteners shall be passivated DEF-130) after cadmium plating, and painted only after assembly (See Paras 7.4.2 and 7.6.3).

8.2.3 Alternatives to cadmium plating in special cases:

(i) Aluminium coatings: Coating with aluminium by an approved process followed by painting, and for large parts and parts with wide tolerances, spraying with aluminium (BS 2569: Part 1) and then painting. Unpainted aluminium coatings may be suitable for use at elevated temperatures.

(ii) Zinc coatings: Parts subject to temperatures between 235°C and 350°C may be zinc plated (DEF STAN 03-20), chromate passivated (DEF-130) and painted with a suitable heat-resisting paint scheme. Except on Naval rotorcraft, other rotorcraft designed to operate in a marine environment, and on fasteners, zinc plating may be used in place of cadmium plating on normal interior surfaces (as defined in Para 7.5.4) of parts subject to service at any temperature below 350°C.

(iii) Phosphate coatings: Phosphate treatment to Class I or II DEF STAN 03-11 followed either by painting or other suitable approved coating schemes, eg , plastic coatings and anti-fret coatings (See Para 7.5.5), may be used when cadmium plating is not technically feasible or desirable.

(iv) Grade B parts only: These may be zinc coated by electroplating (DEF STAN 03-20) or any other suitable process, nickel or nickel/chromium plated (DEF STAN 03-10), tin plated (DEF STAN 03-8), tin-lead alloy plated or made from tinned steel (BS 2920). Tin coated parts shall not, however, be used in contact with cadmium plated parts where the temperature in service may exceed 120°C. The protective metal coating shall be followed by etch priming and painting. Alternatively Grade B parts may be phosphate treated (DEF STAN 03-11), followed by painting or other suitable approved coating schemes.

(v) Vacuum cadmium coatings: Cadmium coating by vacuum evaporation may be employed (DTD 940) instead of cadmium plating to DEF STAN 03-19.

8.3 HIGH STRENGTH STEELS

8.3.1 Parts made of steel of maximum specified tensile strength exceeding 1450 MPa shall be subject to the special requirements of DEF STAN 03-4.

8.4 STRUCTURAL TUBING

8.4.1 Outer surfaces should be sprayed with aluminium or plated with cadmium if possible but phosphate treatment plus a full paint scheme is acceptable.

8.4.2 It shall be assumed that the inside of tubular structures will breathe and that water will condense and cause corrosion from the inside unless substantial protection is given. Prior to welding or other assembly procedures, any manufacturing or heat-treatment scale shall be removed, (eg, by pickling). Phosphate treatment plus slushing with an inhibitive primer is desirable, but if aqueous treatments are impossible, (eg, in a complicated welded structure), then insides shall be thoroughly slushed with a supplementary protective, (eg, PX-28), suitably thinned. Access holes shall be provided if needed and, whenever possible, shall be securely sealed after the organic protective has been introduced. Where secure sealing is not possible, drainage holes shall be provided.

8.5 EXCEPTIONS AND SPECIAL CASES

8.5.1 Springs: The advice in Defence Guide DG-10 shall be followed. Where protection is required, cadmium plating and paint is the preferred protective.

8.5.2 Gun parts: Steel gun parts classified as airframe components (eg, deflector chutes, ammunition belt feeds and guides) shall be treated by Sherardising (BS 4921), hard chromium plating (DEF-160) or electroless nickel plating (DEF STAN 03-5).

8.5.3 Parts subject to wear: Where increased wear resistance is required, hard chromium plating (DEF-160) is in many cases an acceptable process. Electroless nickel (DEF STAN 03-5) or nickel plating (DTD 905) may be used in some cases. A number of other approved hard surfacing treatments are available for special application (DTD 941, DTD 943 and DTD 900). In making a choice, the contractor shall take into account the adequacy of the treatment in protecting against corrosion in the environment in which the parts will operate, and its effect upon fatigue life.

8.5.4 Wire Ropes: Zinc coated wire ropes which have been treated with a lubricant during manufacture shall be used, and shall be installed as supplied.

8.5.5 Lubricated parts: Surfaces that run in a maintained environment of oil or grease need not be protected against corrosion. Phosphate (DEF STAN 03-11, Class I) or black oxide (DTD 900) treatment may be given if desired.

Note: Black oxide treatments should be avoided wherever possible, and shall not be used on steels of maximum specified strength exceeding 1450 MPa. This restriction does not apply to surface hardened parts or to parts made from 1% C to 1% Cr bearing steel.

8.5.6 Safety Harness Components. When Safety Harness Components are made from non-corrosion resisting steels and particles of paint produced in use by abrasion/wear could cause detriment to the operation/function of a mechanism, then cadmium plating and passivation without further painting is the preferred protection. (For design of safety harness components see Chapter 111).

9 TREATMENT OF CORROSION RESISTING STEELS

9.1 These requirements apply to all parts, including fasteners, made of steel containing not less than 9% chromium.

9.2 GENERAL REQUIREMENTS

9.2.1 Designs embodying corrosion resisting steels shall avoid crevices where practicable. Where crevices are unavoidable, these shall be sealed wherever possible with an approved sealant (See Paras 23. 2, 2 3. 3). Where sealing is not practicable, (eg, on moving parts), specialist advice shall be sought on the selection of a corrosion resisting steel with the optimum combination of required properties and resistance to crevice corrosion (See Leaflet 407/6).

9.2.2 All parts made from corrosion resisting steels shall be cleaned by appropriate methods in accordance with DEF STAN 03-2 to ensure freedom from surface contamination. In particular, the surfaces of forgings, castings, welds, "black" bars, heat treated parts etc., shall have sufficient material removed to ensure freedom from contamination caused by manufacturing operations, where these surfaces are to be present on the finished parts.

9.2.3 Corrosion resisting steel wire ropes shall be installed as supplied.

9.2.4 In certain environments, especially maritime, corrosion resisting steels can suffer severe pitting corrosion and advice shall be taken on the choice of alloy and suitable protective schemes (See Leaflet 407/6) .

9.2.5 Parts for use above 235°C shall not be coated with cadmium.

9.3 PARTS IN CONTACT WITH ALUMINIUM ALLOY (See Para 5.2)

9.3.1 These parts shall be treated by one of the following procedures:

(i) Cadmium plated (DEF STAN 03-19), optionally followed by painting, and then wet assembled (See Para 23).

(ii) Zinc plated (DEF STAN 03-20), optionally followed by painting, and then wet assembled (See Para 23).

(iii) Zinc sprayed (BS 2569, Part 1) optionally followed by painting and then wet assembled (See Para 23).

(iv) Coated with aluminium by any approved method (DTD 900) optionally followed by painting and then wet assembled (See Para 23).

(v) Painted, and then wet assembled (See Para 23) using, optionally, a sacrificial metal shim.

(vi) Painted and assembled with a non-metallic separator.

(vii) Painted and assembled using an elastomeric sealant.

9.4 PARTS IN CONTACT WITH MAGNESIUM ALLOY (See Para 5.2)

9.4.1 Surfaces that will be wet assembled to contact magnesium alloy shall be treated by one of the following procedures and then painted where this is feasible prior to wet assembly (See Para 23).

(i) Cadmium plated (DEF STAN 03-19) and chromate passivated to the requirements of DEF STAN 03-33,

(ii) Zinc plated (DEF STAN 03-20) and chromate passivated to the requirement of DEF STAN 03-33,

(iii) Zinc sprayed (BS 2569, Parts 1 and 2) and chromate passivated to the requirements of DEF STAN 03-33,

(iv) Shimmed with zinc which has been chromate passivated to the requirements of DEF-130, chromate conversion coated 5000 and 6000 aluminium series washers to DEF STAN 03-18, anodised aluminium washers, or with insulating material, eg, Nylon. The shim shall extend from the area of contact to an extent sufficient to break a likely electrolytic path.

10 TREATMENT OF ALUMINIUM AND ALUMINIUM ALLOYS

10.1 GENERAL REQUIREMENTS

10.1.1 All parts shall be cleaned (DEF STAN 03-2) and with the exception of those listed in Para 10.2, shall be given one of the following treatments followed by painting:

(i) Anodised (DEF STANs 03-24, 03-25, & 03-26).

(ii) Chromate filmed (DEF STAN 03-18).

(iii) Etch primed (DEF STAN 80-15 and proprietary alternatives covered by approved specifications for paint schemes) when treatments (i) or (ii) are not technically feasible or desirable and when etch primer is acceptable (See Para 7.5.1).

(iv) Sprayed with aluminium, aluminium-zinc alloy, or zinc (DEF STAN 03-3), followed by etch primer if etch primer is acceptable (See Para 7.5.1) or, in the case of aluminium or aluminium-zinc alloy, by chromate filming (DEF STAN 03-18)).

10.1.2 Selection of treatment: The consideration affecting the choice of the treatment from those listed in Para 10.1.1 shall include the following:

(i) The suitability of a casting for anodising, and to some extent the type of process used, will depend on the alloy.

(ii) Sprayed metal coatings are not suitable for close tolerance parts or for thin sheet.

(iii) Some alloys containing a high proportion of alloying elements do not give good coatings by anodising or chromate filming. The treatment shall be chosen that gives a film of good appearance and properties.

(iv) Where anodising is required as an aid to inspection for cracks, laps etc., or of grain flow, the chromic acid process (See DEF STANs 03-24, 03-25, & 03-26) shall be used.

(v) Selection of the treatment to be used will be affected if parts are to be adhesive bonded.

(vi) While sulphuric acid anodising processes may reduce fatigue strength, the chromic acid process has comparatively little effect on fatigue properties.

(vii) Where fatigue is a major design consideration, holes should not be anodised.

10.1.3 Parts that cannot be painted: These parts shall be anodised. Thin anodic films with relatively poor resistance to corrosion can result when overaged aluminium-zinc alloys (eg, BS L160, BS L161, BS L162) are anodised by the standard chromic acid process.

10.2 EXCEPTIONS AND SPECIAL CASES

10.2.1 Aluminium alloy fasteners other than rivets: Except where used in electrical circuits, these shall be anodised before wet assembly. All fasteners shall be painted with a complete scheme after assembly.

10.2.2 Solid rivets: Solid rivets used as supplied need not be given any surface protective treatment prior to wet assembly (See Para 23). In the case of rivets placed in the freshly solution treated condition, care shall be taken to remove completely all salt residues from the heat-treatment process. After assembly, exposed areas of all rivets shall be painted with the paint scheme as specified in Para 7.6.3.

10.2.3 Structural and other tubing: Tubing shall be treated externally as in Para 10.1.1. Structural tubing shall be treated internally as in Para 10.1.1. For the interior of tubes carrying operating fluids, See Para 7.7.4.

10.2.4 Welded parts: When welded parts are to be anodised, the chromic acid process shall be used.

10.2.5 Parts subject to wear: Where the standard anodising processes give sufficient resistance to wear, parts may be hard anodised, chromium plated (DEF STAN 03-14) or coated with hard facing material (DTD 900). Anodic films may be impregnated with approved solid film lubricants (DTD 900). In selecting a process, the contractor shall consider the effect which the coating may have upon fatigue life and corrosion resistance of the aluminium alloy.

10.2.6 Honeycomb core: Core materials made of aluminium alloys shall, at the foil stage of manufacture, be given a chromate filming treatment (DEF STAN 03-18) or be chromic acid anodised if this is practicable. Alternatively, the core material may be given an etching treatment (DTD 900) but only if an inert organic coating is to be applied subsequently. At the expanded honeycomb stage of manufacture, an inert organic coating shall be applied wherever practicable to ensure good corrosion resistance of the honeycomb core material and good durability of adhesive bonds.

10.2.7 Items adhesively bonded to honeycomb core: The treatment of skins, etc., bonded to the core shall be selected so that a good bond is obtained and so that external surfaces are suitable for subsequent protection (See Para 10.2.8).

10.2.8 Completed honeycomb structure: The external surfaces of the completed honeycomb structure shall be protected by the appropriate paint scheme (See Para 7.5).

10.3 INFLUENCE OF PROCESSING TEMPERATURE, ON ALUMINIUM ALLOYS

10.3.1 Temperatures used in processing (eg, curing of organic coatings or adhesives) parts made from aluminium alloy shall be such that there is no deleterious effect upon the alloy properties.

11 TREATMENT OF MAGNESIUM ALLOYS

11.1 All parts shall be given the complete treatment specified in DTD 911.

11.2 Other protective schemes such as plastic coatings (eg Nylon) and anodic treatments (eg the HAE or DOW 17 processes) may be used subject to compliance with the requirements of DEF STAN 05-123.

12 TREATMENT OF COPPER AND COPPER BASE ALLOYS

12.1 GENERAL REQUIREMENTS

12.1.1 All parts shall be cleaned (DEF STAN 03-2). Except as required by Para 12.2, no further treatment is needed. Parts may be painted if desired, in which case the copper base material should be given a mild abrasive blasting, or be etched (DEF STAN 03-2) or etch primed (See Para 7.5.1).

12.2 PARTS IN CONTACT WITH OTHER METALS

12.2.1 Aluminium alloys: The copper base material shall be cadmium plated (DEF STAN 03-19) or tin plated (DEF STAN 03-8) and optionally painted prior to wet assembly (See Para 23.2).

12.2.2 Magnesium alloys. The copper base material shall be cadmium plated (DEF STAN 03-19), chromate passivated to the requirements of DEF STAN 03-33 and optionally painted prior to wet assembly (See Para 23.2).

12.2.3 Non-corrosion resisting steels: A cadmium coating shall be interposed between the two metals by plating (DEF STAN 03-19), optionally followed by painting, at least the steel surface and preferably also that of the copper base part prior to wet assembly (See Para 23.2).

12.3 PARTS SUBJECT TO CONDENSATION

12.3.1 Surfaces from which condensation can drip on to other metals shall be coated to prevent dissolution of traces of copper. Suitable coatings are: cadmium plate (DEF STAN 03-19) preferably painted, tin plate (DEF STAN 03-8), or paint (See Para 12.1).

13 TREATMENT OF TITANIUM AND TITANIUM BASE ALLOYS

13.1 These requirements apply to all parts including fasteners.

13.2 GENERAL REQUIREMENT

13.2.1 All parts shall be cleaned (DEF STAN 03-2). Except as required by Para 13.3, no further treatment is needed. Parts may be painted if desired, in which case the metal shall be pretreated by one of the following processes:

(i) mild wet abrasive cleaning (DEF STAN 03-2, Methods D2 and D3),

(ii) etch primer (Para 7.5.1),

(iii) pickling (DEF STAN 03-2, Method S),

(iv) anodising (DTD 942).

13.3 PARTS IN CONTACT WITH OTHER METALS

13.3.1 Magnesium alloys. Contact with magnesium alloy shall be avoided wherever possible. Surfaces that will be assembled to contact magnesium alloy shall be treated by one of the following procedures and then painted where it is feasible prior to wet assembly (Para 23):

(i) zinc plated (DEF STAN 03-20) and chromate passivated to the requirements of DEF STAN 03-33,

(ii) zinc sprayed (BS 2569 Parts 1 and 2) and chromate passivated to the requirements of DEF STAN 03-33,

(iii) coated with aluminium or an aluminium rich coating by an approved method or conversion coated to the requirements of DEF STAN 03-18,

(iv) Shimmed with zinc, and chromate passivated to the requirements of DEF STAN 03-33, chromate conversion coated 5000 and 6000 aluminium series washers to DEF STAN 03-18, or with insulating material, eg, Nylon. The shim shall extend from the area of contact to an extent sufficient to break any likely electrolytic path,

(v) anodised (DTD 942) and coated with a resin containing molybdenum disulphide (DEF STAN 91-19).

When an interlayer is not possible both the magnesium alloy and titanium surfaces shall be fully protected as defined in Chapter 407, Para 11.1, 11.2 and 13.2.

13.3.2 Aluminium alloy: Surfaces that will be assembled to contact aluminium alloy should first be painted but where painting is omitted, special attention shall be paid to wet assembly (See Para 23.2). Alternatively, the titanium base material may be plated with zinc (DEF STAN 03-20), metallised with zinc (BS 2569) or coated with aluminium or an aluminium-rich coating by an approved method. Titanium based materials may be anodised (DTD 942) and where required coated with a resin containing molybdenum disulphide (DEF STAN 91-19).

13.3.3 Non-corrosion-resisting steel: No treatment of the titanium is required. The steel may be coated with aluminium or an aluminium-rich coating or coated with cadmium before wet assembly. It shall be noted that cadmium can penetrate and embrittle titanium alloys under certain circumstances (See Leaflet 407/1).

13.4 PARTS IN CONTACT WITH NON-METALLIC MATERIALS

13.4.1 Fluorinated organic polymers: The possibility of corrosion or stress corrosion of titanium alloys by contact with fluorinated sealants shall be considered in the choice of materials (See Leaflet 407/2).

13.4.2 Phosphate ester hydraulic fluids: Titanium alloys shall not be used in situations where contamination with phosphate ester hydraulic fluid can occur at temperatures (eg, over 120°C) at which the fluid can cause corrosion of the alloy.

14 TREATMENT OF METAL TANKS (INCLUDING INTEGRAL TANKS)

14.1 GENERAL REQUIREMENTS

14.1.1 The exterior surfaces of tanks shall be protected by a treatment appropriate to the metal from which they are made. Interior surfaces shall be treated as described below.

14.2 TANKS FOR AVIATION FUEL (INCLUDING INTEGRAL TANKS)

14.2.1 Aluminium alloy tanks: The inside surfaces shall be anodised in chromic acid or chromate filmed (DEF STAN 03-18) and shall then be painted with a chromated epoxy primer (DTD 5567, Scheme 1) or with an approved (DTD 900) alternative scheme resistant to fuel corrosion and to micro-biological attack. In addition to the surface treatment, an inhibitor cartridge or bag containing calcium chromate to Specification DTD 495 or strontium chromate in an approved cartridge, bag or tablet form shall be provided in tanks not having sumps, (eg, shallow wing tanks or integral tanks). The cartridge shall be installed in a position where free water present in the tank will come into contact with it when the rotorcraft is in the standing position. There shall be no risk of the cartridge or its contents getting into other parts of the system.

14.2.2 Titanium alloy tanks: The inside surfaces need no protective treatment.

14.2.3 Use of copper-base materials: Consideration shall be given to the possibility of adverse effects due to attack by aviation fuel on copper-base materials with consequent degradation of fuel.

14.3 TANKS FOR WATER-METHANOL AND FOR DE-ICING FLUIDS

14.3.1 Tanks made from corrosion resisting steel or titanium need no protection inside. Tanks made of aluminium base material shall be anodised, and the presence of other metallic components may necessitate further internal protection with a suitable paint scheme.

14.4 TANKS FOR DRINKING WATER

14.4.1 Tanks made of corrosion resisting steel and titanium need no protective treatment inside. Those made of aluminium alloy shall be anodised and painted with clear varnish to DTD 5562, the material used being suitable for contact with drinking water.

14.5 TANKS FOR HIGH TEST PEROXIDE (HTP)

14.5.1 The internal surface of HTP tanks shall be treated as described in DEF-60 and 61 as appropriate.

14.6 TANKS FOR AVPIN

14.6.1 Tanks for Avpin shall be made of corrosion-resisting steel containing not less than 5% nickel. They shall be treated only as in Para 9.2.2; paint shall not be applied internally.

15 TREATMENT OF WOOD (See also Chapter 400, Para 6)

15.1 EXTERNAL PLYWOOD SURFACES

15.1.1 The surfaces shall be covered with a suitable fabric (BS-F series) applied with an approved adhesive (DTD 900) according to DTD 912. They shall then be painted with the standard paint scheme appropriate to the adjacent structure.

15.2 INTERNAL SURFACES

15.2.1 After any adhesive bonding is complete, surfaces shall be either painted with the standard interior paint scheme, or varnished (BS 4X17).

15.3 FLOORS

15.3.1 Wooden parts in floors shall be treated with wood preservative chosen by reference to BS 1282. Types OS or WB 1, 2 or 3 are preferred to Type TO. Type WB34 shall not be used. Surfaces shall then be either varnished (BS 4X17) or painted with the standard interior paint scheme.

15.4 ASSEMBLY OF WOOD TO METAL

Note: The impregnants used to preserve and to improve the flame resistance of wood may seriously increase the corrosion of metals with which the wood is in contact.

15.4.1 Except where adhesive bonding is used, all joints between wood and metal shall be wet assembled (as described in Para 23) after the metal surfaces have been painted.

15.4.2 Fasteners used in contact with timber should preferably be made of brass or corrosion-resisting steel. If made of non-corrosion resisting steel, they shall be coated by or zinc (DEF STAN 03-20) electro-plating.

15.4.3 Magnesium base alloys shall not be used in direct contact with wood, but shall be separated by a shim of zinc, aluminium or a suitable non-metallic material.

16 TREATMENT OF FABRIC COVERING OPEN STRUCTURES

16.1 The external fabric covering open structures shall be tautened and protected with dope to BS X26. Any paint scheme used over the doped area shall not significantly affect the tautening characteristics of the dope.

17 NATURAL FIBRES

17.1 Natural fibres, whether in the form of fabrics, tapes, webbing, cordage or threads, shall be protected against rot. Parachute materials shall be treated in accordance with Specification DTD 928. Other materials shall be treated in accordance with BS 2087 by one of the following processes:

(i) Pentachlorphenyl laurate process (aqueous emulsion type) - this is generally applicable and is colourless and odourless.

(ii) Chrome copper process (with stitch finish) - this is applicable to cellulose threads but should not be used where there is contact with rubber or aluminium or magnesium alloys. It is light green in colour and odourless.

(iii) Mineral khaki process followed by (i) above (normal process with stitch finish) - this is applicable to cellulose fabrics, tapes and webbing. It is khaki in colour and odourless.

Note: Information is given in Leaflet 407/4 on the deleterious effects of light and heat on fibre properties.

17.2 Polymer coated fabrics. Where appropriate to the end use the coating shall contain additives which provide the material with resistance to degradation.

18 SYNTHETIC FIBRE MATERIALS

18.1 Synthetic fibre materials are resistant to rotting but dressings, added in the course of manufacture should be removed.

18.2 When synthetic fibre materials are intended for applications in which they may receive much exposure to light, fibre of the 'bright' type shall be used, ie, fibre which is substantially free from delustering pigment. (Information is given in Leaflet 407/4 on avoidance of degradation due to light, heat and abrasion).

19 PLASTIC MOULDED MATERIALS

19.1 No additional finish is normally required.

20 TREATMENT OF SYNTHETIC RESIN COMPOSITES

20.1 Internal surfaces may be left untreated, except where painting is required to attain a desired colour.

Note: The fire retardant properties of thin gauge laminates may be compromised by the application of some paint schemes.

20.2 External surfaces shall normally be painted with an appropriate approved scheme preceded by an appropriate pretreatment.

Note: Some methods of surface abrasion and chemical paint removers may cause severe damage to some types of composite materials, particularly those containing carbon fibres. For this reason expert guidance should be sought on suitable surface treatments and paint schemes.

20.3 Where surfaces of composites containing carbon fibres are in contact with metals, special attention shall be given to design (See Para 5) and to assembly (See Para 23), to prevent galvanic corrosion. In particular, direct contact of carbon fibre composite with magnesium alloy, aluminium alloy, non-corrosion resisting steel, cadmium and zinc surfaces shall be avoided either by using adhesive bonding, techniques to electrically insulate the composite and metallic components or by painting the carbon fibre composite component before it is wet assembled (See Para 23) to the metallic component which should be protected to the normal pre-assembly requirements (See Paras 8, 10 and 11).

Note: Carbon fibre behaves as a noble metal in its galvanic corrosion effects on metals, that is, its effect will be similar to that of gold etc, in Column 1 of Table 1 of Leaflet 407/3. In particular, it will give D class contacts when in contact with aluminium base and magnesium base alloys.

21 TREATMENT OF RADOMES

21.1 The general considerations of Para 20 apply but due attention shall be paid to the possibility of certain materials, (eg, pigments), affecting radar transparency (DTD 926, DTD 856, BS 2076).

22 BATTERY STOWAGES

22.1 Surfaces that may be affected by battery electrolyte, including by spray, shall be given the appropriate surface treatment and then painted to DTD 5567 or DTD 5580. If mechanical damage to the paint coating is expected, consideration should be given to providing suitable supplementary non-metallic protection.

23 PRECAUTIONS AND TREATMENTS DURING ASSEMBLY

23.1 FIELD OF APPLICATION

23.1.1 These requirements apply to all static joints and interfaces; that is, all contacting surfaces whether between similar or dissimilar materials (one or both of which is a metal) that are not intended to move relative to one another in service. The requirements apply also to static joints that will be parted from time to time, (eg, inspection covers).

23.2 WET ASSEMBLY

23.2.1 Except as detailed in Para 23.4 all static joints shall be wet assembled with an approved sealant or an approved jointing compound (See Para 23.3).

23.2.2 Sealants: These shall be applied to the mating surfaces so as to completely fill any crevice and produce at the edge of the joint a specific fillet which shall not be subsequently removed. Where fasteners form part of the assembly, the sealant shall be present down the shank and under the head and tail.

23.2.3 Jointing compounds: These shall be applied to the mating surfaces so as to completely fill any crevice, but excess material at the edge of the joint should be largely removed by wiping prior to painting. Where fasteners form part of the assembly, compounds shall be applied to the hole and/or the shank and thread so that in the assembled joint the compound completely fills any space under the head and tail of the fastener.

23.3 MATERIALS APPROVED FOR WET ASSEMBLY

23.3.1 Sealants curing to produce an elastomer which adheres to one, or preferably both, surfaces (DTD 900). This type of crevice filler should be used for fuel tanks, cabin skins and pressure capsules.

Note: Possible dangers to titanium alloys from fluorinated sealants when at elevated temperatures are described in Leaflet 407/2, and it is recommended that the Rotorcraft Servicing Manual calls for titanium parts in contact with Viton sealants at temperatures of 200°C and above to be subject to spot inspection for incipient cracking after every 1000 hours at temperature.

23.3.2 Jointing compounds:

(i) Pigmented jointing compound non-hardening type (DTD 5604, DTD 900).

(ii) Pigmented varnish jointing compound, (DTD 369).

(iii) Alternatives to (ii) above (DTD 900).

23.4 EXCEPTIONS AND SPECIAL CASES

23.4.1 Spot and seam welded joints: All spot and seam welds shall be made through a coating of approved (DTD 900) weld-through primer, sealant, or adhesive. The coating shall be applied to one or both of the surfaces, after suitable cleaning in such a way as to fill the crevice of the welded joint completely.

23.4.2 Joints made with an adhesive: Jointing compound shall be omitted from adhesively bonded areas.

23.4.3 Screwed unions in liquid and gaseous systems: Jointing compound other than those specifically designed for the purpose shall be omitted from unions in oxygen systems and may be omitted from unions in other systems.

23.4.4 Lubricated joints: Lubricated joints, including those with grooves or holes for oiling, shall be assembled with the lubricant to be used in service, or with an inhibited oil or grease compatible with the service lubricant.

23.4.5 Joints where anti-fretting treatments are required: Anti-fretting treatments (DEF STAN 91-19 and DTD 900) may be used in place of wet assembly materials.

23.5 NON-METALLIC SHIMMING AND PACKING MATERIALS

23.5.1 Such materials shall be non-corrosive, effectively non-absorbent, and electrically insulating (BS 2848, 3964, 6746). Wet assembly while not normally required, may be used to eliminate potential crevices.

23.6 METAL SHIMS

23.6.1 Sacrificial metal shims used in joints between dissimilar metals shall be wet assembled (See Para 23.2).

24 TREATMENT AFTER ASSEMBLY: TOUCHING UP

24.1 GENERAL REQUIREMENTS

24.1.1 Requirements for painting before and after assembly have been given in Para 7. Before final painting, all areas shall be inspected for the integrity of any protective treatment previously applied. Areas where the protective treatment is not intact, due for example to damage or to deliberate removal for electrical bonding, shall be treated as described below:

24.2 TOUCH UP OF SURFACE TREATMENTS

24.2.1 Surface treatments shall be repaired by the process originally applied except that, where there is a possibility that any solution used may be trapped and cause subsequent corrosion, any other process appropriate to the part and not giving rise to this danger may be used.

24.2.2 Non-corrosion-resisting steels: Brush plating of cadmium to meet the requirements of DEF STAN 03-19 is particularly suitable. Baking treatments (DEF STAN 03-19 and DEF STAN 03-4) may be omitted on steels of less than 1800 MPa max UTS by agreement with the Design Authority.

24.2.3 Aluminium and its alloys: Parts that have been anodised may be touched up by a chromate filming treatment (DEF STAN 03-18) applied by brushing, or by etch priming (DEF STAN 80-15 and approved proprietary alternatives) provided that etch primer is acceptable (See Para 7.5.1).

24.2.4 Magnesium alloys. Chromate films shall be touched up as described in DTD 911. Surface sealing (DTD 5562) may be touched up with room temperature curing chromated epoxy primer (DTD 5567). The selenious acid method is not permitted.

24.2.5 Paint films: The original primer and/or finish shall be used for touching up room temperature cured paints. Stoved paint schemes may be touched up with room temperature cured paints (See DEF STAN 03-7).

25 SPARE PARTS

Note: This section applies to spare parts and should be considered for parts placed in long term in-work storage.

25.1 GENERAL REQUIREMENTS

25.1.1 Spare parts should preferably be supplied fully protected according to the appropriate sections of this chapter. Where this is not possible, they shall be treated in accordance with the following paragraphs.

25.1.2 Parts supplied in the primed condition should preferably be protected from contact with oils, greases and other materials which would interfere with subsequent treatment. Otherwise, any such contamination shall be thoroughly removed prior to over-coating (DEF STAN 03-7)

25.1.3 Temporary protectives shall be appropriate approved compounds (Corrosion Manual AP119A-0200-1D Table 1). For long term and tropical storage, the requirements of DEF-1234 shall be observed. The requirements do not apply if maintained desiccated packs are used.

25.2 NON-CORROSION-RESISTING STEEL PARTS (See Para 8).

25.2.1 Plated parts shall be coated with temporary protective except where the storage is known to be in warm indoor conditions.

25.2.2 Parts metallised with aluminium should not be treated with temporary protective.

25.2.3 Parts phosphated as a pretreatment before painting shall always be supplied with at least a coat of the primer of the specified paint scheme, and shall not be coated directly with temporary protective.

25.2.4 Lubricated parts, including those phosphated before lubrication, shall be supplied in the lubricated condition or coated with temporary protective. It shall be stated whether the temporary protective must be removed immediately prior to service, or whether it is compatible with the service lubricant.

25.3 ALUMINIUM AND ALUMINIUM ALLOY PARTS (See Para 10)

25.3.1 Parts that have been anodised or chromate filmed before painting shall be supplied with at least a coat of the primer of the specified paint scheme. Temporary protectives may be used on parts that are not to be painted.

25.3.2 Parts which are not required by Para 10 to be anodised or chromate filmed shall be either etch primed (See Para 7.5.1) or coated with temporary protective.

25.3.3 Parts metallised with aluminium or aluminium-zinc alloy should normally be supplied with at least one coat of the primer of the specified paint scheme. If supplied bare, they shall not be coated with temporary protective. Parts metallised with zinc shall be etch primed (See Para 7.5.1) and preferably then coated with the primer appropriate to the specified paint scheme.

25.4 MAGNESIUM ALLOY PARTS (See Para 11)

25.4.1 Parts shall be supplied in the fully protected condition, with the possible exception of the final finishing paint coat.

25.5 CRATING AND STORAGE

25.5.1 Packing and crating for storage and shipment shall be carried out with due regard to the hazards of transport, the climate and environment to be withstood and to the dangers of accelerated corrosion caused by stagnant conditions inside packs or inside components. Guidance should be sought from DEF-1234 and DEF STAN 03-13.

LEAFLET 407/0

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

REFERENCE PAGE

Note: See also list of references given in Leaflet 407/4.

AGARD Publications

AGARD-LS-84	The theory, significance and prevention of corrosion in aircraft (1976)
x x x x x x	Aircraft Corrosion*

Defence Specifications

DG-8	Schedule of protective finishes, of which PART TWO is 'Advice to designers on corrosion and its prevention'

D Mat Reports

165	Mechanical properties of paint films in relation to their use on aircraft
169	Factors influencing the performance of corrosion inhibited (paint) systems
170	Evaluation of the requirements for the approval of weld through primers
174	The evaluation of anti-fretting compounds for the prevention of fretting fatigue of aluminium alloys at temperatures up to l50°C.
175	Factors affecting the electrode position of paint on aircraft alloys
179	The corrosion properties of airframe contaminants
191	Stress corrosion of titanium alloys by Viton B (see also Leaflet 407/2)
192	Evaluation of DOW No. 17 treatment for magnesium alloys
193	Impregnation of anodic films for the protection of magnesium alloys

* To be issued

D Mat Reports (continued)

196	Evaluation of high temperature resisting coatings for the protection of magnesium alloys

RAE Technical Reports

65073	The effect of environment on fatigue crack propagation Part I
65158	Corrosion fatigue and stress corrosion cracking of an aluminium-5% magnesium-4% zinc alloy totally immersed in 3% NaCl and other corrodents
65219	The corrosion of steel in contact with molybdenum disulphide
66167	An assessment of the rain erosion resistance of infra-red transmitting materials and components
66168	A study of size effects in the plating embrittlement of high strength steels
66176	Hydrogen permeation measurements on some electro-deposited protective coatings for high strength steels
67019	Multiple impact rain erosion studies at velocities up to 450 m/s (M 1.3)
67042	The water absorption of an unsaturated polyester resin and glass fibre/resin composites
68041	The effect of environment on fatigue crack propagation. 2 Measurements on aluminium alloys at different frequencies
68053	Vacuum deposition of adherent cadmium layers on steel
69133	Rain erosion Part VII. An assessment of various protective coatings
69198	Rain erosion Part VIII. An assessment of miscellaneous materials and composites
69212	Zinc-nickel alloy electroplating for the protection of high strength steels
69253	Rain erosion Part IX. An assessment of inorganic non-metallic materials
70081	The effect of outdoor exposure on stressed and unstressed adhesive bonded metal to metal joints. Trial I Part 1 - Two year summary

<u>RAE Technical Reports (continued)</u>

71244	The removal of hydrogen from cadmium plated steel
71247	The electrochemical characteristics of carbon fibre reinforced plastic aluminium alloy couples
72027	An evaluation of the Dingley and Bednar cadmium plating bath
72160	The resistance to corrosion of some insulating carbon fibre reinforced plastic to aluminium alloy joints
73016	Effect of outdoor exposure on stressed and unstressed adhesive bonded metal-to-metal joints. Trial 1. Part 2 - Four year summary
73035	Corrosion risks at contacts between carbon fibre reinforced plastics and metals
74005	Factors affecting the elimination of hydrogen fluoride from a vinylidene fluoride/hexafluoropropene/ tetrafluoroethylene terpolymer
74096	The resistance of carbon fibre reinforced plastics to certain aviation fluids
74141	Cadmium plating by the high current density (Cleveland) process and the diffusion of hydrogen through 300M steel
74153	A study of the stress corrosion cracking of three aluminium plate alloys using a variety of tests
75104	Polysulphide sealants
75145	The effect of water on carbon fibre composites Part V - The dependence of the interlaminar shear strength of epoxy and vinyl ester composites on temperature, humidity and total immersion in water
76031	Review of textile materials research in Materials Department, RAE 1968-1975
76063	The corrosion resistance and paint adhesion properties of chromate conversion coatings on aluminium and its alloys
76144	Thermal degradation of polyepichlorohydrin elastomers
77031	The breaking strength and extension of weathered rubber-coated fabrics

RAE Technical Reports (continued)

78114	The axial loading corrosion fatigue properties of 25mm thick high and low strength X116 plate (DTD 5120 and DTD 5130)
79078	Evaluation of the corrosivity of paint removers toward aluminium alloys used in aircraft structures
79092	The axial loading air and corrosion fatigue properties of BSL 93, BSL 95, and BSL 97 aluminium alloys
80137	Metallurgical aspects of fatigue cracking at fastener holes

Air Publications

AP 119A-0200-1A to 1D	Corrosion manual

British Standards

PD 6484	Commentary on corrosion at bi-metallic contacts and its alleviation

LEAFLET 407/1

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

THE PENETRATION OF TITANIUM ALLOYS BY SOLID CADMIUM

1 INTRODUCTION

1.1 Cadmium can penetrate the surface of titanium alloys and embrittle them at temperatures as low as ambient. This effect can occur equally from cadmium plated on to titanium or from cadmium on steel parts with which titanium is in contact.

1.2 The conditions under which penetration occurs, though not fully established, appear to be: (a) tensile stresses in the surface of the titanium alloy, and (b) surface pressure of the cadmium, perhaps necessarily accompanied by sufficient strain of the titanium to break the natural oxide film (present even under electrodeposited cadmium) and so allow actual lattice contact of the two metals. Penetration has been found to occur more readily in a vacuum.

1.3 Penetration was first observed in the USA during environmental tests on structures, and failures in service caused by embrittlement due to penetration have subsequently been reported. Cadmium alloy fasteners have been used on British rotorcraft for many years and although no failures have occurred, evidence of penetration has been found.

1.4 It seems certain that liability to penetration increases with increase of temperature, and it is likely that, given tensile stress, pressure and strain, penetration is a real hazard at temperatures below, perhaps considerably below, the 250°C hitherto accepted as the safe limit for titanium in contact with cadmium.

2 RECOMMENDATIONS

2.1 It is recommended that contact between cadmium and titanium should be avoided wherever the following circumstances exist or are likely to arise:

(i) Contacts containing local areas under high assembly stresses and fasteners or press fits in which stress is not uniformly distributed. This applies equally to cadmium plated steel fitments in titanium.

(ii) Contacts where the fatigue properties of the titanium are critical so that the notch effect of even slight penetration would be dangerous.

(iii) Contacts which will operate at temperatures higher than those at which experience has hitherto shown similar contacts to be safe.

(iv) Contacts which will operate in a vacuum.

2.2 Titanium alloys should not be coated with cadmium by vacuum deposition because of the danger that this method may be too effective in giving that intimate lattice-to-lattice contact that, in most other situations, would be welcomed for its promotion of high adhesion.

LEAFLET 407/2

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

THE STRESS CORROSION OF TITANIUM ALLOYS BY FLUORINATED SEALANTS AT ELEVATED TEMPERATURES

1 Results of laboratory tests carried out in the USA have shown that fluorinated sealants can cause stress corrosion cracking of titanium alloys at elevated temperatures. An investigation into this effect has been made in UK using a two-part mix of Viton B sealant PR 1720F applied directly to the metal.

2 With Ti-6A1-4V alloy sheet stressed at 85% and 95% of the 0.1% proof stress, stress corrosion cracking occurred at 250°C but not at 200°C during continuous exposure lasting 1000 hours. Under extreme conditions of plastic deformation and elastic strain, minor superficial cracks, up to 0.015 mm (0.0006 in) deep, developed during a test lasting 1000 hours at 200°C; no cracks formed under these conditions at 150°C.

3 With material that possessed directional properties, specimens that were stressed longitudinally, (i.e. in the direction in which sheet had been rolled), were appreciably less susceptible to cracking than specimens that were stressed transversely. Susceptibility to cracking was somewhat less in static air than in flowing argon.

4 Under the conditions in which Ti-6A1-4V alloy was susceptible to cracking, alloy Ti-8A1-1Mo-1V was somewhat more susceptible and alloy Ti-5A1-2½ Sn somewhat less.

5 The relevance of these results to behaviour in service depends on the temperature at which Viton B is likely to be used in service. It is assumed that Viton sealants will not be used on fuel tanks at temperatures above 160°C. In the fire zone region it is assumed that the sealants would not used continuously at temperatures above 225°C and that short surges of temperature will not exceed 250°C.

6 It is concluded that Ti-6A1-4V alloy sheet is not susceptible to stress corrosion cracking from contact with Viton B at temperatures up to l50°C. At 200°C under extreme conditions of tensile loading there is evidence of incipient cracking and under these conditions at 250°C some stress corrosion hazard exists. The hazard is greater in material stressed in the transverse direction. However, it is likely that the hazard will be less if the Viton and metal are separated by a coat of primer, and directional stress corrosion effects are likely to be unimportant with metal that has no pronounced directional strength properties.

LEAFLET 407/3

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

AVOIDANCE OF GALVANIC CORROSION AT BIMETALLIC CONTACTS

1 INTRODUCTION

1.1 This leaflet gives recommendations on the avoidance of corrosion due to galvanic action between dissimilar metals in the presence of moisture.

1.2 The leaflet should be read in conjunction with BS PD6484 "Commentary on corrosion at bimetallic contacts and its alleviation". The recommendations are based on service and laboratory experience, and are more reliable than those derived from, tables of potentials in seawater.

2 GENERAL

2.1 Galvanic action can occur when two metals in electrical contact are also bridged by an electrolyte, (e.g. condensation, rain, seawater). As a result, the less noble metal is corroded and the more noble metal protected (for an exception see Note (d) in Table 1). Although the potential difference between metals is the prime driving force of the corroding current, the magnitude of the potential is not always a reliable guide to the amount of corrosion suffered in any particular contact.

2.2 Table 1, taken from the HMSO Publication "Corrosion and its prevention at bimetallic contacts" (1956), records the degree of galvanic corrosion which can be expected when any two metals in contact are bridged by an electrolyte. Although more extensive tables appear in BS PD6484, experience indicates that Table 1 by itself is a very useful guide in the context of aircraft structures, but it should not be regarded as infallible.

2.3 Table 1 divides galvanic effects into four classes A to D. The letters record only the acceleration of corrosion by galvanic attack and do not relate to the intrinsic corrosion resistance of any one metal. The letters have the following meanings:

A The corrosion of the first metal is not increased by the second metal.

B The corrosion of the first metal may be slightly increased by the second metal.

C The corrosion of the first metal may be markedly increased by the second metal.

D The corrosion of the first metal may be very seriously increased by the second metal.

Where two symbols are given, the corrosion varies markedly with the conditions. Each member of a contacting pair should be regarded in turn as the first metal, and the degree of likely attack by the other metal read from the table.

2.4 Electrolytic corrosion can also be caused by stray or earth currents passing from one metal to another through an electrolyte. Attack occurs at the anode metal, i.e. the metal from which the positive current enters the electrolyte, though if the cathode is aluminium this too can be attacked. The effects of stray currents can be reduced by:

(i) providing as widespread and good a metallic conducting path as possible, so that concentrations of current are avoided, and

(ii) wet assembly with sealants or jointing compounds (see Chapter 407, para 23.2) so that electrolytes are excluded from joints.

3 RECOMMENDATIONS

3.1 The class of contact which can be used safely in an assembly depends on the environment and the degree of protection which can be given, in particular, whether wet assembly techniques are used.

3.2 The following recommendations are made:

Environment		**Class of contact**	
		Safe WITH wet assembly	Safe WITHOUT wet assembly
I	Interior parts of hermetically sealed components and equipment (see Note 1).	A to D	A to D
II	Interior parts subject to condensation but not to direct contamination by salt (see Note 2)	A to C	A and B (See Note 3)
III	Parts liable to be wetted with salt water, or normally exposed to the weather.	A and B (See Note 4)	NONE (See Note 3)

Notes: 1 Imperfect containers which can 'breathe' should be regarded as Class II environments.

2 If the condensate forms from an industrial or marine atmosphere it may contain corrosive salts, and should be regarded as tending to a Class III environment.

3 Some metals are susceptible to crevice corrosion, and wet assembly is required for all contacts liable to be wetted by salt water and is recommended for parts subjected to condensation.

4 In places where seawater can collect and remain, and bridge wet assembled joints, only Class A contacts are safe.

3.3 Both electrical and electrolytic paths are required for galvanic corrosion to occur between dissimilar metals or other electrically conducting materials. Consequently electrical insulation of the dissimilar materials will prevent galvanic corrosion, e.g. when carbon fibre reinforced composite material is adhesively bonded to aluminium alloy the latter will not suffer accelerated galvanic corrosion provided that an adequate and complete glue line is achieved.

REFERENCES

No	Title, etc
1	Commentary on corrosion at bimetallic contacts and its alleviation, BS PD6484.
2	Tropical exposure tests on bimetallic couples, RAE Report Mat 82, December 1954.

TABLE 1

DEGREE OF CORROSION AT BIMETALLIC CONTACTS

Where a metal is plated, the behaviour should be sought under that of the plated coating.

Class A. The corrosion of the first metal is not increased by the second metal
Class B. The corrosion of the first metal may be slightly increased by the second metal
Class C. The corrosion of the first metal may be markedly increased by the second metal
Class D. The corrosion of the first metal may be very seriously increased by the second metal

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Second Metal First metal	Gold platinum rhodium silver	Monel Inconel, nickel/ molybdenum alloys	Cupronickels, silver solder aluminium-bronzes tin-bronzes, gunmetals	Copper brasses 'nickel silvers'	Nickel	Tin and soft solders also lead	Steel and cast iron	Cadmium	Zinc	Magnesium and magnesium alloys (chromated)	Stainless steels			Chromium	Titanium	Aluminium and aluminium alloys
											Austenitic 18/8 Cr/Ni	18/2 Cr/Ni	13% Cr			
1 Gold, platinum, rhodium, silver	-	A	A	A	A	A	A	A	A	A	A	A	A	A	A(z)	A
2 Monel, Inconel, nickel/molybdenum alloys	B	-	A	A	A	A	A	A	A	A	A	A	A	A	A	A
3 Cupronickels, silver solder, aluminium-bronzes, tin bronzes, gunmetals	C(f)	B or C	-	A	A	A	A	A	A	A	B or C	B	A	B or C	B or C	A (c)
4 Copper, brasses, 'nickel silvers'	C(f)	B or C	B or C	-	B or C	B or C (k)	A	A	A	A	B or C	B or C	A	B or C	B or C	A (c)
5 Nickel	C	B	A	A	-	A	A	A	A	A	B or C	B or C	A	B or C	B or C	A
6 Tin and soft solder, also lead	C	B or C (m)	B or C	B or C	B	-	A or C	A	A or C	A	B or C	B or C	B or C	B or C	B or C	A
7 Steel and cast iron	C	C	C	C	C(f)	C(f)	-	A (g)	A (g)	A	C	C	C	C(f)	C	B (g)
8 Cadmium	C	C	C	C	C	B(n)	C	-	A	A	C	C	C	C	C(y)	B
9 Zinc	C	C	C	C	C	B	C	B	-	A	C	C	C	C	C	C(e)
10 Magnesium and magnesium alloys (chromated)	D	D	D	D	D	C	D	B or C	B or C	-	C	C	C	C	C	B or C(e)
11) Stainless Steel — Austenitic 18/8 Cr/Ni	A	A	A	A	A	A	A	A	A	A	(x)	A	A	A	A	A
12) Stainless Steel — 18/2Cr/Ni	C	A or C(1)	A or C(1)	A or C(1)	A	A	A	A	A	A	A	(x)	A	A	(j)	A
13) Stainless Steel — 13% Cr	C	C	C	C	B or C	A	A	A	A	A	C	C	(x)	C	C	A
14 Chromium	A	A	A	A	A	A	A	A	A	A	A	A	A	-	A	A
15 Titanium	A (z)	A	A	A	A	A	A	D (y)	A	A	A	A	A	A	-	A
16 Aluminium and aluminium alloys (h)	D	C	D(c)	D(c)	C(f)	B or C	B or C	A	A	A (d)	B or C	B or C	B or C	B or C(b)	C	(h) (x)

/See over for "NOTES"

NOTES TO TABLE 1:

(a) Where contact between magnesium and aluminium alloys is necessary, the use of aluminium alloys with low or negligible copper and iron content is preferred.

(b) In contact with thin (decorative) chromium plate, the symbol is C, but with thick plating (as used for wear resistance) the symbol is B.

(c) When contacts between copper or copper-rich materials and aluminium alloys cannot be avoided, a much higher degree of protection against corrosion is obtained by first plating the copper-rich material with tin or nickel and then with cadmium, than by applying a coating of cadmium of similar thickness. The aluminium in contact with the copper-rich material should be anodised when practicable.

(d) When magnesium corrodes in sea-water or certain other electrolytes, alkali formed at the aluminium cathode may attack the aluminium.

(e) When it is not practicable to use other more suitable methods of protection, (e.g. spraying with aluminium), zinc may be useful for the protection of steel in contact with aluminium, despite the accelerated attack upon the coating.

(f) This statement should not necessarily discourage the use of the second metal as a coating for the first metal provided that continuity is good; under abrasive conditions, however, even a good coating may become discontinuous.

(g) In these cases the second metal may provide an excellent protective coating for the first metal, the latter usually being electrochemically protected at gaps in the coating.

(h) When aluminium is alloyed with appreciable amounts of copper it becomes more noble and when alloyed with appreciable amounts of zinc it becomes less noble. These remarks apply to bimetallic contacts and not to the inherent corrosion resistance of the individual aluminium alloy. Such effects are mainly of interest when the aluminium alloys are connected with each other.

(j) No data available.

(k) In some immersed conditions, the corrosion of copper or brass may be seriously accelerated at pores or defects in tin coatings.

(l) Serious acceleration of corrosion of 18/2 stainless steel in contact with copper or nickel alloys may occur at crevices where the oxygen supply is low.

(m) Normally the corrosion of lead-tin soldered seams is not significantly increased by their contact with the nickel-base alloys but under a few immersed conditions the seams may suffer enhanced corrosion.

(n) Tin should not be used in contact with cadmium in joints liable to be heated above 120°C.

(x) Joints liable to crevice corrosion when the oxygen supply is limited.

(y) Under some circumstances cadmium can penetrate titanium alloy and embrittle it, a warning of the danger is given in Leaflet 407/1.

(z) There is evidence that at elevated temperatures in certain atmospheres (e.g. exhaust gases), silver coatings may cause cracking of stressed titanium alloy parts.

LEAFLET 407/4

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

DETERIORATION OF FIBROUS MATERIALS

1 INTRODUCTION

1.1 This leaflet gives information on precautions to minimise deterioration of textiles and cordages.

1.2 A general account of the appearance of damaged aerospace textiles is given in RAE Technical Memorandum Materials 290 (Ref 1).

2 RESISTANCE TO ROTTING

2.1 Synthetic fibre materials are resistant to rotting, (i.e. to attack by micro-organisms), and require no protective treatment. Some microbiological attack, generally harmless in itself, though possibly unacceptable aesthetically, may occur on surface finishes applied or where the material is contaminated.

2.2 Natural fibre materials are susceptible to rotting and require protective treatment in accordance with Chapter 407, para 17.1.

3 RESISTANCE TO ACTINIC DEGRADATION

3.1 Textile fibres are susceptible to degradation and weakening by actinic attack. This needs to be borne in mind when the use or the position of the component in the rotorcraft results in much exposure of the material to light.

3.2 D Materials Technical "Memorandum No 8 (see Ref 2) describes results of continuous exposure tests of nylon, Terylene and flax webbings under tropical and temperate conditions. Considerable losses of strength occurred within an exposure period of six months. For example, webbings made from 'bright' nylon, 'bright' Terylene (see para 3.6 for definition of 'bright') and flax yarns, exposed for six months in Australia, lost 53%, 48% and 41% respectively of breaking strength when exposed to direct light and 32%, 27% and 25% respectively when exposed behind 'Perspex'.

3.3 A review of the literature on the weathering of nylon is given in RAE Technical Note Chem 1389 (see Ref 3); results of weathering trials of nylon and the assessment of some protective treatments is described in RAE Technical Report 64081 (see Ref 4). It was found that 2:4 dihydroxybenzophenone applied from solution in benzyl alcohol or methylated spirit provided some protection. For extra high tenacity nylon, 90% of the original strength was retained after 312 days exposure, and 70% after 609 days. Corresponding figures for the untreated material were 65% and 31%. For delustered nylon, the strength retention of treated fibres was 90% after 123 days, whilst untreated fibres retained only 32%. Acrylic fibre materials are more resistant to sunlight than most other fibres, but repeated immersions in water cause strength losses (see RAE Technical Report 77156, Ref 5).

3.4 Where it could be applied, (e.g. in certain types of rope), neoprene sleeving resulted in better than 90% strength retention after 5 years. Sleeved ropes after exposure were indistinguishable from new, and they did not snarl or shrink; they were also protected from abrasion.

3.5 Correlations of fading of dyestuffs with strength loss showed that for single dyes on high tenacity nylon threads and webbings, coefficients of up to 0.75 were obtained, and fading constituted a useful non-destructive test for strength (see Ref 6). For mixtures of dyestuffs on extra-high tenacity nylon threads, no similar worthwhile correlation existed (see Ref 7).

3.6 With synthetic fibre materials, the presence in the fibre of delustering pigment accelerates actinic degradation. Fibre of the 'bright' type should preferably be used, i.e., fibre substantially free from delustering pigment. When the application involves much exposure to light, only bright type fibre shall be used (see Chapter 407, para 18.2).

3.7 When dyeing of nylon is required, it is recommended that CI Acid Black 132, Acid Green 43, Disperse Yellow 3, Disperse Orange 3 or Mordant Yellow 34 should be used. Dyes to be avoided because they enhance actinic attack on nylon are CI Acid Red 211, Mordant Blue 49 and Disperse Black 1. A number of vat dyes are known to increase attack on cellulosic fibres (see RAE Technical Report 74179, BS 3FI00, Refs 8, 9). Continuous exposure to fluorescent lighting in a room does not cause noticeable damage, but contact with the lamp must be avoided (see RAE Technical Memorandum Mat 302, Ref 10).

3.8 The use of unsuitable marking ink may cause or accelerate degradation of textiles or polymeric materials; marking ink meeting BS F100 requirements is recommended.

3.9 Coated Textiles may stiffen on weathering. Polyurethane coatings may delaminate and give a tear strength similar to that of the uncoated fabric. The tensile strength of coated fabrics is less affected by weathering than the equivalent uncoated fabrics (see RAE Technical Report 77016, 77031, 78005, Refs 11, 12, 13).

3.10 Certain chloroprene adhesives are susceptible to decomposition under the action of daylight with the formation of acidic products which can cause degradation of fabric, particularly cellulosic fabric, in contact with the adhesive.

4 RESISTANCE TO HEAT

4.1 Textile fibres are subject to degradation and weakening when exposed to elevated temperatures. RAE Technical Note Chem 1270 (see Ref 14) describes the results of tensile tests on nylon and Terylene yarns before and after ageing at temperatures up to 180°C. After 16 hours exposure in air at 150°C the room temperature strength of the nylon yarn was reduced by approximately 40%. The same duration of exposure of the Terylene yarn in air at 156°C reduced the room temperature strength by approximately 10%.

4.2 Extra-high tenacity nylon yarn of improved heat resistance showed approximately 5% loss of room temperature tensile strength after exposure in air at 150°C for 24 hours, and approximately 35% loss after exposure in air for the same period at 175°C (see RAE Technical Report 75113, Ref 15).

4.3 When nylon is introduced suddenly into a hot atmosphere, its temperature takes time to rise, and useful performance may be obtained during this time. For example the energy from a shock load may be absorbed before excessive physical deterioration has taken place. The temperature history in cordage at several load levels when exposed to ambient temperatures up to 340°C is described in RAE Technical Note CPM 7 (see Ref 16). The lifetimes, i.e. times for which a cord will bear loads without breaking, at temperatures up to 440°C are also reported. For example, in a 3.56kN (800 lbf) braided cord under loads of 0, 0.45, 1.78, 2.23 kN(0, 180, 400, 500 lbf), 50 second lifetimes could be obtained when exposed to air temperatures of 380, 360, 310, 210°C respectively; for lifetimes of 500 s the corresponding temperatures were 300, 250, 210, 150°C .

4.4 For circularly woven acrylic and aromatic polyamide cordages of similar construction to nylon of specification minimum strength of 6.68 kN (1500 lbf) (see Ref 17) aromatic polyamide was superior to nylon at low loads or high ambient temperatures, but there was little difference at temperatures below 200°C under a high load of 3 kN (675 lbf). Aromatic polyamide was superior to acrylic under all the temperatures and loading conditions studied. Nylon was superior to acrylic except under zero load, when acrylic was more resistant to heat.

5 RESISTANCE TO ABRASION

5.1 Fibrous materials are susceptible to abrasion. Studies have been made (see Refs 18, 19, 20 and 21) of the abrasion which occurs when synthetic fibre cordage runs at high speed over surfaces of nylon, concrete and asphalt.

5.2 For a nylon surface, it was found that the abrasion mechanism was largely controlled by the rate of heat production in comparison with the rate of heat loss. It is essential to avoid the use of synthetic materials such as nylon when high temperatures may be developed, unless a heat sink or suitable protective layer can be incorporated.

5.3 With concrete, heating plays a less important role, damage being principally a matter of progressive breakage of filaments by the rough surface. For example for a 3.56 kN (800 lbf) cord under a load of 5N (1.125 lbf) a velocity of 10 m/s (32 ft/s) for nylon and polyester, and of 5 m/s (16 ft/s) for polypropylene are sufficient to cause failures in a few seconds. Higher velocities can be tolerated on asphalt than on concrete.

5.4 When a figure of eight descendeur is used for abseiling, descent speeds should be limited to 2 m/s at 100 kg, or 1.3 m/s at 150 kg, so as to avoid damage to the rope by frictional heating (see RAE Technical Report 76049, Ref 22).

6 RESISTANCE TO WATER

6.1 The strength of some textile materials is affected by water. For example, wet nylon is 15% weaker than at standard humidity, and aromatic polyamide 20% weaker. Natural cellulosic fibre (cotton, flax) are a few per cent stronger, while Terylene and polypropylene are unaffected.

6.2 Undrawn nylon degrades chemically in warm water unless protected, e.g. by oxine (see Ref 23).

6.3 Ply-tear webbings which are satisfactory in dry conditions may become dangerous when wet because of ply failure; use of a binder/warp strength ratio of not more than 0.08 is recommended (see Ref 24).

6.4 Immersion of nylon webbing in sea water can cause reductions of a few per cent in strength, even after rinsing, due to retention of deliquescing substances contained in sea water which are inadequately washed out: these raise the local relative humidity, with effects as described in para 6.1. Sodium chloride crystals themselves do not cause noticeable deterioration (see RAE Technical Report 78043, Ref 25).

7 DETERIORATION BY JOINING

7.1 Joining of fabrics, cordage etc changes the properties compared with the unjoined materials.

7.2 When joins are made by knotting, strength losses of up to 50% can be expected. To join two ends, the blood knot or reverse figure of eight are recommended. To form non-slipping hitches, the bowlines are good knots (see RAE Technical Memorandum 251, Ref 26).

7.3 When joins are made in webbing by stitching, enough stitches should be used to ensure that the stitching does not break under load; the webbing itself should fail at the join, with a strength reduction of 10 to 20% of the unsewn material (see RAE Technical Report 74183, Ref 27).

REFERENCES

No Title, etc

1 Damage to nylon textiles in aerospace use.
RAE Technical Memorandum Mat 290, March 1978

2 Fibrous nylon in aeronautical equipments - Resistance to actinic degradation. D Mat Technical Memoranda No 8, January 1961

3 The deterioration of nylon by natural weathering: a literature survey. RAE Technical Note Chem 1389, February 1962

4 Weathering of fibrous nylon and assessment of some protective treatments. RAE Technical Report 64081, December 1964

5 Strength and other properties of yarns extracted from cordage exposed to weather or immersion in water.
RAE Technical Report 77156, October 1977

REFERENCES (contd)

No	Title, etc
6	Correlations of strength losses on weathering, strength losses under a xenon are and fading of dyed textiles. RAE Technical Memorandum 50, January 1969
7	Correlations of strength losses and fading of dyed extra-high tenacity nylon threads on weathering. RAE Technical Memorandum 98, September 1970
8	The effects of dyes and finishes on the weathering of nylon textiles. RAE Technical Report 74179, January 1975
9	Inspection and testing of textiles. British Standard 3F100, February 1975
10	Effects of fluorescent lighting on nylon and aramid cordage. RAE Technical Memorandum Mat 302, September 1978
11	The flexibility of weathered rubber-coated fabrics. RAE Technical Report 77016, January 1977
12	The breaking strength and extension of weathered rubber-coated fabrics. RAE Technical Report 77031, March 1977
13	The tearing of weathered rubber-coated fabrics. RAE Technical Report 78005, January 1978
14	The effect of heat on the strength of nylon and Terylene yarns. RAE Technical Note Chem 1270, December 1955
15	Effects of time and temperature on a heat-resistant nylon yarn. RAE Technical Report 75113, October 1975
16	Load capacity and heat transmission for nylon cordage thrust into a hot environment. RAE Technical Note CPM7, February 1963
17	Lifetimes of cordages when loaded and thrust into a hot environment. RAE Technical Memorandum Mat 76, February 1970
18	High speed abrasion of nylon cordage on nylon. RAE Technical Note CPA183, August 1964
19	High speed abrasion of textile cordage on concrete. RAE Technical Report 65227, October 1965
20	High speed abrasion of textile cordage on an asphalt surface. RAE Technical Report 67023, January 1967

REFERENCES (contd)

No	Title, etc
21	High speed abrasion of nylon cordage on concrete and resin coated concrete, RAE Technical Memorandum Mat 40, September 1968
22	The heating of abseil ropes by friction. RAE Technical Report 76049, April 1976
23	Wet oxidation of undrawn nylon 66 and model amides. RAE Report Chem 531, May 1962
24	The energy absorbing efficiency of ply-tear webbings of various constructions. RAE Technical Report 72051, June 1972
25	The effect of salt and sea water contamination on the mechanical properties of cord and webbing. RAE Technical Report 78043, April 1978
26	The strength of a knotted braided nylon cord. RAE Technical Memorandum Mat 251, June 1976
27	Some observations on the behaviour of superimposed and lap sewn joints. RAE Technical Report 74183, February 1975

LEAFLET 407/5

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

TOILET AND GALLEY INSTALLATIONS

1 INTRODUCTION

1.1 This leaflet gives recommendations for the design features of toilet and galley installations to avoid corrosion of the surrounding structure.

2 TOILET INSTALLATIONS

2.1 The installation should be fabricated and where possible installed as a self-contained unit which is in contact with the rotorcraft structure only at its points of attachment. At these points the best available protective materials and sealants should be used. The unit should be easily removable from the structure.

2.2 The complete unit should stand on a one-piece non-metallic tray which should also enclose the drainage and replenishment pipes. The edges of the tray should extend upwards, except at the entrance which should be well away from any potential spillage, and should be well sealed to the side walls of the compartment. The unit should be designed to be easily cleaned with no sharp corners, or, water or dirt traps. It is recommended that the maximum use of glass fibre reinforced plastic or plastic materials should be made.

2.3 The floor and under floor tray should have adequate drain paths to ensure that any fluid that might be spilt in the compartment would quickly reach the drain point.

2.4 Both the compartment and the air space around it should be adequately vented.

2.5 The necessary services (e.g. water supplies), drainage and venting should be contained within the installation and should also be non-metallic. Only electric power services should be supplied from the rotorcraft system. Water replenishing points should be readily accessible from the outside of the rotorcraft; they should be located as close as possible to the toilet or galley installations which they serve.

3 GALLEY INSTALLATIONS

3.1 The design features of para 2 should be incorporated in the design of galley installations where applicable.

3.2 There should be no leak paths from working surfaces and sink units.

LEAFLET 407/6

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

THE SELECTION AND USE OF CORROSION RESISTING STEELS

1 INTRODUCTION

1.1 This leaflet gives recommendations on the selection and use of corrosion resisting steels in order to optimise their performance in corrosive environments.

1.2 The corrosive environments considered are those normally encountered in rotorcraft, e.g. heavy condensation, salt spray etc. Selection of materials for contact with more specialised environments, e.g. containers and pipework for aggressive fluids, are outside the scope of this leaflet.

2 CHARACTERISTICS OF CORROSION RESISTING STEELS

2.1 The corrosion resistance of "stainless" steels is attributed to the presence of a thin oxide film on the surface of the metal. This film protects the metal from corrosion, provided conditions are such that the film is maintained intact, under which conditions the metal is said to be passive. The range of passivity may be broad or narrow, depending on the composition of the steel and the conditions to which it is exposed, and passivity may be destroyed by changes in conditions, e.g. contact with deoxygenated water which can occur in crevices especially in the presence of chloride ion which is a common contaminant in aircraft structures. When passivity is destroyed, the material corrodes in much the same way as non stainless steel.

2.2 The most significant alloying element in developing passivity in steels is chromium, and a minimum of 10% of that element is required to provide a continuous oxide film which will provide basic protection. Other alloying elements also have a significant effect in modifying the corrosion resistance under specific conditions and also in determining the mechanical properties of the steel.

2.3 Corrosion resisting steels can be classified into three basic types as determined by chemical composition and its affect on their metallurgical structure.

2.3.1 Austenitic (and semi-austenitic) steels: These grades have a high chromium content (above 17%) together with nickel and in some cases other elements which confer a high degree of corrosion resistance. The standard grades do not respond to heat treatment and can only be hardened by cold working. There are a number of special purpose and proprietary materials which offer exceptional combinations of mechanical properties and corrosion resistance at high cost and/or limited availability.

2.3.2 Precipitation hardening steels: These steels contain chromium and nickel together with other alloying elements which enable them to be heat treated to give high strength. Their corrosion resistance is generally intermediate between the martensitic and austenitic grades, but where the higher tensile strength materials are used, stress-corrosion cracking may be a problem (see Leaflet 406/1).

2.3.3 Ferritic and Martensitic steels: These contain chromium as the principal alloying element. Their corrosion resistance increases progressively with increasing chromium content, and in some grades is further improved by smaller amounts of other elements. The optimum corrosion resistance in this category is achieved in the steel containing 18% chromium and 2% nickel (S80); this is not available in sheet form.

3 SELECTION CRITERIA

3.1 With few exceptions, selection of corrosion resisting steels involves a sacrifice in corrosion resistance for increased mechanical properties. Where their mechanical properties and poor friction and galling characteristics can be accepted, the austenitic grades should be chosen for all exterior, heavy duty interior or maritime applications. Plating, dry film lubricants or other coatings to approved processes may be used to improve the frictional characteristics but nitriding or other diffusion processes must not be employed as these have a detrimental effect on corrosion resistance.

3.2 Where higher mechanical properties are required and/or where coatings cannot be employed, the choice lies between the precipitation hardening and martensitic steels. The former have generally higher resistance to corrosion, the latter lower manufacturing costs and are more readily available. The 11-13% chromium steels should not be used for exterior, heavy duty interior or naval applications unless some specific property makes their use imperative.

4 DESIGN CONSIDERATIONS

4.1 The maintenance of the passive oxide film on the surface of the steel is critically dependent on a ready supply of air to all surfaces. Therefore, design should wherever practicable avoid the occurrence of crevices or other areas where there may be a restricted access of air and where corrosive fluids may be trapped.

4.2 Where crevices are unavoidable, consideration should be given to the use of sealants to prevent the ingress of moisture. It is essential that sealing can be and is fully effective, as ineffective sealing may give rise to more severe crevice corrosion effects than would occur if sealing was omitted.

4.3 Where sealing is not practicable (e.g. on moving parts etc), then steels with the highest corrosion resistance which are compatible with other requirements shall be selected as these in general have the highest resistance to crevice corrosion. Of the standard steels, the austenitic grades are preferred if their other properties are acceptable, followed by precipitation hardening and 18% chromium, 2% nickel martensitic types. For very severe environments (e.g. maritime rotorcraft), especially where small amounts of corrosion products may affect the function of the parts, consideration should be given to the use of special purposes or proprietary materials which have been developed to give exceptional resistance to crevice corrosion, e.g. BS S537.

4.4 Due account should be taken of the effects of bi-metallic corrosion where corrosion resisting steels are in contact with other materials, (see Leaflet 407/3).

4.5 Where stainless steels are to be painted in particular applications (see Chapter 407, para 9.2.4), it is important to select a suitable pre-treatment from DEF STAN 03-2 to ensure good adhesion.

5 MANUFACTURING CONSIDERATIONS

5.1 As the corrosion resistance of stainless steels is dependent upon their chemical composition as it affects the surface, it is essential that all surface contamination which may be introduced by manufacturing operations is removed so that parts are put into service with the chemical composition at the surface in conformity with the material specification.

5.2 Surface contamination will inevitably occur as a result of such processes of manufacture as casting, forging, hot rolling etc. It is therefore essential that such surfaces have sufficient material removed to ensure freedom from contamination, either by machining, acid pickling, abrasive blasting or using non-metallic abrasives in accordance with DEF STAN 03-2.

5.3 Prior to operations involving heating, (e.g. welding, heat treatment etc), it is important that all oil, grease or other surface contaminants are removed by thorough degreasing or other appropriate methods. Inert gas shielded processes are preferred for welding. Vacuum or inert gas atmospheres are preferred for heat treatment and it is important that any atmosphere used must not introduce carbon or nitrogen into the surface. Any scale, oxidation or other surface discolouration resulting from heating operations shall be removed by appropriate methods given in para 5.2.

5.4 Precautions should be taken at all stages during manufacture to avoid as far as practicable surface contamination of parts which are to final dimensions on any surface. In particular, good house-keeping and protection should be practiced to prevent swarf and similar material being impressed into finished surfaces. Where the possibility of such surface contamination exists as a consequence of manufacturing operations e.g. sheet metal forming operations, then passivation in accordance with DEF STAN 03-2 Method M should be carried out before final inspection.

6 PROTECTION

6.1 Corrosion resisting steels should be selected so that they can resist the environment in which they will operate. For certain purposes (such as for identification or to suppress galvanic effects), it may be necessary to paint corrosion resisting steels. After a suitable pre-treatment (see DEF STAN 03-2) epoxy spray primers to DTD 5567 may be used, and they should be force dried to ensure good adhesion. Alternatively, clear baking resins to DTD 5562, stoved on at temperatures up to 200°C, will adhere well to pretreated surfaces. Good protection can be obtained, for example, for 11-13% chromium steels from aluminium-rich coatings (DTD 900), but these must be cured at significantly higher temperatures.

LEAFLET 407/7

PRECAUTIONS AGAINST CORROSION AND DETERIORATION

DEHUMIDIFICATION OF AIRCRAFT STRUCTURE AND SYSTEMS

1 Trials have shown that there are significant benefits to be achieved by providing a supply of dry air, from a commercial dehumidifier, to the aircraft structure and systems whilst the aircraft is on the ground. These benefits are a reduction in metal corrosion in the structure and engine components and an increase in reliability of the electrical and avionic systems.

2 Dry air may be introduced to the aircraft structure, through the Environmental Control System, engine intakes and/or specifically designed intakes, using blanks and trunking if this is called up in the Rotorcraft Specification. Dehumidification may be applied to individual aircraft in the open air or by de-humidifying the hangar or Aircraft Hardened Shelter during long term storage. Either method requires the structure and systems to allow for the free passage of air.

CHAPTER 408

PLASTICS MATERIALS

1 INTRODUCTION

1.1 For the purpose of this Chapter the term 'Plastics' has been taken to mean the following:

> "Material which consists of an organic polymeric substance incorporating when appropriate; stabilisers, anti-oxidants, fillers, fire retardants, plasticisers, particulate, and short fibre reinforcement."

These materials are normally used in extrusions, injection , compression/transfer and blow mouldings and thermoformed shapes but exclude those structurally significant materials, often termed "composites" such as carbon fibre reinforced plastic and glass reinforced plastic which can, in certain circumstances, be moulded using similar techniques.

1.2 Requirements for general aircraft glazing and transparencies are in Chapter 720 and are not considered in this Chapter.

2 SELECTION OF PLASTIC MATERIALS

2.1 When selecting Plastics materials the properties and characteristics listed below shall be considered. (Further amplification is given in Leaflet 407/1).

2.1.1 The mechanical properties of the component to be manufactured.

2.1.2 In component design, the properties of the material and the method of manufacture. Direct substitution of a plastics material for metal without a redesign of the component is frequently unsatisfactory.

2.1.3 The probability of fire and its effects.

2.1.4 The temperature range over which the component is intended to operate:

(i) Possible reduction in mechanical properties as temperature is increased within the normally acceptable working range (See also Chapter 409 Para 8).

(ii) The embrittling effect of low temperatures on many thermo-plastics.

(iii) The effect of temperature on the life of the component.

2.1.5 Degradation resulting from the temperature used in processing the material; it is important that process conditions are maintained to reduce this effect to a minimum.

2.1.6 Dimensional stability which is dependent on a number of factors including thermal expansion, creep, manufacturing process and moisture.

2.1.7 Changes in properties resulting from the absorption of water or water vapour other than the dimensional effect in Para 2.1.6 above; most mechanical and electrical properties will deteriorate with increasing moisture content.

2.1.8 Interaction of plastics with other organic materials with which they may come into contact and their possible corrosive effects on metals.

2.1.9 The effect of exposure to direct sunlight or other sources of ultra violet radiation.

2.1.10 Environmental stress cracking of plastics materials by liquids, particularly organic fluids.

3 GRADE A APPLICATIONS

3.1 All plastic materials to be used in Grade A applications (for details See Chapter 400) shall be procured to specifications defined as acceptable in Chapter 400. Material and process specifications to be used for the manufacture of plastic parts shall be approved in accordance with DEF STAN 05-123.

LEAFLET 408/1

PLASTICS MATERIALS

SELECTION OF PLASTICS MATERIALS

1 INTRODUCTION

1.1 This Leaflet amplifies the requirements of Chapter 408 para 2 relating to the selection of Plastics materials.

2 MECHANICAL PROPERTIES

2.1 When considering mechanical requirements of the component to be manufactured all aspects of the environment in which it is to function shall be taken into account, including temperature, proximity of other materials and contact with liquids whether by design or by accidental contamination.

3 EFFECTS OF FIRE

3.1 All plastics will burn under certain conditions. Some, once ignited, will continue to burn in the presence of sufficient air. Others will only burn if heat is supplied from another source. In many cases the smoke and fumes arising from the burning plastics are asphyxiating and/or toxic to an extent which varies with the nature of the plastics, the temperature of the fire and the degree of combustion. In general, partial combustion arising, from lack of air is more likely to produce toxic emissions.

4 DIMENSIONAL STABILITY

4.1 Dimensional stability is an important aspect in the selection of plastics materials. It is dependent on a number of factors including:-

4.1.1 Thermal Expansion. The coefficient of thermal expansion of plastics is higher than that of metals by approximately a factor of ten. In addition some plastics e.g., PTFE undergo a volume change due to allotropic change. Thus unless the component is to operate over a relatively small temperature range design should avoid the need for close tolerances and small clearances in, for example bearings where the journal is metallic. The effect is reduced by dilution in plastics containing fillers, e.g., glass fibre or ballotini.

4.1.2 Creep. All plastics and particularly unfilled thermoplastics are subject to creep under sustained stress at ambient temperatures. The effect increases with temperature. Fibre filled materials and thermo-setting plastics exhibit considerably better behaviour in this respect.

4.1.3 Manufacturing Process. Internal stresses resulting from the method of manufacture result in dimensional change with time, usually manifested as distortion. It is important that the process should be devised and adequately controlled to minimise this effect. Depending on the application stress relieving may he desirable and is usually essential before machining to required tolerances.

4.1.4 Moisture. All plastics absorb moisture to some extent, and some, particularly polyamide, attain a high equilibrium water content and hence are sensitive to the relative humidity of the environment. The resulting dimensional changes can be significant but occur slowly. Where frequent changes of environment are involved conditioning to the average humidity may be satisfactory. Filled plastics are less sensitive to this effect.

5 INTERACTION OF PLASTICS WITH OTHER ORGANIC MATERIALS

5.1 Plastics containing certain plasticisers may cause deposits on surrounding equipment such as windows. In addition these migratory chemicals can interfere with the functioning of electrical contacts.

6 ENVIRONMENTAL STRESS CRACKING

6.1 In some cases the effect can arise when the plastic is only lightly stressed and can result from internal stress which has not been relieved. Account should therefore be taken of all possible contact with liquids including operational liquids whether by design or accident, cleaning agents, paints etc.

CHAPTER 409

RUBBERS

1 SELECTION OF RUBBERS

1.1 Rubbers shall be selected with due regard to the static and dynamic requirements of the application and the environment in which the component is required to function. Environmental factors include, but are not limited to, temperature range, the presence of liquids and gases, whether by design or by contamination from adjacent systems, and other materials with which the rubber may come into contact.

1.2 Consideration shall also be given to the following cases.

1.2.1 The corrosive effect, particularly at elevated temperatures, of some rubbers including their components on metals e.g. , those rubbers based upon polymers containing halogens, such as polychloroprene etc.

1.2.2 Attack by ozone. Most natural atmospheres contain sufficient ozone to affect susceptible rubbers and where a rubber is used in a strained condition in these atmospheres it shall be based on an ozone resistant polymer. Account shall also be taken of the increased ozone concentration arising from the operation of certain types of electrical equipment and the presence of ultra-violet radiation.

1.2.3 Incompatibility arising from contact between different rubbers, e.g. , that between some silicone and polyurethane rubbers. The risk of interaction with other non-metallic materials e.g., rubber seals causing stress cracking in acrylic or polycarbonate components or migration of plasticizer from flexible PVC into a rubber component causing softening and swelling.

1.2.4 Subsequent treatment of a component or assembly of which the rubber forms a part e.g., that necessary prior to adhesive bonding, plating or painting.

WARNING: Particular care should be exercised when silicone polymers are involved as they can act as "release agents" consequently preventing effective adhesive bonding or painting. Contamination from products embodying these materials can occur by both contact and in some cases proximity alone.

1.2.5 Effects caused by the use of surface coatings and protectives including paints and dewatering fluids etc.

1.2.6 The probability of fire and its effects. All rubbers will burn under certain conditions. Some, once ignited, will continue to burn in the presence of sufficient air. Others will only burn if heat is supplied from another source. In many cases the smoke and fumes arising from the burning rubbers are asphyxiating and/or toxic to an extent which varies with the nature of the rubber and the temperature of the fire and the degree of combustion. In general, partial combustion arising from lack of air is more likely to produce toxic emissions.

2 GRADE A APPLICATIONS

2.1 All rubber materials for use in Grade A applications (for details see Chapter 400) shall comply with specifications defined as acceptable in Chapter 400, and all components shall be manufactured in accordance with *DEF STAN 93XX in addition to any drawing and any other special requirements which may be necessary. Whenever practicable test requirements for the components, sufficient to establish the suitability of the manufacturing process, shall be stipulated.

3 PACKAGING AND STORAGE

3.1 Rubber components shall be packaged in accordance with BS F69 and stored in accordance with BS F68.

* Awaiting publication.

PART 4 APPENDIX No.2
DETAIL DESIGN AND STRENGTH OF MATERIALS
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 400: GENERAL DETAIL DESIGN

400	MIL-HDBK-5	METALLIC MATERIALS AND ELEMENTS FOR AEROSPACE VEHICLE STRUCTURES
	MIL-HDBK-131	IDENTIFICATION MARKINGS FOR FASTENERS
	MIL-HDBK-132	PROTECTIVE FINISHED FOR METAL AND WOOD SURFACES
	MIL-HDBK-723	STEEL AND WROUGHT IRON PRODUCTS
	MIL-STD-490	SPECIFICATION PRACTICES
	MIL-STD-680	CONTRACTOR STANDARDISATION PROGRAM REQUIREMENTS
	MIL-STD-961	MILITARY SPECIFICATIONS AND ASSOCIATED DOCUMENTS
	MIL-STD-965	PARTS CONTROL PROGRAM
	MIL-STD-970	STANDARDS AND SPECIFICATIONS, ORDER OF PREFERENCE FOR THE SELECTION OF
	MIL-STD-1472	HUMAN ENGINEERING, DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT &, FACILITIES
	MIL-STD-1754	FASTENING DEVICES PREFERRED FOR DESIGN, LISTING OF
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NON-NUCLEAR SURVIVABILITY PROGRAM
	MIL-STD-2072	SURVIVABILITY, AIRCRAFT; ESTABLISHMENT AND CONDUCT OF PROGRAMS FOR
	MIL-STD-2154	INSPECTION, ULTRASONIC, WROUGHT METALS, PROCESS FOR
	MIL-STD-2175	CASTINGS, CLASSIFICATION AND INSPECTION OF
	MIL-STD-6866	INSPECTION, LIQUID PENETRANT
	MIL-I-6870	INSPECTION PROGRAM REQUIREMENTS, NON-DESTRUCTIVE, FOR AIRCRAFT AND MISSILE MATERIALS AND PARTS
	MIL-F-8490	FASTENER, CASE, FOR EQUIPMENT RACK SYSTEM IN AIRCRAFT
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-Q-9858	QUALITY PROGRAM REQUIREMENTS
	MIL-F-25173	FASTENER, CONTROL PANEL, AIRCRAFT EQUIPMENT
	MIL-T-31000	TECHNICAL DATA PACKAGES, GENERAL SPECIFICATION FOR
	MIL-A-81264	ASBESTOS FELT OR MAT, RESIN IMPREGNATED

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1. GENERAL

400 1.

2. GRADING OF PARTS AND ASSEMBLIES

400 2.	MIL-STD-490A	PARA: 3.1.3, 3.1.3.2.2, 3.1.3.2.3, 3.1.3.3.2.2, 3.1.3.3.3, 3.1.3.3.4
	MIL-STD-961	PARA: 3.3.3
	MIL-STD-965A/1	PARA: 1.2, 4.8
	MIL-STD-2154	PARA: 4.1
	MIL-STD-2175	PARA: 1.3.1
	MIL-STD-6866	PARA: 4.2

400 2.	MIL-I-6870E	PARA: 3.2, 3.3.3, 6.2.2, 6.2.6
	MIL-Q-9858A	PARA: 1.2, 1.3, 4.1, 4.5
	MIL-T-31000	

3. STANDARD PARTS

400 3.	MIL-STD-680	PARA: 5, 5.1.4
	MIL-STD-965A/I	PARA: 3.3.1, 3.9, 4.1, 4.3.3, 3.3
	MIL-STD-970	PARA: 4, 5
	MIL-STD-1472C	PARA: 4.2
	MIL- Q-9858A	PARA: 4.1

4. MATERIALS AND PROCESSES

400 4.	MIL-HDBK-132	
	MIL-STD-490A	PARA: 3.1.3.4, 3.1.3.5, 4.3.3.1
	MIL-Q-9858A	PARA: 1.2

5. STRENGTH OF MATERIALS

400 5.	MIL-S-8698	PARA: 3.1.1

6. USE OF WOOD

400 6.	MIL-F-7179F	PARA: 3.6.21

7. LOCKING OF THREADED FASTENERS

400 7.	MIL-HDBK-31
	MIL-STD-1754
	MIL-F-8490
	MIL-F-25173

8. USE OF LEAD SEALS ON LOCKING WIRE

400 8. MIL-HDBK-173

9. USE OF COLD FORGED STEEL BOLTS

400 9. MIL-HDBK-5

10. USE OF MAGNESIUM ALLOYS

400 10.

11. USE OF ASBESTOS

400 11.

12. CONTROLLED TIGHTENING OF BOLTS

400 12.

13. REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

400 13.	MIL-STD-2069	PARA: 5.3.3.2, 5.3.3.3
	MIL-STD-2072	PARA: 5.3.3.2, 5.3.3.3

CHAPTER 401 DESIGN DATA FOR METALLIC MATERIALS

401	MIL-HDBK-5	METALLIC MATERIALS AND ELEMENTS FOR AEROSPACE VEHICLE STRUCTURES
	MIL-HDBK-693	MAGNESIUM AND MAGNESIUM ALLOYS
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS

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CHAPTER 402: PROCESSES AND WORKING OF MATERIALS

402	MIL-STD-22	WELDED JOINT DESIGN
	MIL-HDBK-337	ADHESIVE BONDED AEROSPACE STRUCTURE REPAIR
	MIL-STD-490	SPECIFICATION PRACTICES
	MIL-HDBK-691	ADHESIVE BONDING
	MIL-HDBK-723	STEEL AND WROUGHT IRON PRODUCTS
	MIL-STD-970	STANDARDS AND SPECIFICATIONS, ORDER OF PREFERENCE FOR THE SELECTION OF
	MIL-STD-2154	INSPECTION, ULTRASONIC, WROUGHT METALS, PROCESS FOR
	MIL-1-6870	INSPECTION PROGRAM REQUIREMENTS, NON-DESTRUCTIVE, FOR AIRCRAFT AND MISSILE MATERIALS AND PARTS
	MIL-F-7179	FINISHES, COATINGS, AND SEALANTS FOR THE PROTECTION OF AEROSPACE WEAPONS SYSTEMS
	MIL-F-7190	FORGINGS, STEEL, FOR AIRCRAFT/AEROSPACE EQUIPMENT AND SPECIAL ORDINANCE APPLICATIONS
	MIL-A-8860	AIRPLANE STRENGTH AND RIGIDITY, GENERAL SPECIFICATION FOR
	MIL-A-9067	ADHESIVE BONDING, PROCESS AND INSPECTION REQUIREMENTS FOR
	MIL- Q-9858	QUALITY PROGRAM REQUIREMENTS
	MIL-A-25463	ADHESIVE, FILM FORM, METALLIC STRUCTURAL SANDWICH CONSTRUCTION
	MIL-A-46146	ADHESIVE SEALANTS, SILICONE, RTC, NON-CORROSIVE (FOR USE WITH SENSITIVE METALS AND MATERIALS)
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	MIL-P-46112	PLASTIC SHEET AND STRIP, POLYAMIDE
	MIL-P-83310	PLASTIC SHEET, POLYCARBONATE, TRANSPARENT

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2. SELECTION OF PLASTIC MATERIALS

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PART 5

AERO-ELASTICITY AND VIBRATION

CONTENTS

*** In Preparation**

CHAPTER 500

AERO-ELASTICITY

1 INTRODUCTION

1.1 The requirements of this Chapter are concerned with:

(i) The control of flutter, divergence, ground and air resonance and allied aero-elastic and aero-mechanical self-excited phenomena.

(ii) The restriction of structural distortion of the rotors, control systems and airframe within limits that will not dangerously impair stability and control.

1.2 Rotorcraft airframes and equipments are subjected to vibratory forcing from a number of sources. These forced response aspects of vibration are dealt with in Chapter 501.

1.3 Compliance with the requirements shall be established by calculation or by compliance with the relevant criteria given in the Leaflets of this Chapter, supported in each case by tests. Calculations shall be discussed with and approved by the Director, RAE.

2 STRUCTURAL DISTORTION

2.1 Structural distortion of the airframe shall not prevent the full range of all controls by being available to the pilot in all flight and ground running conditions. Parasitic control inputs to the rotors resulting from structural distortion of the airframe shall have no significant adverse effect on the handling of the rotorcraft.

2.2 Structural distortion of the control circuit shall not be such as to significantly reduce the range of control movement available to the pilot. Control circuits include those to any non-rotating surfaces, such as to those used for trim or stability augmentation, as well as those to the engines and rotors.

2.3 Structural distortion of the rotor blades shall not dangerously impair the handling of the rotorcraft in any ground or flight condition.

3 FLUTTER

3.1 DEFINITIONS

(i) DANGEROUS FLUTTER is an instability which can generate loads in any component of the rotorcraft in excess of the static and fatigue strength of the component, or dangerously impair the stability or control of the rotorcraft.

(ii) BENIGN FLUTTER is an instability which limits cycles at a known and acceptable amplitude, and causes no significant degradation of the rotorcraft's stability and control and induces no excessive loads in any component.

(iii) A ROTATING SURFACE FLUTTER can involve flap, lag or torsional motion of the surface, or any combination of these motions. It may also be influenced by flexibilities in the transmission system, control circuit and airframe.

(iv) NON-ROTATING, OR FIXED, SURFACE FLUTTER refers to the self-excited flutter instabilities of items on the rotorcraft other than the rotors. Such items might be:

(a) permanently attached aerodynamic surfaces, e.g. wings and tail-planes,

(b) temporarily attached aerodynamic surfaces, e.g. external loads and stores,

(c) non-aerodynamic surfaces that nevertheless might flutter, e.g., aerials.

3.2 The rotorcraft shall be free from dangerous flutter in all flight and ground conditions cleared for normal operations. Information on flutter of rotating surfaces is given in Leaflet 500/1. Leaflet 500/2 deals with the particular phenomena of Ground and Air Resonances which are self-excited instabilities involving the whole rotorcraft. Flutter of non-rotating surfaces is dealt with in Leaflet 500/3. Associated information that might also be of interest can be found in the corresponding Leaflets of Vol. 1, Chapter 500, for aeroplanes.

3.3 A benign flutter within the limits of the flight envelope may be permissible provided the associated oscillatory loads are known and accounted for in the usage spectrum of the rotorcraft. In these circumstances a cockpit indicator of the severity of the flutter is a desirable feature.

3.4 The mass balance of the rotor and blades must be sufficient to prevent pitch-flap flutter. Due consideration shall be given to the effects of erosion, humidity and repair schemes on the position of the inertia axis. The structural integrity of the mass balance shall be substantial at a rotor speed of 1.2 times the Never Exceed RPM.

3.5 The probability of failure of any device whose malfunction could cause flutter shall be extremely improbable. The effect of backlash in the control circuit shall be either eliminated, or shown not to result in flutter. In control circuits with manual reversion both power on and power off cases must be considered, and with power off both stick fixed and stick free conditions shall be covered.

3.6 Freedom from all forms of dangerous flutter, and the amplitude of any benign flutter, shall be demonstrated, initially on the ground, for safety, covering the widest permissible range of rotor speeds and blade pitch, and then later in flight. The flight demonstration shall be, where possible, at speeds up to 1.1 times the Never Exceed Speed for rotor speeds up to the maximum power off, excluding transients, and at speeds up to the Never Exceed Speed for rotor speeds up to the maximum power off including transients.

3.7 The effect of variations during service of any component which could influence flutter, e.g., control circuit backlash, shall be investigated during, the testing.

4 GROUND AND AIR RESONANCE

4.1 The rotorcraft shall have no dangerous tendency to oscillate on the ground with rotors turning, for all possible rotor speeds and landing gear arrangements, including the effects of fuselage picketing.

4.2 The rotorcraft shall have no dangerous tendency to oscillate in flight, over the entire flight envelope and possible range of rotor speeds, both power on and power off. Leaflet 500/2 gives background information on both ground and air resonance.

4.3 The reliability of the means for preventing ground and air resonance must be shown either by analysis and tests, or by reliable service experience. Alternatively, it shall be demonstrated by test that the malfunction of any single means used to prevent ground or air resonance will not cause either instability to occur. In particular, the malfunction of a single lag-plane damper shall either be extremely improbable, or it's failure must not result in ground or air resonance.

4.4 The probable range of variations, during service, of the damping action of the ground and air resonance prevention means must be investigated during the testing.

LEAFLET 500/1

AERO-ELASTICITY

ROTATING SYSTEM FLUTTER AND ALLIED PHENOMENA

1 INTRODUCTION

1.1 Aeromechanical and aeroelastic stability problems for rotorcraft may be conveniently divided into two groups - those which involve the motion of the complete rotorcraft, and those confined to the rotor systems. In the former group the major problem throughout the history of rotorcraft has been the mechanical instability known as 'ground resonance', and more recently with the advent of semi-rigid rotor systems the similar phenomenon which may occur in flight and is therefore called 'air resonance'.

These problems are discussed in detail in Leaflet 500/2.

1.2 Instabilities which are confined to the rotor system include pitch-flap flutter (akin to the fixed wing flutter problem), pitch-lag instability, stall flutter and flap-lag stability problems on tail rotors. In all of these rotor instabilities the basic mechanisms consist of couplings between the flap, lag and torsional motions of the rotor system. Problems usually occur, or are made more severe when the frequencies of the rotor modes are in close proximity. Thus, although the flexibility of the rotor hub mounting or the pitch control circuit does not feature directly in the phenomenon, it can still be of vital importance due to its effect on the frequency of the rotor modes. In particular the blade torsional mode is usually dominated by the control circuit flexibility, and will therefore be different for collective , cyclic and reactionless rotor modes. Before proceeding to a detailed discussion of some of the instabilities the concept of rotor degrees of freedom will be discussed.

2 ROTOR MODES

2.1 In order to fix ideas, consider a four blade rotor, with each blade identical to the others. The motion of each blade may be expressed in terms of a number of degrees of freedom, which for convenience may well be the normal modes of the blade. If each blade has n degree of freedom then a total of 4n generalized coordinates are required to completely specify the motion of a four blade rotor system.

2.2 Rather than using the individual blade motions as degrees of freedom it is sometimes useful to transform the equations to 'multi-blade' coordinates. Let p_{ij} be the amplitude of motion of the i^{th} mode in the j^{th} blade. Then the transformation is of the form:

$$u_i = \frac{1}{4} \sum_{j=1}^{4} P_{i_j}$$

$$v_i = \frac{1}{4} \left(P_{i_1} \quad P_{i_2} + P_{i_3} - P_{i_4} \right)$$

$$q_i = \frac{1}{2} \sum_{j=1}^{4} P_{i_j} \sin \psi_j$$

$$r_i = \frac{1}{2} \sum_{j=1}^{4} P_{i_j} \cos \psi_j$$

$$\text{where} \quad \psi_j = \Omega + \frac{\Pi}{2} (j - 1)$$

is the instantaneous azimuth angle of the j^{th} blade and Ω denotes the rotor speed.

2.3 The degrees of freedom u_i, v_i, q_i and r_i are referred to as 'multi-blade coordinates'. These coordinates have certain advantages over the individual blade degrees of freedoms, especially in problems which involve coupling between the rotor motion and fuselage motion (including the control circuit). This is because the variables u_i etc., represent the total rotor behaviour. For example, if P_{ij} is the fundamental flapping motion of the j^{th} blade then u_i is the collective or 'umbrella' rotor mode, being the average instantaneous coning angle; q_i and r_i represent cycle lateral and longitudinal disc tilts about the rotor hub, and v_i is a 'reactionless' mode. This last term arises from the fact that if the rotor motion consisted solely of response in the v_i mode no net shear or bending moment would be transmitted to the fuselage. In the disc tilts net moments are applied to fuselage, but no shears, whereas in the collective mode a net shear but no moments are transmitted.

2.4 Because the variables u_i etc., represent the motion of the rotor when seen from a non-rotating frame of reference, whereas the individual blade freedoms are relative to a frame rotating with the rotor, the frequency of the cyclic modes appear to be different from the individual blade modes. If w_i is the frequency of the i^{th} mode, as seen by an observer rotating with the rotor, then to an observer fixed in the airframe the frequencies of the cyclic i^{th} mode appears as $w_i \pm \Omega$. This fact can be most simply understood by considering a blade flapping at once per rev. To an observer in the rotor system this motion has a frequency of 1Ω, but to an observer in the airframe it appears as a steady disc tilt, at zero frequency.

2.5 Fixed-axis system modes with a frequency of $w_i + \Omega$ are referred to as progressing modes, and modes with a frequency of $w_i - \Omega$ are referred to as regressing modes. For flapping modes the q and r multi-blade coordinates represent disc tilts about a fixed set of axes, whereas for lead-lag modes q and r are proportional to the displacement of the centre of mass of the rotor system from the centre of the hub, again relative to a fixed frame of reference. For torsion modes q and r represent the cyclic pitch.

3 PITCH-FLAP FLUTTER

3.1 This is a two degree of freedom instability, the freedoms being flat and torsion, with some mechanism coupling the two motions. The coupling between torsion and flap is aerodynamic; a perturbation in blade pitch induces changes in blade lift loads and causes the blade to flap. The coupling between flap and torsion may be of a number of forms, for example chordwise centre of mass offsets from the feathering axis or kinematic pitch-flap coupling due to the orientation of the hinges and control circuit.

3.2 Flutter can generally be avoided on main rotors by mass balancing the blade so that the chordwise position of the centre of gravity of each spanwise element is forward of the ¼-chord point. This is usually accomplished by the introduction of non-load-carrying mass balance weights along the full length of the blade. In the region of the attachment of the root end of the blade to the inner spar, local reinforcing usually has the effect of moving the centre of gravity well aft of the ¼-chord point. However, analysis shows that for a blade which is reasonably stiff in flatwise bending and torsion, the product of inertia of the blade about axes coincident with the flapping and feathering hinge lines is the significant parameter. Therefore the effect of an aft movement of the centre of gravity in the region of 20% radius is counteracted by a relatively small weight positioned near the leading edge of the blade at the tip. For a blade where complete de-coupling is not achieved by mass balancing, then flatwise stiffness and torsional stiffness of the blade and pitch change linkage at the blade root are important parameters.

3.3 Flutter of tail rotors is more likely to occur due to the existence of a large pitch-flap kinematic coupling introduced to reduce tail rotor stresses and flapping. The coupling is usually of the order of one degree reduction in blade pitch per degree of blade flapping. A pitch-flap coupling can also influence main rotor blade flutter.

3.4 For both main and tail rotors the stiffness and damping of the pitch control circuit between the blade and the operating jack plays an important part. In general, stable solutions can be found with both low and high pitching mode frequencies in the absence of pitch circuit damping.

3.5 It is necessary to consider the collective, cyclic and reactionless modes of the rotor, since the effective control circuit stiffness, and hence torsion mode frequency, may well be different for all of these modes.

4 STALL FLUTTER OF ROTOR BLADES

4.1 This is a single degree of freedom flutter problem, involving torsional oscillation of the blade. It can be expected on any rotor, and is observed as high oscillatory loads in the blade pitch control linkage. The aerodynamic conditions leading to its onset will initially occur over a small proportion of the rotor disc. The mechanism of the instability is negative damping in pitch due to aerodynamic moment hysteresis caused by periodic shedding of intense vorticity at a blade angle of attack considerably greater than the static stalling angle.

4.2 Stall flutter is a benign flutter in so far as it results in a limit cycle motion of the blade. However, the severity of the flutter increases as the blade loading is increased. Thus its boundary must be properly mapped in order to establish the normal flying limitations. Cockpit instrumentation capable of indicating the severity of the flutter is a desirable feature if stall flutter can occur at speeds less than 1.1 times the rotorcraft Never Exceed Speed.

5 PITCH-LAG INSTABILITY (PLI)

5.1 Although this phenomenon is usually referred to as pitch-lag instability, the basic degrees of freedom are flap and lag, with the instability arising from the presence of a kinematic pitch-lag coupling. The effect of this coupling is to induce aerodynamic lift loads in response to blade lag deflections, which cause the blade to flap and via the Coriolis loads induce more lag motion.

5.2 One aspect which can degrade main rotor stability in certain flight conditions is the change in kinematic pitch-lag coupling with steady coning angle, steady lag deflection and impressed blade pitch due to changes in the orientation of the track rods. This will be aggravated by the use of short track rods. With hingeless rotors the steady flap and lag deflections tend to be much smaller when compared with articulated values, thus the instability is less likely on these rotors. Pitch-lag instability is generally experienced as a limit cycle oscillation of the rotor blades which transmits a stirring motion to the airframe, due to the oscillatory shear forces generated by the blades in the plant of rotation.

5.3 Parameters which affect pitch-lag stability include:

Steady coning angle
Steady elastic bending in flap
Blade lag stiffness
Pitch-flap coupling
Lead-lag damping

6 TAIL ROTOR STABILITY

6.1 Accurate prediction of stability boundaries for tail rotors is inherently more difficult than for main rotors due to the very disturbed airflow in certain flight conditions in the vicinity of the rotor. However considerable understanding of the aeroelastic behaviour can be obtained by the use of 'free flow' aerodynamic models; that is to say models which ignore such complicating effects as fin blockage and main rotor wake interaction.

6.2 A flap-lag limit-cycle instability which occurs when the lag mode frequency is close to the flap mode frequency has been observed on some rotorcraft, and due to its limit-cycle nature is known as 'Buzz'. Tail rotors usually have lag mode frequencies greater than once per rev, but to avoid excessive hub and blade loading this frequency must be well below two per rev. As tail rotor blade pitch is increased the lower flatwise blade stiffness compared with the edgewise stiffness reduces the lag frequency from the zero pitch value. Furthermore, with the large amount of pitch-flap coupling employed on tail rotors to reduce the once per rev flapping the fundamental flap mode frequency will be well above once per rev. Consequently, unless special care is taken the flap and lag mode frequencies will coalesce at high values of blade pitch, and therefore result in Buzz.

6.3 A more severe instability has been encountered on some tail rotors, which has resulted in structural damage to the rotor. The mechanisms causing this instability are not well understood, but it is believed that all three blade motions are involved, i.e., flap, lag and torsion. Any rotor for which the three fundamental blade modes are in close frequency proximity should be investigated with caution, especially if significant amounts of pitch-lag or pitch flap coupling is present.

7 FLUTTER ANALYSIS

7.1 The degrees of freedom involved in various types of known flutter have been discussed above, and clearly these are the degrees of freedom which must be included in any analysis. However, the above list is not necessarily complete, other forms of flutter may also exist.

7.2 To accurately predict flutter boundaries it is important that all couplings between flap, lag and torsion are properly accounted for, including coupling which can arise from perturbations about a state of steady stress. These later couplings include for example pitch-flap-lag-couplings arising from the non-matched stiffness of the rotor blade, i.e.,

$$M_{PITCH} = M_{FLAP}\, M_{LAG}\, (1/EI_{FLAP} - 1/EI_{LAG})$$

Where M_{PITCH}, M_{FLAP} and M_{LAG} are the moments about the pitch, flap and lag axes of the rotor and EI_{FLAP}, EI_{LAG} are the flap and lag stiffnesses.

7.3 The effect of possible variations in parameters which could influence flutter, such as control circuit backlash, wear in bearings, loss of damping in hydraulic dampers, variation in blade mass or stiffness due to usage etc., need to be assessed in the analysis.

7.4 The effects of the dynamics of the transmission system, control circuit and airframe on the rotor modes must be considered. The influence of other parameters such as the type of hub (articulated, semi-rigid, elastomeric), order of hinge placements, and blade construction can only be assessed for specific cases; in general no universal recommendations can be made.

8 FLUTTER TESTING

8.1 Testing for all forms of rotor instability should form a normal part of the development flying of any new rotor configuration. Careful and progressive exploration of the flight envelope should be conducted, starting from ground running, and low level hovers, with continuous monitoring of important parameters. Parameters to be monitored will include flap and lag bending stresses of the rotor and control system loads.

8.2 A typical exploration of the flight envelope will be:

(i) On the ground, establish the minimum practical rotor speed, at minimum rotor pitch.

(ii) Excite the rotor with small disturbances and check for satisfactory decay.

(iii) Increase the rotor speed in small steps, say 5% N_R, and repeat the inputs. Continue to maximum possible rotor speed.

(iv) Repeat this procedure at increased rotor pitch, up to the maximum safe level on the ground.

(v) Establish a hover and cautiously explore the full range of sideways, rearwards and quartering flight, at intervals of 30° in azimuth and 5 knots in speed. Note especially the rotor behaviour in recoveries from the manoeuvre to the hover, as this frequently generates the biggest rotor stresses.

(vi) Explore spot turns up to the maximum permissible rate. For tail rotors attempt to obtain full left and right tail rotor pitch during the airfield manoeuvres.

(vii) Explore the forward flight envelope up to 1.1 x Vne, including banked turns. At Vno explore the effects of power variations and rotor speed variations.

(viii) Explore autorotations, including the maximum and minimum rotor speeds.

8.3 During these tests on-line monitoring of flap, lag and torsional stresses should be made. Real time spectral analysis of critical components is also very desirable, to enable transient modal responses to be distinguished from forced response.

8.4 It is important that the effect of degradation in means of preventing any of the instabilities should be covered by flight testing. The following effects should be investigated:

Loss of damping in lag-plane dampers.
Loss of stiffness and damping in elastomeric dampers and bearings.
Backlash in the control circuit.
Permitted failure modes in the case of duplicated power controls.

8.5 If the control circuit has a manual reversion capability then all instabilities should be examined both with the control circuit powered and unpowered, and in the unpowered case both stick-fixed and stick-free conditions should be investigated.

8.6 During these investigations proper and adequate attempts to excite flutter should be made, and freedom from flutter should be shown by analysis of the measured response of the rotor.

LEAFLET 500/2

AERO-ELASTICITY

GROUND AND AIR RESONANCE

1 GROUND RESONANCE

1.1 EXPLANATION

1.1.1 Any rotorcraft with a main rotor fundamental lag mode frequency less than once per rev is susceptible to ground resonance. The degrees of freedom involved are the lead-lag motion of the rotor blades and any fuselage mode containing in-plane motion of the rotor hub. The instability occurs in the vicinity of a frequency coalescence between the regressing lag mode and the fuselage mode, provided the rotor speed is greater than the fundamental lag frequency; the phase relationships between the couplings are such that a frequency coalescence when the lag mode frequency is greater than the rotor speed does not produce an instability, neither does a coalescence with the progressing lag mode.

1.1.2 Important parameters with regard to ground resonance are lag mode frequency and damping, fuselage frequency and damping and fuselage mode shape. Of lesser significance are flap mode stiffness and aerodynamic loads: ground resonance is a purely mechanical instability which can exist in vacuo, although on some rotorcraft with only a very marginal degree of instability rotor couplings can be used to stabilise the rotorcraft via the aerodynamic loads.

1.1.3 Accurate predictions of ground resonance stability characteristics require a fairly extensive mathematical model with flap and lag rotor degrees of freedom and at least four fuselage freedoms - pitch/longitudinal and roll/lateral. Rotorcraft stability augmentation systems (SAS) can also affect ground resonance, and if significant response of these systems is expected at the frequencies associated with ground resonance they must be included in the analysis. (A simpler approach may well be to ensure that the feedback systems are filtered in such a way as not to respond at the frequency of the ground resonance oscillations).

1.1.4 The standard way of suppressing ground resonance is to add damping to the system, but it is a feature of ground resonance that it is the product of the damping in the rotor and fuselage which is important, not the individual values.

1.1.5 Ground resonance is completely eliminated if the blade lag frequency is greater than once per rev. A rotor with this characteristic is termed 'super critical'. If this is achieved then no frequency coalescence can take place at any rotor speed with the regressing lag mode. This solution is found in the 2-blade teetering rotor.

1.1.6 For a conventional rotorcraft, the significant contributions to the flexibility and damping of the airframe come from the undercarriage. The important rigid body modes are those containing significant motion of the rotor head in the rotor plane of rotation.

A significant point is that the full available range of motor lift should be explored when demonstrating freedom from ground resonance since undercarriage geometry, oleo stiffness and tyre stiffness and damping rates are all functions of the reaction on the undercarriage leg. The non-linear effects of the oleo stiffness are very significant and due to the oleo sticking when the load becomes less than the breakout load in the high lift condition, the oleo cannot dissipate any energy, and hence apart from the effects of stiffness change, damping is only provided by hysteresis effects of the tyres and airframe structural damping. Ground resonance can also occur due to flexibility in the rotor pylon restraint system.

1.1.7 The theoretical treatment of the problem assumes linear characteristics of the airframe and hence a variety of conditions appropriate to a prescribed lift condition are normally evaluated. Viscous damping is normally assumed in both the airframe and the rotor and once stability boundaries have been established, a conversion to the particular form of damping employed in the rotor can be made. The conversion of the viscous damping requirement in the blade to the other forms of damping, e.g., a friction damper, is based on equating energy dissipation per cycle of oscillation. This leads in the case of the friction damper to the concept of an allowable blade swing angle (in the ground resonance mode) above which the rotorcraft will become divergent. Disturbances below this level will however subside. Thus it is required to know the levels of hub acceleration in the plane of the rotor which are experienced in service, and which may force the blades to oscillate in the ground resonance mode. Friction dampers are not the only ones exhibiting these effects. A common design of hydraulic damper employs a very high rate of V^2 damping followed by a constant torque cut off, which approximates very well to a friction damper, but does not suffer from the 'sticking' effect of the true friction type. This sticking effect can cause severe forced oscillation during rotor acceleration on the ground, and once per-revolution vibration following power changes in the air which cause the datum lag angle to change.

1.1.8 To reduce the number of different combinations of undercarriage, rotor thrust and rotorcraft weight requiring analysis the rotor lag damping could be sized on that required to prevent ground resonance with the worst possible combination of parameters. It can be shown that more damping is required as the fuselage frequency increases, provided the frequency coalescence still occurs below the maximum possible rotor speed. Thus a pessimistic assumption is that the frequency coalescence occurs at the maximum operating rotor speed.

1.1.9 Increasing the ratio of the mass of the rotor to the mass of the fuselage also increases the required damping; thus the combination of a light fuselage and heavy rotor system produces the most unstable situation. On a real rotorcraft it is unlikely that the fuselage mode consists of pure translation of the rotorcraft, some pitch or roll rotation will be involved. Thus it is necessary to include fuselage pitch and roll degrees of freedom in the complete analysis.

1.1.10 Ground resonance can also occur with tail rotors, if the first in-plane frequency is below the rotational speed. In this case the coupling with the fuselage is more likely to occur with an elastic mode of the airframe, rather than the whole rotorcraft moving on the undercarriage.

1.1.11 If ground resonance does occur it will be seen as a violent shaking of the rotorcraft, often resulting in the complete destruction of the rotorcraft.

1.1.12 Elastomeric components (in the rotor pylon support system, possibly in the landing gear, and possibly in the rotor head) are significantly affected by ambient temperature prior to warm up. Their damping characteristics require thorough investigation for the range of rotorcraft operating environment.

1.2 TESTING PROCEDURES

1.2.1 For such configurations which are not susceptible to ground resonance (first order in-plane frequency above rotor turning speed), a simple rotor r.p.m. run up and run down with appropriate cyclic control displacement (i.e., excitation of any inherent vibrations) is adequate demonstration that a ground resonance condition does not exist. Hingeless "rigid" rotors, such as Bell Rotorcraft two blade designs, belong to this type of rotor system.

1.2.2 For configurations that are susceptible to ground resonance (i.e., first in-plane frequency is below the rotor turning speed), ground resonance is generally prevented by dampers on the blade acting in the plane of rotation, dampers on the landing gear (sometimes serving as oleo struts), or by proper placement of the landing gear-frequencies combined with rotor and/or landing gear dampers.

1.2.3 For each rotorcraft configuration tested, the rotorcraft should be positioned on the ground in flat pitch with the rotor stabilised at the minimum practical rotational speed, or optionally at a speed shown analytically to have significant margin from indicated resonant conditions. Control system inputs should be used to disturb the system for evaluation of subsequent damping.

1.2.4 For each incremental increase in rotor speed and for each rotor speed setting at increments of collective pitch settings, cyclic and collective inputs should be investigated prior to proceeding to the next rotor speed setting. These inputs should cover the appropriate range and combinations of amplitude and frequency.

1.2.5 Cyclic pitch inputs should be made either by the pilot through the cyclic stick or through a signal generating device working in conjunction with the cyclic controls. For each frequency of input, amplitude of the inputs should be increased incrementally and ultimately should be large enough to generate responses representative of normal ground and flight operation on the rotor and support system. The inputs should continue for a time sufficient to execute five complete counterclockwise circles of the cyclic stick (about neutral) at the selected frequency.

1.2.6 At each amplitude of cyclic input, the excitation frequency should be incrementally increased over the range of the blade in-plane frequency in the fixed system. Rotor speed settings should be increased to 1.05 times the maximum power on rotor speed. Collective pitch settings should be increased in increments of not more than 20% to maximum collective or alternately to the collective setting required to become partially airborne (when the cyclic is displaced as noted).

1.2.7 Typically, articulated rotorcraft have natural frequencies on the blade in lag of approximately 0.3 times the power on main rotor r.p.m.; soft in-plane rotors have natural frequencies approximately 0.7 times the main rotor r.p.m. Therefore, for example, for a rotorcraft with an in-plane frequency of 0.3/rev, operating at 300 r.p.m., and with 6 inches of total lateral cyclic stick displacement, the stick should be rotated for 5 revs in a 0.6 inch-diameter circle at ((1 - 0-3) x 300 r.p.m.) or 3.5 cycles per second to attempt excitation of possible resonant frequencies. At the conclusion of the excitation, the cyclic stick should be returned to the neutral position while continuing the recording of data listed in the table below.

1.2.8 The complete program should again be repeated with cyclic excitation inputs from the directional and longitudinal controls if critical for the type of rotorcraft being evaluated.

1.2.9 If onset of ground resonance is encountered, the typical recommended corrective action is to increase the collective pitch and rotor speed and become airborne. However, lowering the collective pitch and applying the rotor brake (if installed) has been effective for some designs and is considered a satisfactory procedure if resonance can be consistently stopped.

1.2.10 In operation, the resonance characteristics should be checked during take off and landing at zero speed and during run on landings using various power values. Under all conditions, any oscillations which may be introduced should be damped. However, no instability should occur at any operating condition such as during r.p.m. changes from minimum to maximum and idle to maximum. For rotorcraft with wheeled landing gear, uneven taxi surfaces in conjunction with particular taxi speeds, may excite ground resonance and should be evaluated by taxiing on typical surfaces.

1.2.11 For each configuration, slow vertical landings are made to establish the touch down collective pitch angle for each rotor speed. For those rotorcraft equipped with Stability Augmentation Systems (SAS), all ground resonance investigations should be conducted with SAS on and SAS off. This includes hovering and running take offs and landings, taxi tests, and specific ground resonance tests noted herein. Tests should be conducted in all permitted SAS configurations such as roll channel on and pitch channel off, where such configurations are possible and authorised.

1.2.12 Landing should be made at the maximum touchdown speed proposed with the rotor speed stabilised.

1.2.13 Special considerations:

(i) The influence of variables, including environmental effects, corresponding rotorcraft component characteristic changes, operating parameters, and surface conditions should be investigated over the ranges proposed for certification. Additionally, the potential of mis-servicing and possible failure modes should be evaluated. For ground resonance qualification, where practical, variations from the baseline test configuration may be accomplished by ground run, analyses, component tests, rotorcraft shake test, the specification of special operational procedures in the rotorcraft flight manual, or a combination thereof. Detailed and rational analyses showing acceptable correlation to the baseline tests, and for which the input parameters were verified by drawings, calculations, component static or dynamic tests, or by aircraft shake tests simulating the conditions/configurations in question may be used to limit testing to only those variables and operational conditions showing marginal or unacceptable system damping. All operational limitations should be clearly stated in the rotorcraft flight manual.

(ii) When operating on the ground, there may be a tendency for the rotorcraft to exhibit a "ground bounce". For many configurations, this is a benign, although undesirable phenomenon, which may be aggravated by pilot induced oscillations (PIO), particularly if there is little or no friction on the collective.

1.2.14 Rotorcraft with fully articulated rotor heads and landing gear oleos in either skid or wheel configuration have tendencies for ground bounce to occur when light on the oleos, either just prior to take-off, just after landing contact, or during a ground topping check of the engines. This bounce may induce ground resonance, particularly if the intensity of the bounce is aggravated by PIO. The corrective action is either to lift off to a hover or to positively lower the collective and remain on the ground.

1.2.15 Instrumentation and Data Acquisition

(i) Atmospheric Conditions (to be manually noted):

(a) Altitude

(b) OAT

(c) Wind Velocity

(ii) Rotorcraft Configuration (to be manually noted):

(a) Gross Weight

(b) C.G.

(c) Tyre Pressure

(d) Landing Gear Oleo Pressure.

(iii) Instrumentation (for recording during test)

(a) Main Rotor r.p.m.

(b) Time history of cyclic control fore-and-aft and lateral stick position.

(c) Time history of collective control stick position.

Plus sufficient of the following in order to obtain the modal damping.

(d) Time history of rotor damper motion.

(e) Time history of pylon component motion.

(f) Time history of landing gear (oleo) motion.

(g) Time history of rotorcraft motions.

(h) Time history of rotor lead-lag stress.

2 AIR RESONANCE

2.1 EXPLANATION

2.1.1 This instability is closely related to ground resonance, but as the name suggests it occurs in flight, and more especially it is associated with hingeless rotor systems. The 'fuselage' mode in this case arises from the coupling between blade flapping and fuselage pitch and roll motions. It might be expected that in flight no such stability is possible due to the removal of the undercarriage restoring forces and the resulting zero frequency airframe modes. This is broadly true for articulated rotors, but for a semi-rigid rotor the regressing cyclic flapping mode can couple with the airframe roll and pitch motion to produce 'slow gyroscopic' or 'pendulum' modes of the rotorcraft at frequencies which are close to the regressing lag mode frequency at normal operating rotor speed. The frequency of the pendulum mode is closely related to the fundamental flap mode frequency and for articulated rotors is so low that coalescence with the lag mode is prevented within the range of rotor speeds encountered in flight.

2.1.2 Because the 'stiffness' term arises from blade flapping, which is aerodynamically heavily damped, the effective damping in the pendulum modes is high. However it has been shown that the important parameter in the suppression of ground resonance is the product of the lag and fuselage damping, not the individual values. Indeed, with high damping in the fuselage and low damping in the lead-lag motion the rotor speed range for which instability can occur is increased. Thus the possibility of instability remains if the lag damping is very low. On the other hand the amount of lag damping required to suppress air resonance will be small, consequently air resonance will only be a problem for rotors with little inherent lag damping and relatively high flap stiffness - a likely combination for both hingeless and bearingless rotors.

2.1.3 In addition to the basic mechanism of air resonance the coupling between flap and lag due to Coriolis force is important. Theoretical analyses indicate that no instability is possible in the absence of a steady coning angle.

2.1.4 Theoretical studies indicate the degree of instability to be low and that generally it can be suppressed by the addition lead-lag dampers or the presence of favourable rotor couplings. Indeed the amount of damping required is much less than is typically needed to control ground resonance. If however, a solution to the ground resonance problem is chosen which does not result in additional lead-lag dampers, such as by undercarriage design to prevent frequency coalescence, air resonance becomes a very real possibility.

2.2 TESTING

2.2.1 If air resonance is a possibility for the rotorcraft the first flight should consist of a low level hover. Cyclic stick stirs of progressively increasing magnitude should be made, with the subsequent behaviour of the rotorcraft being noted. These stirs should be repeated for the complete range of rotor speeds available in flight .

2.2.2 The tests should be repeated for the complete range of rotorcraft weight and centre of mass. Forward flight should be explored in a progressive and cautious manner. The tests should be carried out with SAS equipment both engaged and disengaged, for all permitted combinations of roll channel on, pitch channel off etc. The behaviour of the rotorcraft should be carefully noted during maximum power climbs and autorotation.

LEAFLET 500/3

AERO-ELASTICITY

FLUTTER CLEARANCE PROGRAMME FOR NON-ROTATING SURFACES

1 INTRODUCTION

1.1 This Leaflet describes the programme of calculations and tests likely to be required under Chapter 500, para 3.2.

1.2 Because of relatively low maximum forward speeds of rotorcraft, flutter of any non-rotating surface is an unlikely occurrence, nevertheless it is recommended that all configurations are shown to be free from damaging flutter, however brief.

1.3 Maintenance of rotary wing flight and its safe control comes mainly from the rotating main and tail rotors. In consequence any flutter of a non-rotating surface might not be as serious an event as it would be on a fixed wing aeroplane. An exception to this is where a surface is part of a stability augmentation system when flutter could have an adverse and possibly dangerous effect. Even where flutter is not so catastrophic, the vibration could still produce fatigue damaging loads, or even interfere with the operation of a rotor.

1.4 For rotorcraft the conventional methods of flutter prediction that are used for fixed wing aeroplanes might be complicated by the disturbed airflow from the rotors in some of the flight conditions. For these, wind tunnel tests could be made, but would not cover all possible variations, and so instrumented in-flight clearance tests are more likely to be resorted to.

2 FLUTTER CALCULATIONS

2.1 Flutter calculations should begin as early as possible in the rotorcraft design process so that the consequences of flutter avoidance can be incorporated into the basic design. It will then be necessary to review situations as design and development take place, sometimes establishing limits on some of the design features, e.g., structural and control stiffnesses.

2.2 The scope of flutter calculations will be appropriate to the availability of relevant information. Initially, when little quantitative data is known it is sufficient to identify the types of flutter to be expected. Some analytical modelling can then follow using estimated data about the design. This will establish the theoretical models and determine the actual data and accuracy required for the predictions.

2.3 An important aspect of the identification of the types of flutter in calculations is the assessment of the flutter sensitivity to changes in aerodynamic data. This is required for an appreciation of the effect of inaccuracies in estimated aerodynamic forces and the need for experimental verification; this is especially of concern where a surface is operating in the unsteady aerodynamic environment of the rotors. Possible combinations of flight path and fuselage attitude could result in non-rotating surfaces with pitch angles and relative air velocities not normally encountered by fixed wing experience.

2.4 The effect of stiffness variations on calculated flutter speeds of a structural component can usually be found with little loss of accuracy from calculations for the component in isolation.

2.5 The effect of rotorcraft stability augmentation systems on flutter, and vice versa, should be found as soon as the system is detailed.

3 FLUTTER MARGINS

3.1 With some systems (e.g., some control surfaces mounted on the fuselage) it may be found that there is a range of values of a parameter (e.g., mass, stiffness, backlash or damping) in which there is either no flutter at any speed or flutter at a very high speed. In these cases the objective should be to achieve a value for that parameter in the flutter-free range.

3.2 If flutter cannot be eliminated, the objective should be to achieve a margin between calculated flutter speed and Vd, the maximum design forward speed, regarded here as a speed incapable of being exceeded by the rotorcraft. This implies that there is a maximum allowable forward speed Vne, which is lower than Vd, the difference between the speeds providing a safety margin for the accommodation of inadvertent deviations to speeds above Vne. The calculated flutter speed should not be below 1.1 Vd.

4 WIND TUNNEL MODEL TESTS

4.1 Wind tunnel tests are desirable, where appropriate, to check the aerodynamic assumptions made in any flutter calculations.

4.2 For non-rotating surfaces that are operating in airflows undisturbed by the aerodynamics of a rotor, the conventional wind tunnel testing procedures of fixed wing aeroplanes are applicable. In a direct approach, the pressure distributions or forces on models undergoing prescribed motions are measured and compared with estimates. In an indirect method, the dynamic behaviour, including critical flutter speeds, frequencies and mode shapes, of a structurally representative model is
found and compared with predictions for that model, using the same aerodynamic theory that was used for the predictions, thereby effectively calibrating the theoretical techniques. This assumes that no flutter significant differences are introduced by the wind tunnel modelling techniques.

4.3 Where non-rotating surfaces are operating within the environment of rotor downwash special tests might have to be made. Such testing is, however, likely to be impracticable because of the difficulty or even impossibility of adequately representing actual flight conditions in a wind tunnel. Some tests might however be useful for the development of any theoretical model of a surface operating in a rotor downwash e.g. , for the aerodynamic forcing.

5 STATIC STRUCTURAL TESTS ON COMPONENTS

5.1 Static testing of relevant components to establish the accuracy of data used in calculations. This includes any confirmation of characteristic assumptions made about, for example, the linearity of stiffnesses or whether backlash is present in mechanical components thereby changing effective stiffnesses.

5.2 The static properties of some structural items are difficult to estimate accurately. Examples are, carrier-store pylons and their attachments, the operating mechanism and supporting structure of all-moving surfaces, and conventional aerodynamic surfaces. Where flutter calculations are sensitive to such values testing will normally be required.

6 ACTUATOR IMPEDANCE TESTS

6.1 Impedance tests on control surface actuators will be required. The objective of these tests is to provide information for checking data used or introducing directly into any flutter calculations.

6.2 Similar impedance testing will be required on the control actuators of the rotors for use in the corresponding stability analyses.

6.3 The testing will determine linearity of the actuator impedance and the levels of effective backlash present.

7 STILL-AIR RESONANCE TESTS

7.1 Still-air resonance tests on a complete rotorcraft are ideally required: mostly it is sufficient to replace rotor blades by equivalent masses at their hubs. As such testing is also required for fuselage vibration and stress investigations this testing is likely to be a joint requirement activity. If total rotorcraft resonance testing cannot be made before first flight, some useful information might be obtainable from local response tests or even impulse (bonk) measurements on appropriate surfaces.

7.2 The objective is to measure the nearest practical equivalent to undamped normal modes within a frequency range that covers all modes that may be significant in flutter.

7.3 The aim should be to measure modes of a standard such that they can be incorporated directly into flutter calculations, but the information obtained can usually be used as a basis for improving the structural representation used in calculations.

8 INITIAL FLUTTER CLEARANCE

8.1 For the first flights of a prototype rotorcraft, an initial flutter clearance will be agreed with the Rotorcraft Project Director.

9 FLIGHT FLUTTER TESTS

9.1 Where component flutter is considered not to impair the control of the rotorcraft, mainly because safe flight comes from the rotors, flight flutter testing is most likely to be done by observing or monitoring relevant component motions subject to the ambient vibrational forcing of the flight. The test procedure will usually be that flight at higher airspeeds is allowed only after analysis of measurements at lower airspeeds show no evidence of flutter within the next speed increment.

9.2 Where downwash from the rotors can interfere with the non-rotating surfaces flight testing should be programmed accordingly.

9.3 The objective of the flutter tests is to demonstrate freedom from flutter up to Vd and to obtain compliance to the satisfaction of the Rotorcraft Project Director.

9.4 If practicable, the effect of failures in rotorcraft systems relevant to flutter should be represented. Also where practicable, tests should be conducted with any autopilot or stability augmentation system switched off.

10 FINAL FLUTTER CLEARANCE

10.1 The final flutter clearance will normally be given after flutter testing up to Vd, or to the maximum permitted in the derivation of operational airspeed clearances. The interpretation of flight clutter test results with regard to the basic requirements of Chapter 500 should be agreed with the Rotorcraft Project Director.

CHAPTER 501

VIBRATION AND INTERNAL NOISE

1 INTRODUCTION

1.1 This chapter states the vibration standards for rotorcraft and their equipment, defines the vibration testing of rotorcraft equipment, and states the vibration limits to safeguard personnel from undue discomfort and to enable them to perform their duties.

1.1.1 LEAFLET 501/1 INFORMATION ON ROTORCRAFT VIBRATION provides background information on the sources and nature of rotorcraft vibration.

1.1.2 LEAFLET 501/2 VIBRATION TESTING OF ROTORCRAFT EQUIPMENT gives the test schedules and methods for testing rotorcraft equipment and explains the derivation of test levels and durations.

1.1.3 LEAFLET 501/3 HUMAN EXPOSURE TO ROTORCRAFT VIBRATION gives guidance and defines limits for the effects of vibration on the human body, use of flying controls and visual performance.

1.2 THE PROCUREMENT AGENCY

1.2.1 The Procurement Agency is the body responsible for defining the task or placing the order. This body can be MOD(PE), the rotorcraft designer, main contractor, etc.

1.2.2 The Procurement Agency shall provide sufficient information on the vibration environment of the rotorcraft to enable the equipment supplier to produce an effective design for his equipment.

1.2.3 The Procurement Agency shall provide the equipment supplier with the following design data:

(i) vibration data measured in flight testing, applicable to the location of the equipment and the mission, or

(ii) where measured flight data are not available, estimated vibration data based upon experience or calculation, or

(iii) where no data are available, the data defined in Leaflet 501/2 to be used as a guide in the design

1.3 THE EQUIPMENT SUPPLIER

1.3.1 The equipment shall be designed to function and/or survive satisfactorily in the environment described in para 1.2.3.

1.3.2 The equipment supplier shall test his equipment to ensure that it shall function satisfactorily in the environment derived from para 1.2.3. The test specification used shall be agreed with the Procurement Agency.

1.4 MEASURED FLIGHT DATA

1.4.1 Vibration data measured in flight shall be considered wherever practicable to ensure that the equipment is designed to suit the environment in which it is to operate and to ensure that realistic testing is carried out. See Leaflet 501/2 for guidance.

2 VIBRATION DESIGN REQUIREMENTS FOR ROTORCRAFT

2.1 GENERAL AIRFRAME VIBRATION

2.1.1 The rotorcraft designer shall be able to demonstrate to the Procurement Agency that vibration control has been given proper consideration throughout all phases of the rotorcraft design and development. Every effort shall be made to ensure that vibration is kept to a minimum consistent with other requirements placed on the rotorcraft design.

2.2 REQUIREMENTS AT EQUIPMENT LOCATIONS

For definition of equipment classifications see Leaflet 501/2.

2.2.1 Light (General Purpose and Specific) Equipment. The rotorcraft designer shall ensure, as far as possible, that the flight vibration levels are not so excessive as to render invalid the accepted equipment qualification tests.

2.2.2 Heavy Equipment. The carriage or installation of heavy stores or heavy equipment, because of the interaction with, and modification of airframe structural responses, shall be treated as defined by Leaflet 501/2.

2.3 REQUIREMENTS AT CREW LOCATIONS

2.3.1 This para shall be read in conjunction with Leaflet 501/3 which provides detailed information and requirements for human response to rotorcraft vibration. The design aim shall be to provide a vibration environment for the crew which satisfies the specifications of Leaflet 501/3. Where required by Leaflet 501/3 relevant accelerations shall be measured, the required data analysis carried out and the results reported to the Procurement Agency.

2.3.2 Consideration shall also be given to design factors given in Annexes B and C of Leaflet 501/3 to reduce the effects of vibration, on the ability of the crew to operate the rotorcraft's controls and equipment.

3 INTERNAL NOISE DESIGN REQUIREMENTS FOR ROTORCRAFT

3.1 GENERAL

3.1.1 The rotorcraft designer shall be able to demonstrate to the Procurement Agency that noise control has been given proper consideration throughout all phases of the rotorcraft design and development. Every effort shall be made to ensure that internal noise is kept to a minimum consistent with other requirements placed on the rotorcraft design.

3.2 REQUIREMENTS AT CREW AND PASSENGER LOCATIONS

3.2.1 This para shall be read in conjunction with Chapter 115 which provides specifications and evaluation procedures relating to human exposure to rotorcraft internal noise.

3.2.2 The design aim shall be to provide a noise environment for the crew and passengers which satisfies the requirements of Chapter 115. Where required by Chapter 115 the relevant noise levels shall be measured, the required data analysis carried out and the results reported to the Procurement Agency.

4 VIBRATION TESTING REQUIREMENTS FOR ROTORCRAFT EQUIPMENT

4.1 STANDARD OF EQUIPMENT

4.1.1 The equipment subjected to final qualification shall be, in all significant respects, of the equivalent build standard to the final production standard.

4.2 TEST PROCEDURES

4.2.1 The test procedures used shall be agreed with the Procurement Agency. Preference shall be given to the use of the procedures defined in Leaflet 501/2.

4.3 EQUIPMENT TESTED TO OTHER SPECIFICATIONS

4.3.1 The equipment supplier may offer to supply equipment which has been tested to the requirements of other Specifications. Details of the tests, test methods and results shall be provided for consideration.

4.4 FAILURES IN TESTS

4.4.1 If a failure occurs during testing of the equipment the supplier shall consider the options to deal with the situation. If the fault cannot be rectified without undue cost, weight or time penalties the Procurement Agency shall be consulted. Amongst the options available are:

(i) Reassessment of the relevance of the test method and test specification,

(ii) Feasibility of limited acceptance,

(iii) Reposition of the equipment within the rotorcraft,

(iv) Redesign of the equipment or its mountings.

If a retest is agreed to be necessary the details of the retest shall be agreed with the Procurement Agency.

LEAFLET 501/1

VIBRATION AND INTERNAL NOISE

INFORMATION ON ROTORCRAFT VIBRATION

1 THE SOURCES AND NATURE OF ROTORCRAFT VIBRATION

1.1 In terms of its vibration characteristics, the most important feature of the rotorcraft's behaviour is that its vibrations occur at very specific frequencies, which are directly related to the speed of rotation of the major dynamic components. Because the speed of the rotor is closely governed throughout the flight envelope (typically within 2%) these vibration components have practically constant frequencies.

1.2 In general, rotorcraft vibration can be classified as either:

(a) generated from components within the structure, or

(b) rotor system generated.

Under class (a) it is clear that numerous components within the rotorcraft are generators of vibration, e.g., oil pumps, gearboxes, shafts etc. It is natural that the vibration characteristics measured at any point within the rotorcraft should contain elements from all these sources, the relative magnitude depending largely on the length of the path from the generator and the magnitude of the source. Within this class, the mechanism for the generation of vibratory loads is often a function of the standard of manufacture and subsequent maintenance. The balance standard of shafts or wear of bearings are examples.

1.3 However, what distinguishes the rotorcraft from many other types of flight vehicles is that the levels of vibration from the sources discussed above, although significant, are generally small compared with those generated by the rotor system.

1.3.1 The rotor system has inherent vibratory load generating characteristics which are a function of the design of the rotor. In forward flight, the main aerodynamic lift surface of the rotorcraft (the rotor blade) experiences periodic changes in free-stream air velocity (due to its rotation and the forward velocity of the rotorcraft) and angle of attack (due to the application of cyclic pitch). This generates large periodic air loads with a spectrum of frequencies which are multiples of the blade's rotational speed.

1.3.2 Since the blade has its own dynamic characteristics which are a function of its mass and stiffness distribution together with centrifugal effects, it responds to these loads as a forced dynamic system, generating shear forces and moments at the hub centre-line. Because of the symmetry of the rotor system, the net load resulting from all the blades is such that many of the harmonics cancel across the rotor system. This cancellation is such that the net excitation applied to the fuselage will be only at frequencies equal to an integer multiple of the number of rotor blades multiplied by the rotational speed of the rotor. Thus, for a rotor system with 'n' blades the structural excitation will only occur at frequencies of n, 2n, 3n, ..., etc., per rev of the rotor. In practice, the magnitudes of these loads diminish significantly with increasing frequency and in general the most important source of vibratory loading for the rotor is at a frequency of n/rev.

1.3.3 Thus in nearly every case, the rotorcraft vibration environment contains a relatively large component from the rotor system at n/rev plus a mixture of frequencies at much lower amplitudes from all other sources in the rotor system and airframe.

1.4 For various rotorcraft the frequency content and sources are typically as follows:

0-3Hz	Ride motions and turbulence effects.
3.5-7Hz	Main rotor rotational frequency (1/rev called 1R or 1P or 1)
5-8Hz	Fuselage-bending vibration modes which may be briefly and intermittently excited by turbulence.
6-20Hz	External stores on carriers are sometimes excited by turbulence at their natural frequencies in this range.
11-26Hz	Main rotor blade passing frequency (n/rev. called 4R for four-bladed rotorcraft). For example: Sea King (5R) 17.4Hz Lynx (4R) 22Hz Gazelle (3R) 19Hz Scout/Wasp (4R) 26Hz Wessex (4R) 15.4Hz This is usually the predominant frequency in the vibration spectrum.
20-80Mz	Tail rotor 1/rev frequency (called IT), multiples of main rotor frequency, tail drive-shaft frequencies, pump frequencies, some gearbox shaft rotational frequencies.
100-140Hz	Tail blade passing frequencies, Pump frequencies, gearbox and output shafts.
350-700Hz	Final-reduction gear-meshing frequencies.
500-1000Hz	Hydraulic pump piston frequencies.
700-10kHz	Other gearbox meshing frequencies.
10kHz+	Engine turbine blade-passing frequencies.

The amplitudes of the main rotor blade-passing components of the vibration vary according to the flight condition. During hover and mid-speed flight the levels are generally low. The levels increase at higher forward speeds and at very low speed and during some manoeuvres. Generally, the highest vibration levels occur for periods of a few seconds during transition between forward flight and hover.

1.5 Lightweight pieces of equipment generally have little effect on the airframe vibration, i.e., the vibration levels of the equipment mounting can be assumed to be the same as those occurring at the mounting point on the airframe with the equipment not fitted. For large external or internal stores whose weight is a significant proportion of the rotorcraft's weight, the addition of the store has a fundamental effect on the rotorcraft vibration by significantly altering the natural frequencies and mode shapes of the rotorcraft. Store installation designs cannot be considered separately from the rotorcraft and mounting systems must be designed with care to prevent any adverse effects on airframe vibration and to ensure the store has a satisfactory vibration environment.

2 ROTORCRAFT DESIGN FOR MINIMUM VIBRATION

2.1 It is neither practical nor desirable to specify particular approaches to be followed in order to minimise vibration. The final design of any rotorcraft will be a compromise which will reflect the role for which it is designed. A long range VIP transport will be expected to have low vibration levels whereas agility will be more important for an anti-tank machine. Their respective rotor systems would be designed accordingly. Furthermore, the continuing need to improve rotorcraft vibration levels and the advance of technology will in time lead to the adoption of new approaches. It is possible to give the following advisory paras, but the list is not exhaustive.

2.1.1 Rotor System. Vibratory inputs to the airframe can be minimised by giving careful attention to the selection of the number of blades, to placement of blade vibration modes in the frequency range and to rotor head design - in particular the flapping hinge stiffness and offset from the rotor hub centre.

2.1.2 Airframe Design. Response to rotor forces may be minimised by application of knowledge of the frequencies and shapes of the predominant vibration modes. Structural modifications may be devised to lessen vibration by moving natural frequencies away from the major forcing frequencies (especially nR). The necessary information may be obtained from a mathematical model or an experimental modal analysis.

2.1.3 Vibration Absorbers. Absorbers may be applied at the rotor head to reduce forcing inputs to the fuselage or mounted on the airframe to improve specific regions. Spring-mass absorbers and, in the rotating system, bi-filar pendulum absorbers are commonly employed.

2.1.4 Fuselage Isolation. Active or passive systems may be devised to isolate the airframe from the main rotor vibratory forces. Passive systems require the insertion of 'soft' or anti-resonant mounting between the rotor and fuselage. Active systems employ feedback systems to operate actuators in an attempt to cancel vibratory forces.

2.1.5 Higher Harmonic Control of Blades. Main rotor blade pitch is varied sinusoidally at once per rotor rev in order to achieve forward flight. Higher harmonic control superimposes pitch inputs at multiples of 1R onto each blade to cancel or reduce the blade passing forcing at source.

LEAFLET 501/2

VIBRATION AND INTERNAL NOISE

ROTORCRAFT EQUIPMENT VIBRATION TEST EQUIPMENT

1 INTRODUCTION

1.1 OBJECTIVE

1.1.1 The objective of this leaflet is to provide a means whereby it is possible to describe vibration test specifications for rotorcraft equipment, including externally carried items. It does not include equipment transported by rotorcraft.

1.2 LAYOUT OF LEAFLET

1.2.1 In order to achieve this objective a number of aspects need to be considered and these are dealt with under the following subject headings:

(i) PARA 2: Environmental Management Plan. Defines the aspects of management and the management policy which are essential to the overall vibration programme.

(ii) PARA 3: Vibration Testing Philosophy. Provides background information on various aspects of vibration testing.

(iii) PARA 4: Detailed Test Specifications. Details the procedure to be followed to ensure that testing relates to the conditions which prevail both for a particular rotorcraft type and in the general case.

(iv) ANNEX A: Background Information on Values given in Tests for Categories A and D Equipment.

(v) ANNEX B: Vibration Test Procedure.

(vi) ANNEX C: Data Acquisition and Analysis for Test Derivation.

2 ENVIRONMENTAL MANAGEMENT PLAN

2.1 GENERAL

2.1.1 For rotorcraft, the design of internal and external equipment and any externally carried stores is dependent on many factors including the definition of the environment to which the items will be subjected. To ensure that all environments are considered, it is essential to produce an Environmental Management Plan which takes into account all phases from manufacture to operation. The Plan will also include the associated interpretation of these requirements into design, testing and acceptance specifications. Adherence to this Plan should confirm to both the customer and the supplier that all aspects of the Environment have been satisfied. Flight vibration is but one of the environments which need to be taken into account.

2.1.2 It is important that where more than one party is responsible for the total assembly of the hardware, e.g., rotorcraft and its external weapon, the Plans of the respective parts are compatible. This will be the responsibility of the customer's overall project manager.

2.1.3 A successful project will involve a number of other management plans covering its financial, performance, trials aspects etc. It will also encompass many inter-related technical documents which will need careful integration e.g., the environmental verification trials must be linked with the performance proving trials. It is the project manager's responsibility to ensure that the various plans are combined and, as such, it is his responsibility to seek the advice of the related specialists at an early stage of the project development.

2.1.4 The extent of the Plan will be determined by the hardware under consideration. For instance, for external stores it will be comprehensive but for single items of equipment it will be much simplified. In principle, the following is applicable to all hardware.

2.2 THE PLAN

The Plan will need to include many aspects, including the following:
2.2.1. Overall Policy. This will outline the contractual requirements associated with topics such as:

(i) Related documents e.g., DEF-STANS, AvPs

(ii) Procedures to ensure visibility to both the supplier and the customer that the design of the hardware is progressing acceptably against the environmental design and load descriptions.

(iii) The differences between the equipment supplier who is supplying items that are specific to the one rotorcraft type or the supplier who is providing general equipment suitable for all rotorcraft.

(iv) The interaction between the supplier of an externally carried store and the rotorcraft manufacturer. This is particularly relevant when the store's mass is significant compared to that of the rotorcraft.

2.2.2 Environmental Description. This will include the environment experienced by the rotorcraft airframe, the environment which is modified by the airframe and experienced by the equipment attached to it, and the possible further modification of this environment by the effects of relatively large masses carried by the rotorcraft.

2.2.3 Design and Load Descriptions. Based on the environmental description, the design and load criteria will be formulated which will contain quantitative load and design information. Aspects such as the relative effects of equipment isolators and their influence on the equipment loadings will be defined.

2.2.4 Testing. The testing document will include the rationale for the interpretation of the environment into a series of ground tests. These ground tests will be used for the purpose of design verification and the proving and approval of the hardware against the environmental loads.

2.2.5 Environmental Verification. The initial environmental description will normally be based on preliminary information and, as such, a detailed programme of trials may be necessary to improve confidence in the environmental description. This process will be iterative as the hardware design develops and should be clearly visible to the customer in order to ensure confidence in the design.

2.2.6 Assessment Policy. This will include the procedure for carrying out assessments, which should be stated at an early stage of the design. It will cover such topics as:

(i) Methods to be used for the verification of the environment.

(ii) Criteria for the comparison of flight and ground test results.

(iii) Relative role of paper assessments.

(iv) Criteria for assessing 'failures' occurring during both ground and flight tests.

(v) Criteria to be used for the assessment of the hardware design for approval purposes.

(vi) Assessment of the validity of any mathematical modelling procedures used in the design procedure.

3 VIBRATION TESTING PHILOSOPHY

3.1 INTRODUCTION

3.1.1 This para provides a general philosophy for the vibration testing of equipment installed in or on rotorcraft. The tests and techniques adopted will vary according to the type of equipment and its installation, its position in the rotorcraft relative to the excitation and vibration transmission paths, and the purpose for which the tests are required.

3.2 THE DYNAMIC ENVIRONMENT

3.2.1 The dynamic environment is characterised by the excitation sources acting upon the airframe, as described in Leaflet 501/1. The environment may, nevertheless, be modified in terms of level and spectral content by the presence of equipment installed in or on the airframe if the mass and/or dynamic properties of the equipment are such that it interacts significantly with the structure. If such an item is to be representatively tested it may be necessary to simulate the installed mounting details in order to preserve the correct specimen behaviour.

3.3 TYPES OF TEST

3.3.1 Apart from testing activities which may be performed for design confidence purposes there are, essentially, two main formal categories of test:

(i) Experimental Flight Clearance. Experimental flight clearance tests are often carried out at an early stage of hardware development. They may only be required as evidence of safety or functioning in a limited environment and may, in the absence of definitive environmental information, comprise no more than a simple ruggedness or 'shakedown' test.

(ii) Qualification or Type-Approved. The scope of these tests is usually governed by a contractual requirement and these cover extremes of the service conditions under which it is necessary to demonstrate compliance with the design requirements.

3.4 TESTING REQUIREMENTS

3.4.1 It is virtually impossible to simulate all aspects of a service environment and, in this respect, few vibration tests could be considered to be realistic. However, though complete realism is not essential for a test to be useful and appropriate, it is important that a reasonable degree of evidence is available to show its acceptability.

3.4.2 The Purpose of the Test. Testing in the context of this Leaflet may be considered as demonstrations of adequate robustness and proper functioning. The test durations required for these two categories reflect the philosophies of the tests.

(i) Robustness tests are carried out to demonstrate the adequacy of the basic structure and joints, component mounting fixtures, electrical connectors, etc., for the designed life, which may be several thousands of hours. As it is generally impracticable to test equipment for such long durations, it is common practice to test for much shorter times e.g., 50 hr.

(ii) Functional tests should be carried out when it is required that the equipment be capable of proper functioning in the vibration environment. In such cases, the duration is normally set by the time necessary to make the required measurements.

3.5 VIBRATION RESPONSE INVESTIGATION (RESONANCE SEARCH TEST)

3.5.1 A resonance search test is primary associated with robustness and is required in order to establish the response characteristics of the equipment and to identify resonances within the equipment which could lead to early failure. This is particularly important for rotorcraft equipment because of the nature of the excitation.

3.5.2 The test is normally carried out in the laboratory at low excitation levels in order not to damage, or reduce the life of, the item. If, however, the response frequencies are found to vary with input level, then the search must be carried out at representative excitation levels. For items which are large and dynamically complicated, it may be necessary to perform this test with the item fastened to a representative portion of its mounting.

3.5.3 In most instances it is also necessary to carry out Installed Resonance Frequency checks, see para 4.5.4.

3.6 THE SPECIFICATION

3.6.1 Much will depend upon whether the rotorcraft, for which the equipment is destined, is a new design when the environment may only be predicted; a modified design for which some information will be available; or an established design, for which the vibration environment is known. Whenever practicable, the vibration test levels should be based on measured levels from the rear environment. It no directly relevant information exists, then levels must be extrapolated from such information as is available, or by reference to generalized specifications. Generalized specifications should not be used without a full understanding of the way in which they were derived and the extent of their applicability to a particular case. In certain instances and for particular classes of equipment, it may be more cost-effective to use generalized data, see para 4.6.2. Engineering judgement is often necessary in interpreting the requirements of a specification in relation to the application. In formulating a specification it is necessary to consider the following paras.

3.6.2 Factoring. Results obtained from flight analysis are nominal results. They may be expected to be affected by a number of parameters, e.g., rotorcraft variations, both between and within types, carriage permutations in the case of external stores, manufacturing tolerances of the equipment under investigation, trials conditions, etc., etc. The influence of these parameters should be taken into account at the assessment stage to give confidence that the equipment is adequately tested. The extent of factoring is basically a matter for engineering judgement. The factor applied will be a compromise between gaining sufficient confidence in the equipment under test and avoidance of an overtest situation. The magnitude of a factor should be influenced by past experience with similar equipment and the cost, in broad terms, of failure, both during testing and in service.

3.6.3 Specimen Mounting.

(i) For large dynamically complicated specimens, the mounting characteristics will have a significant effect on the specimen behaviour. In principle, the test mounting arrangement should simulate the dynamic characteristics of the service mounting over the frequency range of interest, in all degrees of freedom. Whilst total simulation is rarely possible, however, sufficient attention must be given to the mounting in order to avoid grossly unrepresentative responses of the specimen.

(ii) For small specimens which are dynamically simple or which are sensibly inertial, the mounting is less likely to significantly influence the specimen response. Such specimens are usually mounted direct to a vibration generator.

(iii) Equipment which is normally mounted on vibration isolators should, wherever practicable, be tested on its isolators. Where this is not practicable, account must be taken of the isolator characteristics when determining the input test levels.

3.6.4 Excitation. For the majority of vibration tests, excitation of the specimen is provided by electrodynamic or electro hydraulic vibration generators using sinusoidal, broad band random or narrow band random vibration sources or combinations of these. Force hammer and random excitation methods are frequently used in conjunction with modal analysis systems where modal information is required. Normally the excitation should be applied in the same manner as in service, as far as this is practicable. The number of input points and directions are normally dictated by the equipment mounting.

3.6.5 Special Considerations. There are two aspects of a test requiring particular consideration:

(i) Synergistic Environments. These are environments of combined parameters which produce, in the specimen, a different response to that which would arise if the environments acted separately. A specimen may perform satisfactorily under each environment but fail when they act simultaneously. Examples of a synergistic environment are a vibration response that may be dependent upon the specimen temperature, or where a steady acceleration is combined with vibration. The affects of combined environments must be considered when formulating the test specification.

(ii) Accelerated Testing. This term is commonly used to describe testing in which the duration is deliberately shortened to make the test programme more economical in terms of time, cost or both. There are two ways in which a test can be defined as 'accelerated' and it is important to distinguish between them:

(a) The first is one in which continuous testing is used to represent the summation of a series of discrete exposures in the service environment which may only occur during part of a mission and which is taken as the most severe case. This is acceptable generally although careful interpretation is necessary. The levels do not exceed the normal maxima and parts of the mission in which the levels are not considered significant are neglected.

(b) The second is one in which the normal service excitation levels are deliberately exceeded in exchange for a reduction in test duration. This form of testing is not acceptable without considerable qualification. Particular consideration must be given to the nature of the specimen, the materials involved, likely synergistic effects and the functioning requirements. It must also be recognised that unrealistic environmental stress levels may excite failure modes which do not exist at normal levels e.g., collision due to loss of clearances, premature failure, fretting, temperature effects. Clearly, limits must be placed upon the maximum levels if the test is to remain representative.

4 DETAILED TEST SPECIFICATIONS

4.1 GENERAL

4.1.1 It is necessary to consider the whole range of environments associated with the life cycle of the rotorcraft and its associated stores/equipment and to plan tests accordingly. For maximum cost-effectiveness this must be done at the commencement of the development programme. However, at that stage, particularly with the vibration environment, there may be little or no data available. In addition, the type of store can have a significant affect on the environment and different types of test exist for different purposes. Whilst this may make the production of the test specification difficult, it is, nonetheless, vital that the task be undertaken.

4.1.2 All of these factors have been discussed earlier in this leaflet but are dealt with in more detail below. Rotorcraft and equipment are divided into three classifications. In addition, a zoning concept and a method for generalizing test data in the absence of actual measured data are given. The possible combinations are listed in tabular form in para 4.6.2. From all of this vibration test specification can be found or derived.

4.1.3 Some background information is given in Annex A to this leaflet, and a flowchart illustrating the whole clearance procedure for equipment is described in para 4.8.

4.2 CLASSIFICATION OF ROTORCRAFT

4.2.1 NEW. A rotorcraft whose major airframe and rotating systems have not seen previous applications. Initially, the rotorcraft design authority will be able to accurately define only the significant vibration forcing frequencies. Estimates of levels will be based on calculations or previous experience with components of similar size, speed etc., where available. Preliminary design/test specifications will be based on this data or on guidelines given herein.

4.2.2 MODIFIED. A rotorcraft with some or many components taken from existing, known machines. It should be possible to describe much of the vibration environment. The uncertainty will depend on the degree of modification, but it should always be possible to define frequency content. For example, internally induced vibrations on a gearbox casing will tend to differ little with application, whereas a new tail structure may modify rotor order vibrations and stresses throughout the airframe.

4.2.3 EXISTING. A rotorcraft for which a library of measured data exists. It should be possible to define equipment design and test specifications with confidence. Care must be taken with equipment of large mass which may interact with and modify structural vibration responses, particularly at rotor order frequencies. In case of doubt reference should be made to the rotorcraft design authority.

4.3 CLASSIFICATION OF EQUIPMENT

4.3.1 LIGHT, GENERAL PURPOSE. This is equipment which is dynamically simple or sensibly inertial and which is small in relation to the area of those rotorcraft in which it will be mounted. There should, therefore, be no significant modification of airframe structural response. This equipment can be designed for general rotorcraft applications.

4.3.2 LIGHT, SPECIFIC. As the 'Light, General Purpose' class but destined for one application only. It should be noted that, in most cases, for equipment which can be classified as 'Light', the most effective course of action will be to regard it as 'General Purpose' and test in that manner. Only rarely is it expected that the design will be so critical that the severity of the generalized curves applicable to 'Light, General Purpose' equipment will be too high. Where there is doubt the rotorcraft design authority should be consulted.

4.3.3 HEAVY. This is large, often complicated, equipment such as external stores and radar scanners, whose mass, dynamic properties and mounting arrangements will be such as to interact with and modify airframe structural responses, particularly at rotor order frequencies.

4.4 CLASSIFICATION OF ZONES

4.4.1 OBJECTIVE. In order to simplify the presentation of data, an attempt has been made to divide a typical rotorcraft structure into zones, see Fig. 1. This is intended as a guide and should be used in the absence of more specific knowledge about the ultimate installation position of a store/equipment.

(i) MAIN FUSELAGE (N). Main structure, not adjacent to engines or gearboxes.

(ii) PERIPHERAL STRUCTURE (V). Areas of the rotorcraft structure away from the main fuselage, such as tail fins, tail planes and undercarriage sponsons.

(iii) ENGINES/GEARBOXES (W). All areas where the vibration environment will include significant contributions from engine, gearbox or hydraulic pump sources.

(iv) HEAVY EQUIPMENT (X). As for para 4.3.3, these are treated separately.

4.5 TESTS

4.5.1 A range of tests is given below and in paras 4.6 and 4.7 which includes those performed on the ground and in the air. Different situations will require different selections to be made from those defined. Generally, some proportion of the vibration response investigation will always be done in conjunction with the Qualification endurance tests. It should rarely be necessary to perform flight measurements for 'Light' equipment, see para 4.7. Deviations from, and additions to, the tests listed may be appropriate but should be agreed with the relevant design authority before implementation.

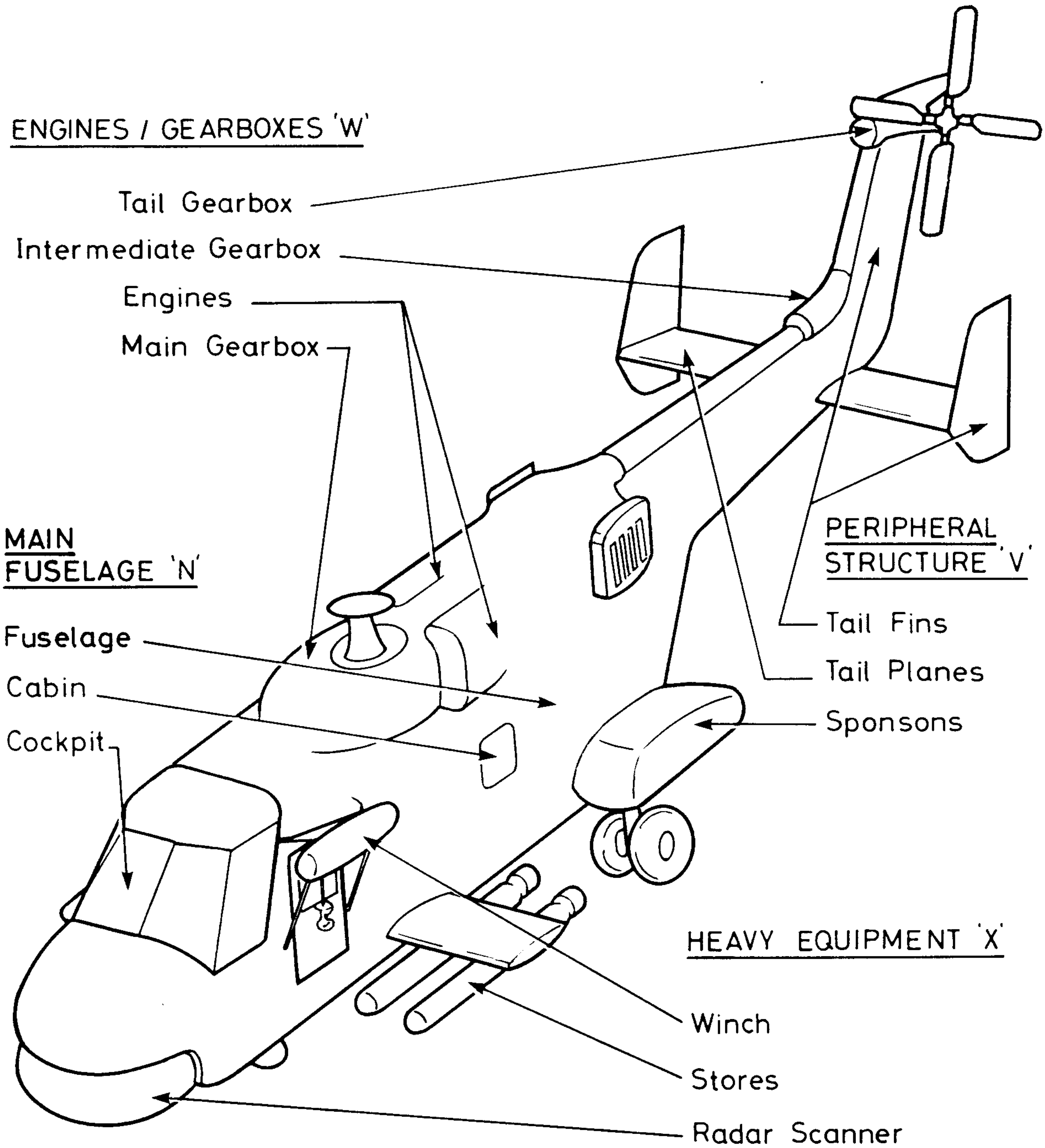


FIG. 1 ROTORCRAFT EQUIPMENT ZONES

4.5.2 Laboratory Vibration Response Investigation (Resonance Search Test). A response investigation must be included in any equipment clearance programme. Responses of major items within the equipment should be monitored over the appropriate frequency range, see Figs. 2, 3 and 4. An appropriate frequency range for 'Heavy' equipment is as shown in Figs. 2 and 3. Coincidence or near coincidence with known significant rotorcraft forcing frequencies should be avoided as far as is practicable. The test procedure to be used is given in Annex B to this leaflet, and has been extracted from BS 2011 Part 2.1 Fc:1983 Vibration (sinusoidal).

4.5.3 Experimental Flight Clearance 'Ruggedness' Tests. Laboratory tests are required on equipment in order to provide the necessary confidence for flight tests. These should follow the procedure given in para 4.6 for CATEGORIES A, B or D, as appropriate, except that the duration shall be for 6 hours divided equally between three mutually perpendicular axes. However, instead of this, it may be convenient to carry out the full Qualification Endurance Test before flight.

4.5.4 Installed Resonance Frequency Checks

(i) Introduction. Normally, checks should be made on the resonance frequencies of equipment when installed in the rotorcraft. The aims of these tests are:

(a) To establish whether there are response peaks near the rotorcraft forcing frequencies which are significant in the area where the equipment is mounted.

(b) To gain information on the mode shape or deformation pattern associated with each main resonance frequency.

For Zones 'N' and 'V' , Installed Resonance Frequency checks may not be required where small equipment e.g. circuit breakers, switches, instruments, is rigidly mounted and it can be confidently predicted, or it has been demonstrated, that there are no natural frequencies below approximately 100 Hz. The results of these tests should be considered when deciding whether flight measurements are necessary for light equipment installations. Even when potential problems are identified, either during Vibration Response or Installed Resonance Frequency checks, it is often worthwhile assessing the severity of the problem in flight before embarking on a modification programme which might be difficult, costly and time consuming.

(ii) Simple 'Bonk' or 'Rap' Tests. In this test, the transient response of the equipment to an impact is monitored by a suitable sensor, usually an accelerometer, whose signal is fed to a spectrum analyser for the extraction of the major response frequencies. Tests are carried out in each degree of freedom e.g., lateral, vertical, fore and aft. This is usually adequate for simple Light equipment installations.

(iii) Modal Analysis. For the installation of heavy equipment, or where problems are encountered with Light equipment, it is usually necessary to gain more information than is provided by the simple 'Bonk' or 'Rap' test. Transfer functions i.e., response/force input are measured so that mode shape data can be derived by comparing the amplitude and phase of responses at different points. The force input can be applied using either an impact device or an electrodynamic shaker, each incorporating a suitable force sensor.

(iv) Frequency Ranges. The analysis range for installed resonance frequencies checks should include the significant forcing frequencies present where the equipment is located. The rotorcraft design authority should be consulted concerning which specific forcing frequencies are likely to be present. The following are suggested as a guide:

(a) Zones 'X', 'N' and 'V'

A frequency range of 3-200 Hz, but with particular attention being paid near rotor order forcing frequencies between 3 and 60 Hz.

(b) Zone 'W'

There are likely to be significant forcing frequencies up to 2500 Hz and possibly well beyond this frequency. However, the interpretation of resonance frequency test results is difficult at high frequencies and firm conclusions can rarely be drawn except when a prominent response is seen at, say, a major mesh or hydraulic pump frequency.

4.6 QUALIFICATION ENDURANCE TESTS

4.6.1 The following sub-paras deal with the selection and/or derivation of appropriate test spectra and levels. However, it must be noted that, where existing equipment is being cleared for a new application, it may be possible to show that previous documented testing is adequate for the new installation. The vibration levels given are those at the equipment/airframe interface i.e., at the airframe side of any vibration isolators which may be used. Endurance testing should be carried out with the correct isolators where appropriate, but if this is not possible the specified levels should be modified according to Fig.5. The test procedure to be used, in conjunction with these levels, is given in Annex B to this leaflet, and has been extracted from BS 2011 Part 2.1 Fc:1983 Vibrational (sinusoidal) and BS 2011 Part 2.1 Fd:1973 Random Vibration - wide band general requirements.

4.6.2 Selection of Test. The appropriate test should be selected from the data given below:

Equipment Rotorcraft	Light General Purpose	Light Specific (1) Purpose	Heavy Equipment
	CATEGORY	CATEGORY	CATEGORY
New	A	D	D→C
Modified	A	D	D→C(2)
Existing	A	B	C

Notes: 1 Particular consideration should always be given to the issues stated in para 4.3.2.

2 The CATEGORY D procedure should always be used for the experimental Flight Clearance 'Ruggedness' tests. The Qualification Endurance Test should be based on flight measurements as described for CATEGORY C.

4.6.3 Description of Test Spectra. The test spectra are described below. Additional guidance for CATEGORIES B, C and D can be found in Annex C to this Leaflet

(i) CATEGORY A. Light, General Purpose Equipment. These are broad-band random vibration curves to include representative contributions from rotor order vibrations at low frequencies and, for Zone W, engine/gearbox induced vibrations at high frequencies. The levels are for use in the clearance of Light, General Purpose equipment to be used in a variety of rotorcraft types, see Figs. 2,3 and 4. The test duration shall be 50 hours divided equally between three mutually perpendicular axes.

(ii) CATEGORY B. Light, Specific Equipment. For this category of equipment it is important to note the comments given in para 4.3.2. If nonetheless, it is considered essential to derive tailored spectra, then the exact details must be agreed with the rotorcraft design authority and need to take account of the following:

(a) Test spectra will probably be of the form of multiple sine, narrow band or wide band random or occasionally a single sine content. Appropriately band limited random vibration may be applied as a practical representation where there are a large number of sinusoidal functions of similar magnitude in measured spectra.

(b) The frequencies of the sine waves, or the centre frequencies of narrow band random inputs, will be those generated by the predominant rotating components in the rotorcraft.

(c) It may be considered appropriate to sweep sine wave frequencies by 5% to cover variations in flight and in equipment.

(d) Test frequency ranges should be based on measured data, tempered by any knowledge of the equipment's susceptibility to particular frequencies etc. The following are included as guides:

Zone N	:	10-500 Hz
Zone V	:	10-500 Hz
Zone W	:	10-2500 Hz

(e) The selection of test levels is a complicated subject but it should be noted that the variation of flight condition tends to have most effect on rotor order vibration levels i.e., nR and 2nR,
where n = number of main rotor blades
R = main rotor rotational frequency

(f) If a specimen resonance is unavoidably close to a major forcing frequency, it should be tested at its resonance frequency in order to cover the worst case with production variations.

(g) The use of 'mission profile' data i.e., time spent in each flight condition, to derive complex simulations is usually impracticable. It is suggested that note is taken of vibration in the high-speed cruise, where rotorcraft spend much of their time, but that allowance is made for higher rotor order levels experienced briefly during transitions from forward flight to the hover. The test duration shall be 50 hours divided equally between three mutually perpendicular axes.

(iii) CATEGORY C. Heavy Equipment.

(a) An iterative process is required when Heavy equipment is installed in rotorcraft. Generally, these items will have several installed frequencies in the frequency range 5 - 60 Hz, which includes the predominant rotor forcing frequencies.

(b) If resonances are present, it is often the case that the resulting levels are unacceptable, not only to the equipment manufacturer but also, because of the induced stresses and vibration levels, to the rotorcraft designer. Modifications will then be needed in order to improve the installation.

(c) It will be seen that whilst experimental flight clearance can be given from a simple 'robustness' test, the generation of an appropriate tailored test spectrum requires analysis of measured data from the final installations.

(d) When there are many possible combinations of stores carriage, such as with four weapon carriers or where there may be a series of variants of the same store, it must be ensured that all permutations are covered. As a minimum, each combination should be checked for coincidence with the nR frequencies by means of 'bonk' or 'rap' tests and the worst combinations should be measured in flight.

(e) The Qualification test duration should be decided through consultation between the equipment/store and rotorcraft design authorities. However, unless exceptional circumstances exist, the total test duration should not exceed 50 hours.

(iv) CATEGORY D New or Modified Rotorcraft.

(a) This covers the derivation of a vibration spectrum where the rotorcraft, or major components thereof, have not been flown or measured; indeed, they may not yet exist. Two of the early decisions when a rotorcraft is designed are the rotor diameter and hence speed 'R', and the number of blades 'n'. Rotor order forcing frequencies will, therefore, be known at an early stage. Other rotational frequencies will be calculable as soon as the transmission and tail rotor systems are designed. A frequency spectrum can then be produced.

(b) The rotorcraft design authority must make this data available and should provide assistance to stores/ equipment designers to estimate the levels of the frequencies which are expected to predominate in the particular locations. Leaflet 501/1 gives a summary of typical rotorcraft spectrum content. A preliminary test spectrum should have a broad band random vibration level with narrow band random or sine peaks superimposed, in accordance with the guidelines given in Table 1.

(c) The test duration shall be 50 hours divided equally between three mutually perpendicular axes.

4.7 FLIGHT MEASUREMENTS

It is not essential to measure vibration in flight for all equipment installations. Measurements are normally required if one or more of the following conditions apply:

(i) The rotorcraft design authority has insufficient knowledge of, and cannot confidently predict, the vibration environment at the position where the equipment is installed.

(ii) The equipment is classified as 'Heavy'.

(iii) The equipment has been previously designed and cleared to a vibration specification which, it is believed, may be significantly less severe than the rotorcraft environment.

(iv) Installed Resonance Frequency Checks or Vibration Response Tests indicate cause for concern.

Procedures for rotorcraft flight vibration measurements are discussed in Annex C.

4.8 FLOWCHART FOR EQUIPMENT CLEARANCE

A Flowchart (Fig. 6) has been produced to assist the user in the interpretation of para 4 of this leaflet. References to the appropriate paras are given on the Fig.

TEST SPECTRUM FOR LIGHT, GENERAL PURPOSE EQUIPMENT - CATEGORY 'A'

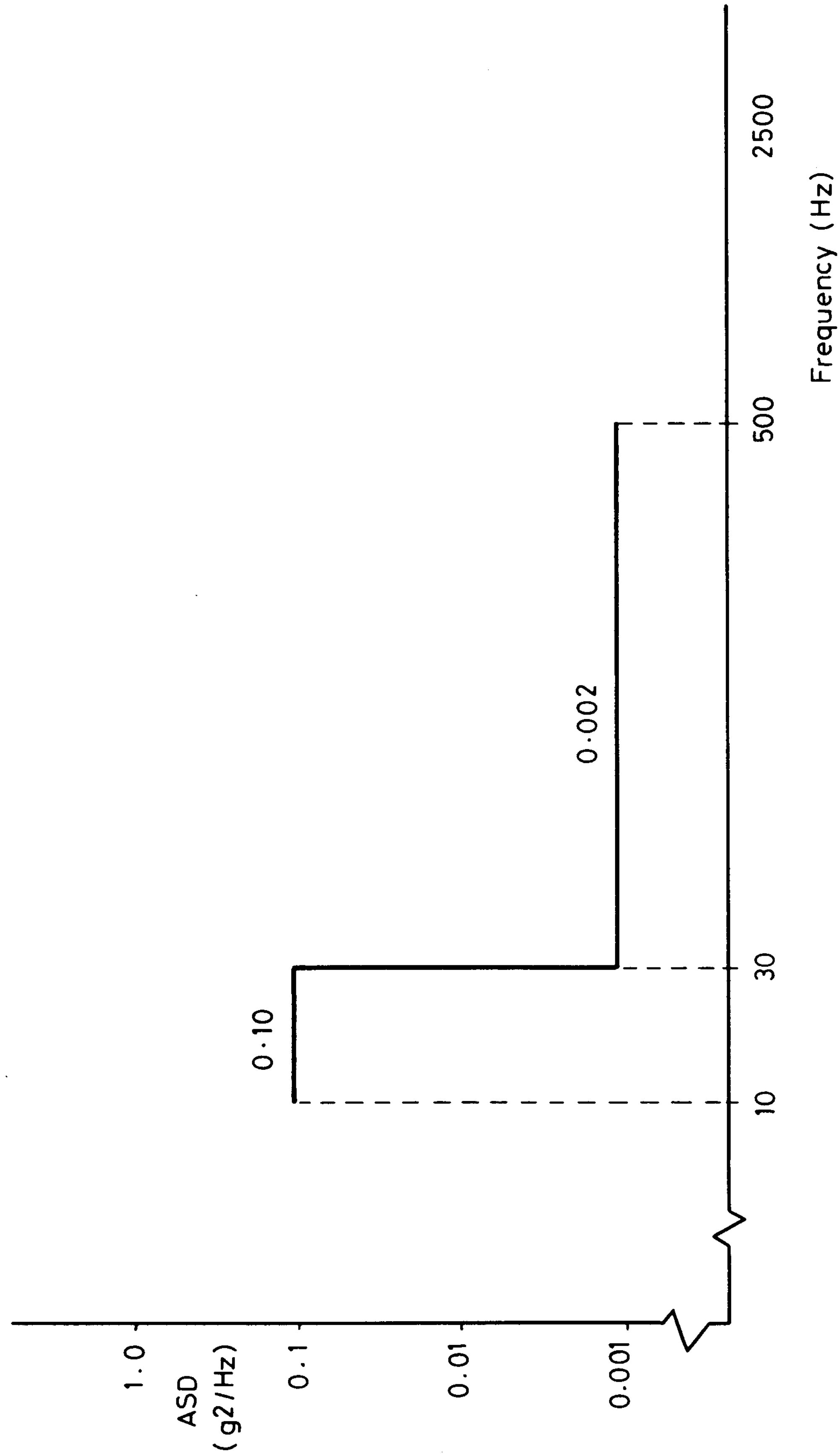


FIG.2 MAIN FUSELAGE - ZONE 'N'

TEST SPECTRUM FOR LIGHT, GENERAL PURPOSE EQUIPMENT - CATEGORY 'A'

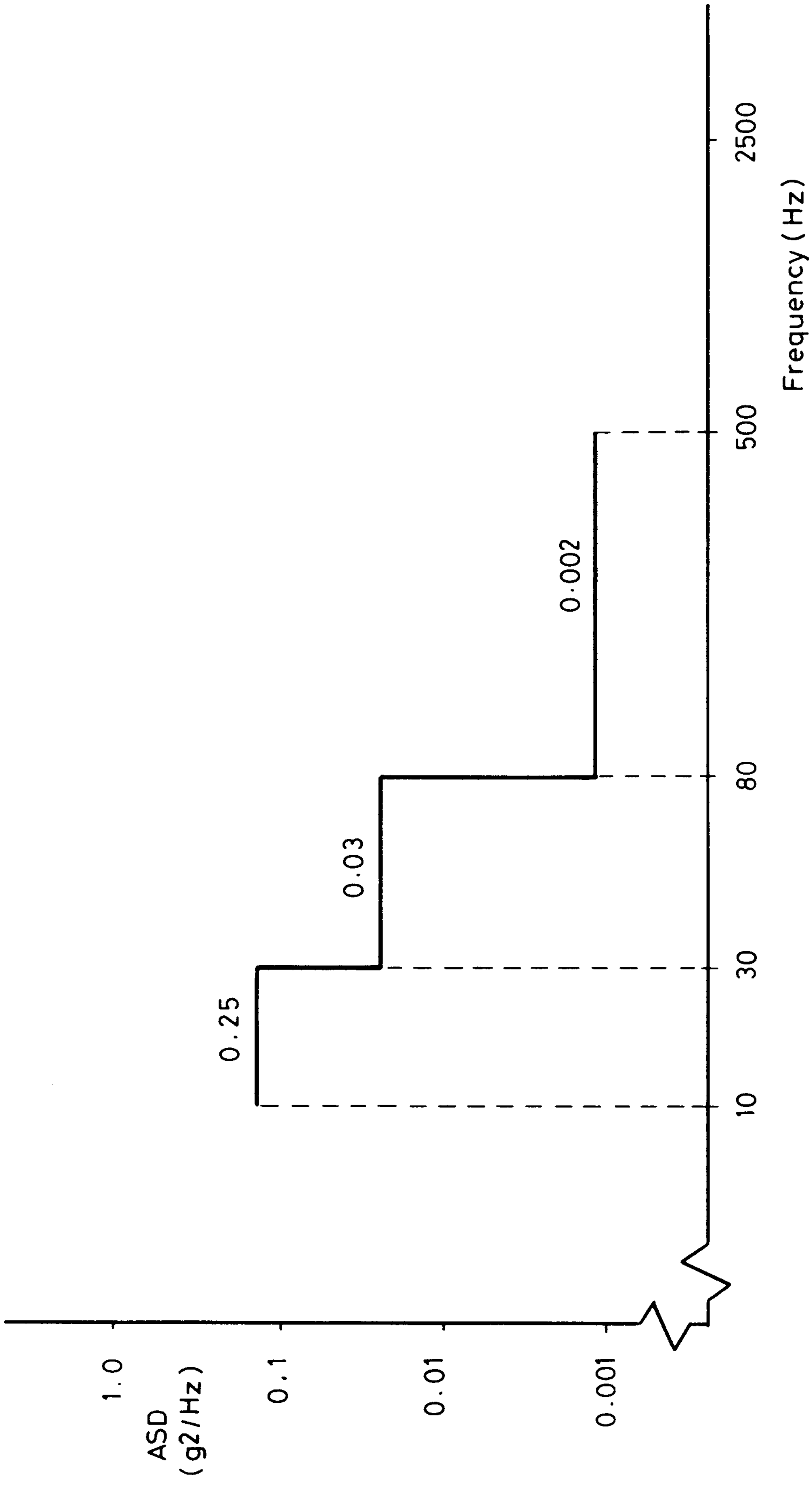


FIG.3 PERIPHERAL STRUCTURE - ZONE 'V'

TEST SPECTRUM FOR LIGHT GENERAL PURPOSE EQUIPMENT - CATEGORY 'A'

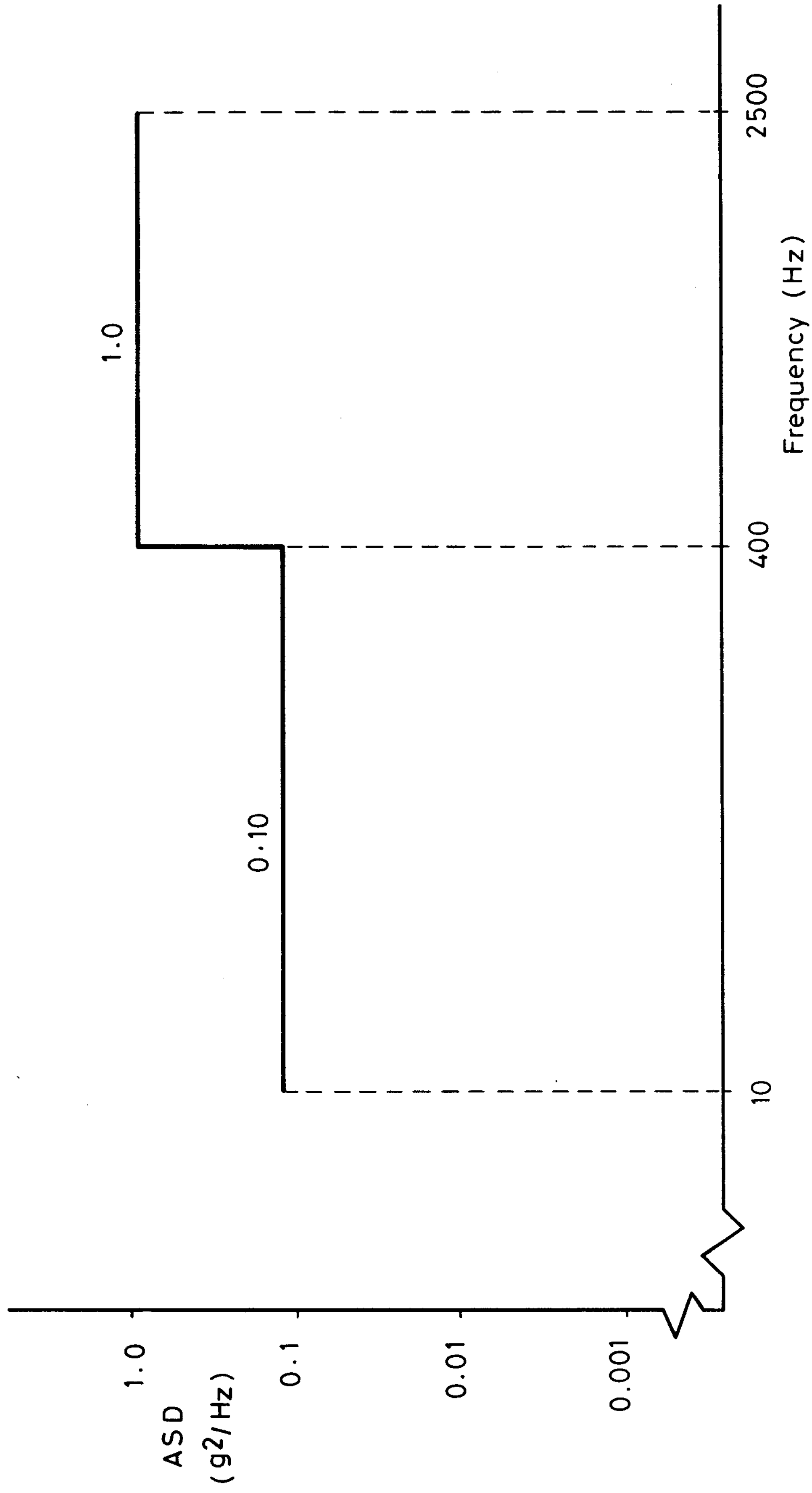


FIG.4 ENGINES/GEARBOXES - ZONE 'W'

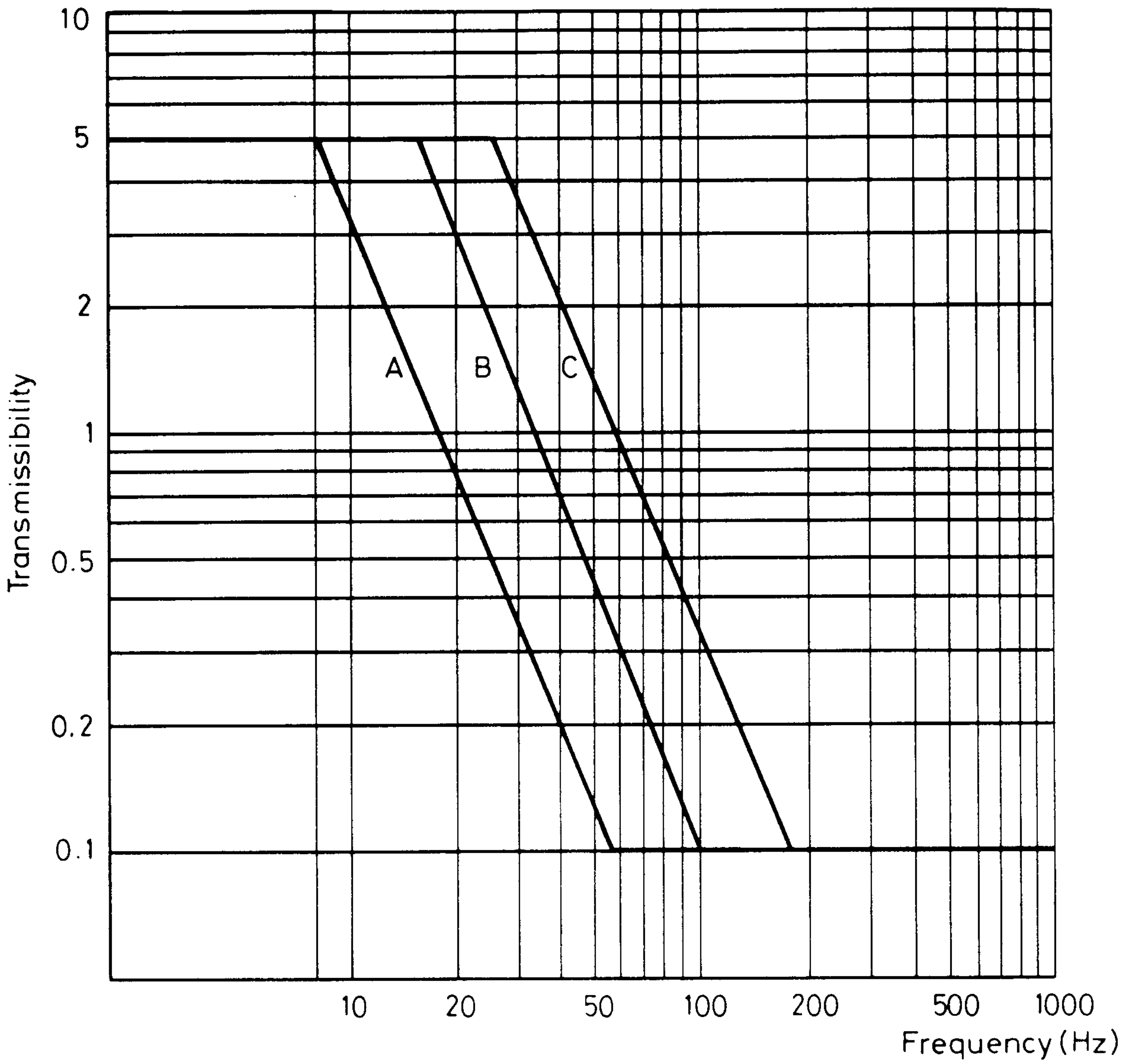


FIG 5 GENERALIZED TRANSMISSIBILITY FACTORS FOR VIBRATION ISOLATORS (REF. BS 2011)

When an equipment would normally be mounted on isolators but they are not available and their characteristics are unknown, it is necessary to modify the specified level in such a way as to provide a more realistic vibration input to the equipment. In the absence of more definitive information it is recommended that this modified level be derived by using values taken from the curves given in Fig.5 above.

NOTES 1 Curve A relates to a type of loaded isolator of high resilience having a natural frequency, when considering a single degree of freedom, not exceeding 10 Hz.

2 Curve B relates to a type of loaded isolator of medium resilience having a natural frequency, as qualified above, in the range 10 Hz to 20 Hz.

3 Curve C relates to a type of loaded isolator of low resilience having a natural frequency, as qualified above, in the range 20 Hz to 35 Hz.

Curve B is derived from vibration measurements made on typical aircraft equipment fitted with highly damped all-metal isolators having a natural frequency of approximately 15 Hz, considering a single degree of freedom. Very little data was available for mounts represented by Curves A and C. These were derived by extrapolation from Curve B, considering natural frequencies of 8 Hz and 25 Hz respectively. Similar curves may be derived for other natural frequencies.

The curves have been estimated to envelop the transmissibility characteristics likely to arise in an installation in which modes are coupled. The use of these curves, therefore, makes an allowance for vibration levels arising at the periphery of a specimen from the combined effects of translational and rotational motions. The most appropriate transmissibility curve should be selected and the specified vibration levels should be multiplied by values taken from this curve over the required frequency range.

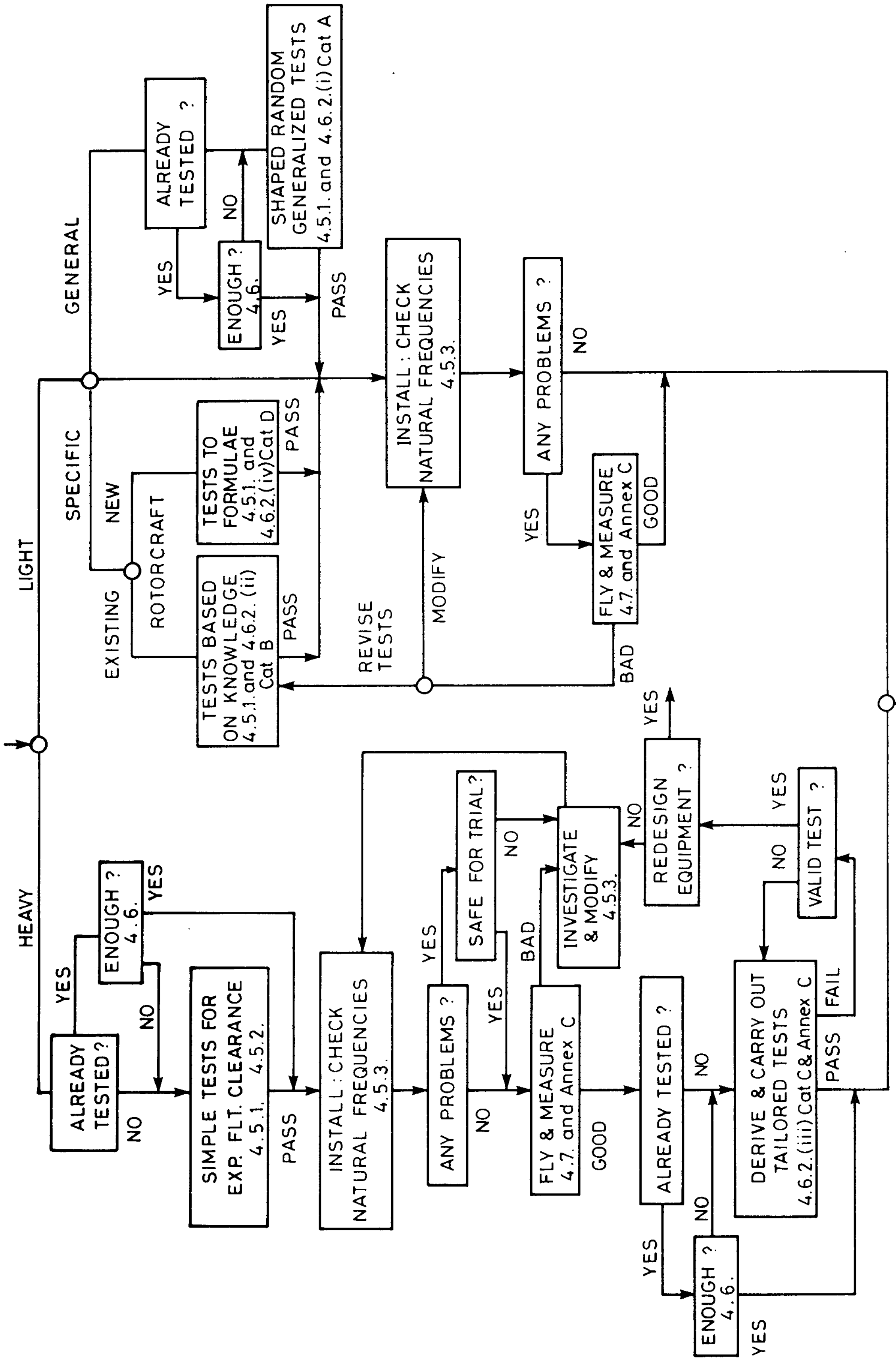


FIG.6 FLOWCHART: ROTORCRAFT EQUIPMENT VIBRATION CLEARANCE PROCEDURES

TABLE 1

Zone	Broad-band Vibration Level (ASD)	Superimposed Peaks (1)		
		Frequency f(1) (Hz)	Sine Level at 'f'	Narrow band Random (2) Vibration Centred on 'f' (rms'g')
Main Fuselage 'N'	0.002 g^2/Hz 10 - 500 Hz	nR 2nR	±g = 0.1f ±g = 0.016f	0.05 x f 0.007 x f
Engines/ Gearboxes 'W'	0.15 g^2/Hz 10 - 400 Hz	On or near Gearboxes (3) M_1) 2xM_1) M_2) (4) 2xM_2) (all 2500 Hz) On or near Engines (3) Up to 3 turbine shaft frequencies, swept over operating ranges. On or near Hydraulic Pump or Hydraulic Components (3)	±g = 5 + 0.01f ±g = 0.04f_c(5) ±30g	2.3 + 0.005f 0.02 x f_c(5) 14
Peripheral Structure 'V'	0.002 g^2/Hz 10 - 500 Hz	nR 2nR 3nR or 1T (tail area)	±g = 0.15f ±g = 0.024f ±g = 0.024f	0.07xf 0.01xf 0.01xf
Heavy Stores 'X'	0.03 g^2/Hz 5 - 80 Hz	nR 2nR 3nR	±g = 0.15f ±g = 0.024f ±g = 0.024f	0.07xf 0.01xf 0.01xf

NOTES:

1 Peaks may be applied as sine waves or narrow-band random peaks. If sine waves are applied the frequencies may be swept ± 5%, but see Note 5.

2 The narrow-band random peaks should be set as close as possible to ±5% of 'f' and the following limits must be observed.
1.02fóf max ó1.10f
0.90fó f minó 0.98f

3 There will be cases where the equipment environment has contributions from two or three sources e.g., pump and gearbox.

4 M_1 and M_2 are major gearbox mesh frequencies.

5 'f'c is the centre of the turbine shaft normal operating speed range. Turbine shaft frequency inputs should be swept over this range.

ANNEX A

BACKGROUND INFORMATION ON VALUES GIVEN IN TESTS FOR CATEGORIES A AND D EQUIPMENT

1 INTRODUCTION

1.1 The curves given in Leaflet 501/2, Figs 2, 3 and 4 are designed to reflect the rotorcraft vibration environment. The levels are derived from a survey undertaken in the U.K. but with data from British, American, French and Italian rotorcraft. Note has also been taken of values given in the appropriate parts of MIL STD 810D.

2 CATEGORY A

2.1 The predominant blade passing frequency for most rotorcraft lies between 11 and 28 Hz. The survey of nR levels at high speed cruise (Vno) for the 'Main Fuselage' areas, Zone 'N', gives values up to ±2in/s but maxima can be up to 3 times this value i.e., approximately ±6in/s or ±2.0g at the mid point of the frequency range. This gives $0.10g^2$/Hz if the energy is distributed evenly over the band adjusted, for convenience, to 10 to 30 Hz.

2.2 For 'Peripheral Structure' areas, Zone V, ±3g is a commonly used criterion which the survey confirms as being realistic. Distributed as above, $0.25g^2$/Hz is calculated.

2.3 'Engine/Gearbox' areas, Zone W, generally exhibit similar rotor order vibration levels to those at the main fuselage. On the fuselage (Zones 'N' and 'V') vibration levels between 30 Hz and 500 Hz are lower than between 10 and 30 Hz. They include harmonics of nR, which are at higher levels in Zone 'V' than Zone 'N'. Significant vibrations above 500 Hz are rarely found in Zones 'N' and 'V'.

2.4 Most major gearbox mesh, engine and gearbox shaft and hydraulic pump piston frequencies lie between 400 and 2500 Hz and very high levels are often measured in casings. The value of $1g^2$/Hz gives 45g rms over the band and reflects the survey results. Where gearbox sources are below 400 Hz they tend to result in lower amplitudes and the $0.10g^2$/Hz specified for the nR band is considered appropriate up to 400 Hz.

3 CATEGORY D

3.1 This test gives 'fallback' levels to be used where other data are not available. Whilst the values suggested are based on real measurements, they are obviously a poor substitute for measured data. Broad band values are derived from the survey and agree with those for Category A tests, where appropriate.

3.2 The narrow band peaks specified for each zone reflect the predominant sources encountered in different areas of the rotorcraft.

3.3 Rotor order levels are effectively specified in velocity terms (i. E., g ÷ frequency) because measured data from rotorcraft with different nR frequencies is most consistent in these units.

3.4 Engine shaft source levels are based on measured data and manufacturers' limits and the hydraulic pump source level is based on U.K. experience.

3.5 Conversion to alternative random values is based on fatigue damage assumptions which are considered adequate for this 'fall back' test.

3.6 Sine frequency sweep ranges for all peaks cover those encountered in gas-turbine powered rotorcraft.

3.7 Bandwidths for the narrow band alternatives also reflect flight ranges but are specified in such a manner as to minimise difficulties in setting up vibration test control equipment.

ANNEX B

VIBRATION TEST PROCEDURE

1 INTRODUCTION

1.1 Leaflet 501/2, para 4 deals with test specifications and describes a range of tests which are relevant to equipment for use in or on rotorcraft. This range includes a Laboratory Vibration Response Investigation (Resonance Search Test) and Qualification Endurance Tests, paras 4.5.2 and 4.6 respectively. In both cases, detailed test procedures are necessary and these are given in the BS2011 documents stated. In addition, the mounting of equipment is dealt with in I.E.C Publication 68-2-47. However, in order to aid the use of this leaflet the more salient features of the BS and I.E.C. documents are given below in edited form. Both the BS and I.E.C. documents also contain guidance material which is not given here. In the case of doubt or possible ambiguity the actual BS/IEC specifications take precedence over this Annex.

2 GENERAL DESCRIPTION

2.1 The purpose of the test is to determine mechanical weakness and/or degradation in specified performance and to use the information, in conjunction with the relevant specification, to decide whether an equipment is acceptable or not. It may be used, in some cases, to determine the structural integrity or equipments and/or to study their dynamic behaviour.

2.2 Whether an equipment has to function during vibration or merely to survive conditions of vibration will need to be stated in the relevant specification.

2.3 It is emphasized that vibration testing always demands a certain degree of engineering judgement, and all should be fully aware of this fact.

2.4 This Annex deals with the methods of controlling the test at specified points and gives the requirements for the vibration motion.

3 DEFINITION

3.1 The terms used are generally as defined in ISO Standard 2041: Vibration and Shock - Vocabulary. However, the two following terms have specific meanings herein:-

(i) Sweep cycle: a traverse of the specified frequency range once in each direction, for example 10 Hz to 150 Hz to 10 Hz

(ii) Distortion $d = \sqrt{\frac{a_{tot}^2 - a_1^2}{a_1}} \times 100\ (\%)$

where a_1 = r.m.s. value of the acceleration at the driving frequency

a_{tot} = total r.m.s. value of the applied acceleration (including the value of a_1)

3.2 In addition, the following terms are not defined in ISO Standard 2041.

3.2.1 Fixing Point. The part of the equipment in contact with the fixture or vibration table at a point where the equipment is normally fastened in service. If a part of the real mounting structure is used as the fixture, the fixing points shall be taken as those of the mounting structure and not of the equipment.

3.2.2 Measuring Point. The test shall be carried out using data gathered at certain specific points. These are of two main types, the definitions of which are given below.

Note: Measurements may be made at points within the equipment in order to assess its behaviour, but these are not considered as measuring points in the sense of this test.

(i) Check Point

(a) A point located on the fixture, on the vibration table or the equipment, as close as possible to one of its fixing points and in any case rigidly connected to it.

(b) A number of check points are used as a means of ensuring that the test requirements are satisfied. If four or fewer fixing points exist, each shall be used as a check point. If more than four fixing points exist, four representative fixing points shall be defined in the relevant specification and these shall be used as check points.

(c) In special cases, for example for large or complex equipment, the check points shall be prescribed in the relevant specification if required to be other than close to the fixing points.

(ii) Reference Point. The point chosen from the check points whose signal is used to control the test, so that the requirements of this test are satisfied.

3.3 CONTROL POINTS

3.3.1 Single point control. This is achieved by using the signal from the transducer at the reference point in order to maintain this point at the specified level (see para 4.1.1(iv)(a).

3.3.2 Multipoint control. This is achieved by using the signals from each of the transducers at the check points. The signals are either continuously averaged arithmetically or processed by using comparison techniques, depending upon the relevant specification (see para 4.1.1 (ii) (a).

4 DESCRIPTION OF TEST APPARATUS

4.1 REQUIRED CHARACTERISTICS

4.1.1 The required characteristics of the vibration generator and fixture when the generator is loaded for the conditioning process shall be as follows:

(i) Basic Motion. The basic motion shall be a sinusoidal function of time and such that the fixing points of the equipment move substantially in phase and in straight parallel lines, subject to the limitations of paras 4.1.1 (ii) and 4.1.1 (iii)

(ii) Transverse Motion. The maximum vibration amplitude at the check points in any axis perpendicular to the specified axis shall not exceed 50% of the specified amplitude up to 500 Hz or 100% for frequencies in excess of 500 Hz. The measurements need only cover the specified frequency range. In special cases, e.g., small equipments, the amplitude of the permissible transverse motion may be limited to 25%, if required by the relevant specification. In some cases, for example for large equipment or high frequencies, it may be difficult to achieve the figures quoted above. In such cases the relevant specification shall state which of the following requirements apply:

(a) Any transverse motion in excess of that stated above shall be noted and recorded in the documentation.

(b) Transverse motion need not be monitored.

(iii) Distortion. The acceleration distortion measurement shall be carried out at the reference point and shall cover the frequencies up to 5000 Hz or five times the driving frequency, whichever is the greater. The distortion, as defined in para 3.1 (ii), shall not exceed 25%. In some instances it may not be possible to achieve this, in which case a distortion value greater than 25% is acceptable if the acceleration amplitude of the control signal at the fundamental frequency is restored to the specified value, for example by use of a tracking filter. In the case of large or complicated equipment, where the specified distortion values cannot be satisfied at some parts of the frequency range and it is impracticable to use a tracking filter, the acceleration amplitude need not be restored, but the distortion shall be noted and recorded in the documentation. The relevant specification may require that the distortion defined as above together with the frequency range affected is noted, whether or not a tracking filter has been used.

(iv) Vibration amplitude tolerances. The actual vibration amplitude in the required axis at the check and reference points shall be equal to the specified value, within the following tolerances. These tolerances include instrumentation errors.

(a) Reference point. Tolerance point on the control signal at the reference point (sinusoidal): ±l5%. Tolerance on the control signal at the reference point (wide band random):

± 6dB ASD

± 1.5db r.m.s. acceleration.

The relevant specification shall state whether single point or multipoint control shall be used. If multipoint control is prescribed, the relevant specification shall state whether the averaged value of the signal at the check points or the value of the signal at a selected point e.g., that with the largest amplitude, shall be controlled to the specified level.

(b) Check points. At each check point:

± 25% up to 500 Hz

± 50% above 500 Hz

In some cases, for example for low frequencies or large equipment at some frequencies, it may be difficult to achieve the figures quoted at certain discrete frequencies within the frequency range. In such cases, it is expected that a wider tolerance or an alternative method of assessment will be stated in the relevant specification.

(v) Frequency tolerances

± 20% up to r Hz.

± 1 Hz from 5 Hz to 50 Hz

± 2% above 50 Hz.

When the critical frequencies before and after endurance are to be compared, i.e. , during a vibration response investigation, the following tolerances shall apply:

± 10% up to 5 Hz

± 0.5 Hz from 5 Hz to 100 Hz.

± 0.5% above 100 Hz

(vi) Sweep. The sweeping shall be continuous and the frequency shall change exponentially with time. The sweep rate shall be one octave per minute ± 10%.

5 MOUNTING

5.1 Equipment shall be mounted by its normal means of attachment unless otherwise stated in the relevant specification. The equipment shall be mechanically connected to the mounting surface of the test apparatus either directly or by means of a rigid test fixture, or as stated in the relevant specification.

5.2 In cases where the normal mounting structure for the equipment is available, the relevant specification will state if it shall be used.

5.3 Any additional stays or straps should be avoided. Any connections to the equipment such as cables, pipes, etc., should be so arranged that they impose similar restraint and mass to that when the equipment is installed in its operational position. In order to achieve this, it may be necessary to fasten the cables, pipes etc., to the fixture.

5.4 Equipment intended for use with isolators should normally be tested with its isolators. If it is not practicable to carry out the test with the appropriate isolators, the specimen may be tested without the isolators at a different severity, as stated in the relevant specification, but see Leaflet 501/2, Fig. 5.

5.5 The relevant specification may require an additional test on specimen with the external isolators removed or blocked, in order to demonstrate that minimum acceptable structural resistance has been achieved. In this case, the severity to be applied shall be given in the relevant specification.

5.6 The relevant specification shall state whether the effect of gravitational force is important. If so, the equipment shall be mounted such that the gravitational force acts in the same direction as it would in use. Where the effect of gravitational force is not important the equipment may be mounted in any attitude.

5.7 If significant to the test results, the relevant specification shall state also:

(i) The temperature limits within which the equipment should be tested (for example the temperature increases in the vicinity of a vibration generator table may be unacceptable for certain equipment).

(ii) The maximum level of magnetic interference which may be imposed on the equipment and/or the orientation of the equipment in relation to the direction of the magnetic field (for example near an electrodynamic vibration generator).

6 INITIAL MEASUREMENTS

6.1 The equipment shall be electrically and mechanically checked if required by the relevant specification.

7 CONDITIONING

7.1 The specimen shall be vibrated in three mutually perpendicular axes in turn which should be so chosen that faults are most likely to be revealed. The control signal at the reference point shall be derived from the signals at the check points and shall be used for single point or multipoint control. When called for by the relevant specification, control of the specified vibration amplitude shall be supplemented by a maximum limit of the driving force applied to the vibrating system. The method of force limitation shall be stated in the specification.

7.2 VIBRATION RESPONSE INVESTIGATION

7.2.1 When called for in the relevant specification, the frequency range shall be investigated in order to study the behaviour of the equipment under vibration. The response investigation shall be carried out over a sweep cycle with the vibration amplitude and the sweep rate adequate for ensuring excitation of the resonance frequencies but without endangering the equipment. Undue dwell time should be avoided.

7.2.2 The equipment shall be functioning during this investigation if required by the relevant specification. Where the mechanical vibration characteristics cannot be assessed because it is functioning, an additional response investigation with it not functioning shall be carried out.

7.2.3 During this stage, the equipment shall be examined in order to determine critical frequencies at which:

(i) equipment malfunctioning and/or deterioration of performance are exhibited which are dependent on vibration.

(ii) mechanical resonances and other response effects, e.g., chatter, occur.

All frequencies and applied amplitudes at which these effects occur and the behaviour of the equipment shall be noted. The relevant specification shall state what action shall be taken.

7.2.4 In certain circumstances, the relevant specification may require an additional response investigation on completion of an endurance test so that the critical frequencies before and after can then be compared. The relevant specification shall state what action is to be taken if any change of frequency occurs. It is essential that both response investigations are carried out in the same manner and at the same vibration amplitude.

8 QUALIFICATION ENDURANCE

8.1 This phase of the test is dependent on the category of test, see Leaflet 501/2, para 4.6.3 but will normally be wide band random vibration, perhaps with superimposed peaks, over the appropriate frequency range. If of a purely sinusoidal vibration nature, direct reference should be made to the B.S. document.

8.1.1 Duration of endurance. The duration of endurance shall be selected appropriate to the equipment category. If the duration leads to an endurance time of 10 h or more per axis, this time may be split into periods provided that stresses in the equipment (due to heating, etc.,) are not thereby reduced.

9 INTERMEDIATE MEASUREMENTS

9.1 When called for by the relevant specification, the equipment shall be functioning during conditioning and its performance checked during the conditioning for the specified proportion of the total time.

10 RECOVERY

10.1 It is sometimes necessary, when prescribed in the relevant specification to provide a period of time after conditioning in which to allow the equipment to attain the same conditions as existed for the initial measurements, e.g., as regards temperature.

11 FINAL MEASUREMENTS

11.1 The equipment shall be electrically and mechanically checked as required by the relevant specification.

ANNEX C

DATA ACQUISITION AND ANALYSIS FOR TEST DERIVATION

1 INTRODUCTION

1.1 This Annex is concerned with vibration measurements of rotorcraft equipment including both internally mounted and externally carried items. The principles described are also relevant with respect to measurements made of the dynamic characteristics of the rotorcraft airframe.

1.2 The criterion against which the success of any trial is judged is whether or not the trial objectives have been achieved. An essential pre-requisite for a trial is, therefore, a set of clearly defined objectives. In this vital sense, the aims of a trial designed to acquire vibration data is no different from any other. A clear statement of the objectives of the trial is required to govern subsequent decisions regarding data acquisition and processing techniques, and so provide the basis for a valid analysis and assessment.

1.3 The following areas are addressed in the context of the derivation of vibration test specification to aid the clearance of rotorcraft equipment.

(i) Data acquisition. A discussion on the design of a flight trial, including the selection of suitable transducers, recording methods and signal conditioning.

(ii) Data processing. A number of processing techniques which are appropriate to the rotorcraft vibration environment are described.

(iii) Analysis and test derivation. Analysis schemes relevant to different categories of equipment are presented, as well as possible methods of test derivation.

1.4 The details of approach and procedures of each of the above subjects is heavily influenced by the nature of the environment being considered. In the case of rotorcraft, the induced environment is of a special nature. There are, at the same time, potential problem areas and opportunities available for innovative design in order to exploit the effects of the environment, as perceived by the equipment.

1.5 Decisions regarding the above aspects will also depend upon the scale of a particular exercise. For example, work directed to derive a vibration test specification to aid the clearance of a store will involve a more rigorous and sophisticated approach than that associated with say, a radio set destined to be fitted in a wide range of rotorcraft types.

2 DATA ACQUISITION

2.1 The acquisition of vibration data is essential where large items of equipment, e.g., externally carried weapons, radars, etc. , are concerned. To aid the flight clearance of such items, measured flight responses may be required at various stages of their development programmes. Often this is necessary to validate early estimates which may have been used for vibration tests designed to demonstrate the general robustness of the equipment prior to any flight trials.

2.2 For smaller items of equipment e.g., internally mounted avionics, flight data, although generally desirable, may not always be necessary. Such equipment, intended for use in a wide range of rotorcraft, may be inherently robust and will, therefore, sustain a degree of overtest that renders a more refined approach redundant.

2.3 TRANSDUCERS

2.3.1 The first element in the data acquisition chain is the transducer, i.e., the device that will convert the effects of interest to a varying proportionate electrical voltage of current that can be recorded for subsequent processing. A number of different types are available which are suitable for a variety of applications. Of these, by far the most popular is the piezo-electric accelerometer. This popularity stems from a number of practical features including its relatively low cost, small physical size, general environmental robustness, frequency response range and sensitivity.

2.3.2 Unfortunately, accelerometers can produce erroneous data due to their sensitivity to parameters other than acceleration, e.g., temperature and pressure. Movement of connecting cables, particularly between accelerometers and their associated charge amplifiers, can also give rise to erroneous results. It is essential to ensure that precautions against these effects have been taken.

2.3.3 It is most desirable to mount accelerometers and indeed other transducers, on to 'prime' structure where possible; this strategy prevents responses being swamped by, for example, panel modes or local resonances, which may not be of primary interest.

2.3.4 Care should be taken when fixing accelerometers to a structure since the type of mounting can affect the upper frequency limit of the results. For flight and other major trials the most suitable methods of mounting are those in which the accelerometer is mechanically attached either by bonding or by use of a screw fastening. This can involve the use of small mounting blocks made of materials such as Tufnol or aluminium. Such blocks, which can be shaped to fit contoured surfaces or to conform to a prescribed measurement axis system, can be bonded directly to the structure and the accelerometers fixed to them by the usual screw fastening. This type of mounting can be expected to provide good data up to frequencies of 2 or 3kHz.

2.3.5 For some trials, when the upper frequency of interest is less than, say, 100 Hz, e.g. , a ground modal survey, magnets, plasticine or bees wax may be used to attach accelerometers. In all instances, the aim should be to ensure that the accelerometer mounting is firm, positive and in the desired axis.

2.3.6 Another type of transducer in wide general use is the strain gauge. Such devices are often employed to acquire data to be used in fatigue life estimations. The proper fixing of strain gauges, often required in physically awkward locations, requires great care if reliable data is to be obtained. Furthermore, strain gauge installations are often vulnerable to damage during the course of a trial and special attention should be given to their physical protection. As is the case for accelerometers, strain gauges can give erroneous results due to temperature effects. This problem can be exacerbated by the levels of gain of approximately 1000 associated with foil gauges, for example. This can be combated by the use of dummy gauges.

2.3.7 The number of transducers needed for a particular exercise will depend upon the type of equipment under investigation and the purpose of the exercise to be carried out. For a small item, a tri-axial or even a single transducer might well be sufficient. Larger, more dynamically complicated structures demand a more comprehensive instrumentation installation. A heavy external store, for example, would require six transducers in order to describe the six rigid body modes of vibration. Additional transducers might also be needed to gain information on critical areas.

2.4 RECORDING OF DATA

2.4.1 Usually it is impracticable to process data in real time. Although there are exceptions to this, e.g., some test house trials, transducer signals are usually recorded for processing at a later date. Under these circumstances, it is critically important that not only is a suitable recording medium used, but also that the supporting documentation, e.g., tape log sheets, is complete and unambiguous.

2.4.2 Failure to accurately document exactly what was recorded during the trial can, at least, cause unnecessary delay and increases costs at the processing stage and, at worst, lead to erroneous conclusions. There are obvious benefits from encoding information about gain settings etc., on the actual data tape, if possible.

2.4.3 Various methods of recording the electrical output of transducers are available. These fall into two main categories, viz. analogue and digital recording on to magnetic tape. Digital recording, in the context of gathering trials data, is somewhat less popular currently, for a variety of reasons including both cost and flexibility.

2.4.4 The performance of magnetic recorders is adversely affected by a number of environmental conditions that may be encountered in the course of trials work, e.g., dirt, extremes of temperature, atmospheric pressure and humidity. The tape transport mechanisms of recorders can also be affected by vibration, which is an obvious disadvantage when measurement of that parameter is being attempted. This problem can be counteracted by the use of a vibration isolator system for the recorder.

2.4.5 Analogue magnetic tape records may be made in either Direct or Frequency Modulation (FM) modes. As its name suggests, direct recording involves recording analogue voltages directly on to magnetic tape, with the addition of a very high frequency bias current to linearise the properties of the medium. In order to correct variations from the ideal flat frequency response, a process of equalisation is necessary during replay.

2.4.6 As a result of the equalisation process, phase information between channels of data becomes corrupted towards the upper and lower frequency bounds of the recording. This effectively rules out the use of Direct recording when information is required, e.g., to obtain mode shapes. Another adverse feature of this means of recording is its inability to record low frequency signals. In view of the above problems, the Direct mode of recording is not normally to be recommended unless high frequency data, greater than, say, 10kHz is required.

2.4.7 A more satisfactory and hence more common method of recording analogue data is that of Frequency Modulation (FM). This process involves the modulation of a carrier frequency in proportion to the voltage varying incoming signal. By this means, frequencies present in the original signal can be faithfully recorded down to steady-state conditions. The penalty for this is a lower frequency limit than for Direct recording e.g., 2.5kHz for a tape speed of 7.5 ips. It is possible, however, to recover data at higher frequencies than those normally associated with a given tape speed at the expense of signal to noise ratio.

2.4.8 Under the FM system of recording, provided only that the frequency of the recorded signal can be detected, the original signal can be recovered by demodulation during replay; results are then unaffected by those types of distortions introduced by the recording medium. A potential disadvantage is that any variation in the, recording speed, e.g., resulting from vibration, appears incorrectly as amplitude variation during replay. This may, however, be compensated for, during replay, by use of a reference signal recorded simultaneously with the data proper.

2.4.9 A recorder running at 7.5 ips in the FM mode will provide an upper frequency limit of 2.5kHz, which is more than adequate for most work; exceptions being when equipment or interest is mounted close to engines or gearboxes.

2.4.10 Most tape recorder manufacturers optimise their machine's mechanical performance for working at a particular tape speed. It is inadvisable to operate a recorder at speeds lower than 3.75 ips because of possible adverse effects of the rotorcraft vibration environment upon movement of the tape through the transport mechanism.

2.5 DIGITAL RECORDING

2.5.1 When data gathered in flight are to be analysed digitally, it is natural to consider digital recording techniques. The most popular recording medium for digital data is magnetic tape. Other media include random access memory (RAM), magnetic discs and bubble memory.

2.5.2 By the process of digital recording, the recording medium can be made completely 'transparent', i.e., there is no difference between the recorded and recovered data. This is achieved by sampling and digitising the analogue signals at source, via sample-hold and analogue to digital converter (ADC) hardware, and representing them as a stream of binary words. A 12-bit word is commonly used which provides an amplitude resolution of 1 in 4096 (0.02%). Whilst this represents excellent resolution, it is seldom achieved in practice because typical utilisation of an ADC may be around only 40%.

2.5.3 Provided that the record quality is sufficient to enable each recorded "bit" to be recovered, then the recording and replay processes are completely transparent, i.e., there is no difference between the recorded and recovered data. This feature is unique to digital recording and makes the technique very attractive. It is inevitable that the technique has disadvantages and these include:

(i) Additional electronic equipment must be carried in the rotorcraft. This can be avoided if data are telemetered to a ground station.

(ii) The packing density of information is much lower, e.g., less than one tenth, than can be achieved with FM recorders.

(iii) It is usually necessary to have a prior knowledge of the signal levels if the best use is to be made of limited resolution digitisers.

(iv) Complex and expensive equipment is normally required in order to recover flight records.

2.5.4 Digital data may be recorded either directly (computer compatible) or indirectly, via an interface, e.g., RS233, IEEE. At the present time standard computer compatible recording formats are rarely used, for a number of reasons, e.g.:

(i) Data must be recorded on tape in discrete blocks. This can only be achieved by using a pair of interleaving storage buffers.

(ii) Computer tape recorders normally include hardware which prevents a block being read if it contains an error.

(iii) Computer standard packing densities are normally much lower than can be achieved technically.

2.5.5 A great many recording formats are available commercially under the generic title of Pulse Code Modulation (PCM).Examples include Non-Return-to-Zero (NRZ), Manchester Code, and MODEM used by telecoms. The available formats differ in the packing densities achieved, error rates, track utilisation and error detection facilities.

2.6 SIGNAL CONDITIONING

2.6.1 The application of gain to transducer signals is usually desirable in order to optimise the dynamic range of the recorder and so ensure the best possible signal to noise ratio in the recording. Monitoring of signal levels during flight, where practicable, allows the appropriate gains to be applied. Similarly, filtering out unwanted high frequencies e.g., those originating from a rotorcraft engine or gearbox, prior to recording, may be desirable.

2.6.2 Further processing of signals prior to recording, e.g., to obtain velocity or displacement data from accelerometers is not recommended. It is better to record the measured parameter and maintain flexibility by carrying out such work subsequently using the basic recorded data. Where the interest is in crew or passenger comfort, or structural fatigue, velocity units usually provide the most correct ranking of components in the rotorcraft vibration spectrum and make optimum use of the vibration recorded dynamic range over the frequency band of interest, typically 3 to 200Hz. Velocity measurements are not normally appropriate for measurements at higher frequencies.

2.7 CALIBRATION

2.7.1 Calibration is a critical part of a data acquisition exercise. The best way to ensure proper calibration is to make the procedure simple. Some instrumentation systems, for example, automatically record calibration signals immediately prior to every recording of transducer signals.

2.7.2 Calibration information usually involves the recording of known voltages on to magnetic tape and logging the equivalent of those voltages in terms of physical units. By this means, scale factors can be recovered during replay in terms of physical units per volt e.g. g/V.

2.7.3 The preferred form of calibration is that of multiple steady state levels. This permits the zero offset, sensitivity and sign of a signal channel to be estimated simply. This kind of calibration can, of course, be used only with recording media capable of reproducing steady-state levels.

2.7.4 An alternative form of calibration is that using a.c. voltages. A potential difficulty here lies in how such signals are measured e.g. , by a true rms meter or by a spectrum analyser. The former will take into account a full bandwidth, including the presence of harmonics, whereas the latter may only measure at the specified frequency. Differences can be of the order of 10%, according to the quality of the signal generator. Similar differences may also be introduced due to the characteristic of the analyser used e.g., windowing.

2.8 RECORD LENGTHS

2.8.1 In considering record lengths, an important factor concerns the statistical accuracy of certain types of results obtained during processing. For band limited white noise, the accuracy associated with estimated mean square values is expressed as the normalised variance error 's'. This quantity depends upon the resolution bandwidth (Be), described in para 3.3.1 (i) of this Annex, and the record length (T) according to the equation:

$$s = \frac{1}{\sqrt{B_e T}}$$

Typical parameters for processing are a record length of 25 seconds and a resolution bandwidth of 1 Hz This corresponds to a variance error of 20%. There are many situations, however, when it is desirable to trade-off frequency resolution against amplitude accuracy, and vice-versa, according to particular requirements.

2.8.2 There are two approaches to recording responses during flight, viz., continuously or intermittently, when response measurements relating to prescribed conditions only are made.

(i) Continuous recording of a typical sortie may well require up to 2.5 hours of tape, including calibration. An advantage of this technique is that any unexpected responses, which may occur between the individual manoeuvres, will be recorded. A possible disadvantage is that great care needs to be taken to ensure that specific events can be recognised during subsequent replay. This is best achieved using an event marker recorded on to one tape channel, backed-up by a recorded voice log.

(ii) When recording intermittently, each steady-state manoeuvre case should be held for approximately 30 secs; transient conditions can require slightly longer record lengths. When deciding upon record lengths, an allowance should be made for a settling period for the electronics response to switching transients. An advantage of this type of recording is that the recorder switch on/off transients enables the various flight conditions to be easily recognised during replay. Also, relatively little tape is required. A possible disadvantage lies in the difficulty in acquiring data in the immediate vicinity of the switching transients, should this become necessary.

2.9 ARCHIVING RESULTS

2.9.1 It is recommended most strongly that results obtained from trials are archived. By doing so, the capability of checking unexpected effects that may only emerge after processing, is maintained. The relative merits of archiving the original magnetic tape records or, where appropriate, digitally acquired data may be debated. In view of the low costs incurred relative to the cost of repeating a flight trial, archiving in both forms is suggested.

3 DATA PROCESSING

3.1 GENERAL

3.1.1 Whilst useful results may be obtained from analogue equipment, data processing is more often carried out using digital techniques; these are discussed below.

3.1.2 Data processing activities fall into two categories, viz., on line and off line, examples being the digital acquisition of strain time history responses and subsequent Rainflow calculations, respectively. Before either of these activities takes place, there is opportunity to condition the data by application of gain. This should be done in order to optimise the use of the dynamic range of analogue to digital converters.

3.1.3 The procedure of digitising the data previously recorded on to analogue magnetic tape has major implications on the eventual results. These implications are in conjunction with the required frequency range, resolution of amplitudes or frequencies and confidence in the measured amplitudes. The governing parameters for these considerations are:

(i) The acquisition rate (samples per second).

(ii) The size of the Fourier transform, where frequency results are concerned.

(iii) The acquisition time.

3.1.4 The required frequency range will depend upon the details of a particular investigation. Generally, an upper limit of 500 Hz is suitable for rotorcraft work, unless responses associated with gearboxes and engines are of interest. In such cases, an upper limit in excess of 2000 Hz would be appropriate. Guidance in this area can be obtained from the Design Authority for the appropriate rotorcraft.

3.1.5 Having determined the maximum frequency of interest, an acquisition rate may be chosen. As a general rule it is desirable to sample data at a rate of approximately four times the maximum required frequency in order to avoid problems regarding aliasing. This factor includes an allowance for the finite roll-off rates of practical filters. If more precision is required in the selection of an acquisition rate, then the actual roll-off rate of the anti-aliasing filters will need to be considered. It is essential to apply low pass filtering at the maximum frequency of interest.

3.1.6 In proprietary data processing equipment, e.g., spectrum analysers, the above parameters are sometimes pre-selected by the manufacturer i.e., once a frequency range is specified the choice of anti-aliasing filter, acquisition rate and, therefore, resolution bandwidth, is made automatically.

3.1.7 The character of most vibration responses measured on rotorcraft during flight, is that of strong forcing at a set of particular frequencies against a background of broad band vibration. There may be additional responses particular to an item of interest. These aspects are addressed in Leaflet 501/1. It is necessary to recognise these special characteristics of the recorded data to properly interpret the results.

3.1.8 The kind of data processing applied to recorded transducer signals will depend on the nature of a particular trial or, more specifically, the requirements of the eventual user of the results. The following paras describe a number of processing techniques that are available. Assumptions implicit in these techniques and their effects upon results obtained, are discussed. Results in both the time and frequency domains are included. Which of these options would be appropriate for a particular task is also considered.

3.2 TIME HISTORIES

3.2.1 Simple analogue time histories from oscillograph recorders e.g., ultraviolet (u.v.), may be used to enable a 'quick-look' to be taken at the measured responses, in order to generally assess their character and integrity. By this means, any obvious instrumentation faults or other anomalies may be recognised and advice sought before proceeding with more sophisticated analysis. Frequency response is unlikely to be a limiting factor in this context. Due to their nature, u.v. records have a finite life and so cannot be regarded as a satisfactory final form of results.

3.2.2 Digitally derived time histories are subject to sample rate and anti-aliasing considerations discussed above.

3.3 SPECTRAL FUNCTIONS FROM FFT ANALYSERS

3.3.1 The following functions are in wide general use as a result of the proliferation of digital spectrum analysers that utilise the FFT algorithm.

(i) Acceleration Spectral Density (ASD). This function provides the mean square amplitude measured in a number of frequency bands. The information gained in the form of acceleration spectral density is the result of two averaging processes:

(a) Averaging in the time-domain over the time record length.

(b) Averaging over the effective analyser bandwidth.

Acceleration spectral density may be defined as:

$$ASD = \frac{\overline{x}^2}{B_e}$$

where B_e = the resolution bandwidth

$\overline{x}^2$ = mean square amplitude

For stationary, broad-band random data, varying either the bandwidth or the record length, within reasonable limits, will not have a great effect on the results, except on frequency resolution and the statistical accuracy (variance error).

In the case of rotorcraft derived measurements, however, the data is not necessarily stationary and the power tends to be concentrated around a set of discrete frequencies. The implication of these observations is that the form of the ASD is dependent on the above two parameters to a far greater extent than is the case for, say, vibration responses measured during fixed-wing flight trials. It must be emphasised that a measurement in unqualified ASD units at a frequency of one of the rotorcraft's periodic peaks is of no value.

(ii) Cumulative Power. It is useful, when making assessments, to be able to obtain the rms value over other than the full frequency range. This requirement is met by the cumulative power function. It is the running integral of the ASD function and is defined as:

$$B_e \sum_{f_o}^{f\,max} \bar{x}^2(f)$$

It is an aid to the assessment of the spectral peaks in terms of their contribution to the overall rms values and provides a quick way of calculating the rms value over a given frequency range.

(iii) Crest Factor (Envelope Ratio). This function evaluates the ratio of the maximum to the mean root mean square values encountered in each frequency bandwidth, over the entire record length. Envelope ratio is defined as:

$$\frac{\bar{x}^2_{max}(f)}{\bar{x}^2_{mean}(f)}$$

The envelope ratio function is an aid in the recognition of sinusoidal components which may be buried and not otherwise detected in random data. For a sinusoid, the peak and mean values of the Fourier coefficient are equal and hence the ratio becomes unity. For random data, the ratio generally lies between 2 and 4, although the actual values obtained are a function of time; the longer one is prepared to wait, the greater becomes the chance of encountering a high amplitude low probability value.

Care must be exercised with recordings which cover other than one steady state flight condition, as the measured crest factor will be influenced when amplitudes of periodic functions change with flight conditions.

(iv) Peak Hold (Equivalent Peak). As discussed above, an ASD plot may be used to identify frequency content and mean square amplitude information from a time history record. This may be combined with the relevant envelope ratio spectrum to obtain a plot of equivalent 'g' peak. This spectrum describes the greatest accelerations derived from any single Fourier transform; hence its alternative title of peak hold. The term is defined as:

$$\frac{\bar{x}^2_{max}(f)}{B_e}$$

It is worth noting that any so-called peak measurements that are derived from spectrum analysers that utilise the FFT or similar algorithms, are based on mean square amplitudes computed within the analysers resolution bandwidth; simply factoring the mean values will not provide a true peak.

3.4 OTHER ANALYSIS PROCEDURES

3.4.1 A number of other analysis procedures are available; these include the following:

(i) Amplitude Probability Density, p (x). This function measures the probability of encountering amplitudes in a processed time-history. It is useful for checking the normality of random responses.

For random data the function is defined as:

$$p\,(x) = \frac{1}{\sigma\sqrt{2\pi}}\;\frac{\exp-(x-\bar{x})^2}{2\,\sigma^2}$$

where $\bar{x}$ = mean value (= zero)
σ = standard deviation (= root mean square)

For a sinusoid,

$$p(x) = \pi\left(\sqrt{2\sigma^2 - x^2}\right)^{-1} \quad \text{for} |x| < x \quad \text{for} |x| \geq x$$

where $\sigma = x \div \sqrt{2}$ = standard deviation (= rms)
and x = peak amplitude

Amplitude probability densities measured on rotorcraft tend to either form depending on the manoeuvre being flown. However, when a signal is made up of more than one sinusoid e.g., nR, 2nR, the function can be difficult to distinguish from random data.

Amplitude probability density is generally presented in a normalised form so that the total area under the curve is unity i.e., there is 100% probability that the amplitude resides between +/- infinity.

Amplitude probability density may be plotted against amplitude squared, or $x.|x|$ which preserves the sign. For normal random the data conveniently yields a straight line graph when plotted on logarithmic axes; this presentation permits deviations from the normal form to be recognised easily.

(ii) Amplitude Probability Distribution P(x) This function describes the percentage of time that a signal spends within specified amplitude bands, D(x). The process is akin to filtering but in the time domain. The value of this function lies in its direct provision of load case amplitudes. The function is defined as the running integral of the amplitude probability density.

(iii) Rainflow Analysis. Rainflow analysis is designed to provide information of the fatigue damage suffered by a structure as a result of vibration or applied loads. The effects of the vibration are recorded using strain gauge transducers. The resulting strain-time histories are subjected to the Rainflow analysis. The major application for this type of analysis is to facilitate comparisons between measured responses and S/N curves in order to assess the fatigue damage potential of a given environment.

4 ANALYSIS FOR TEST DERIVATION

4.1 GENERAL

4.1.1 This para is concerned with the selection of appropriate data processing options, the interpretation of the results obtained and their presentation in a suitable form for assessment. There should be a close inter-relationship between these activities, in order that a valid assessment can be made. It is often the case that an iterative process between these disciplines best yields the desired results.

4.1.2 The type of analysis carried out will depend upon the nature of a particular requirement. In the context of this Leaflet these requirements may be regarded as providing the motivation for collecting flight data for two main purposes:

(i) For use in the derivation or validation of a ground vibration test.

(ii) To permit fatigue life estimations to be made. This usually involves either the airframe or store carriage equipment.

4.1.3 A feature common to each of these general requirements involves the 'scaling up' of a few flight trials, often only one, in order to provide information intended to be relevant to the life of an equipment that may need to be measured in terms of thousands of hours. This problem is exacerbated by the fact that it is impossible to include every condition and manoeuvre case in a small number of flight trials. There are two approaches to counter this. Either a representative mission is flown and vibration responses recorded continuously during the flight, or a programme of common manoeuvres is flown, with recordings made only when the prescribed conditions are met. If it is known that the equipment is to be used only in a limited role, e.g., sonar search routines, then the first method is preferable. If, however, the details of operational service are unpredictable, the second method is to be preferred.

4.1.4 A further complicating factor concerns the character of some rotorcraft response data. During flight, rotorcraft equipment exhibits forced responses, predominantly at the main rotor blade passing frequency. During straight and level flight, the responses tend to be close to sinusoidal form. This may not be the case when the rotorcraft carries out manoeuvres, when peak levels can increase to three or four times those of steady conditions. The lengths necessary to reduce the uncertainties associated with the above problem, depend upon the type of equipment being considered. For some items, a gross over-estimation of the vibration environment may not be an embarrassment when that item is tested. In such cases, a simple description of the environment which envelopes every conceivable variation in a 'worst-on-worst' manner, could form the basis of a test. For some types of equipment this may not cause difficulties. For heavy equipment, however, greater insight into store behaviour and more representative tests are necessary for the following reasons:

(i) Because such items react dynamically with the rotorcraft airframe, a detailed knowledge of their behaviour is necessary to enable fatigue estimations to be made regarding both the rotorcraft airframe and the equipment.

(ii) To lessen the impact on the operational capabilities of the rotorcraft, weight is at a premium. Over-design is, therefore, to be avoided.

(iii) Modern heavy stores, particularly those of an external nature, often depend for their proper functioning upon their sophisticated and delicate electronic packages. Each of these will have been procured by the prime contractor against a vibration test specification. Unnecessary robustness in this area can lead to significantly increased weight and cost.

4.1.5 For the purpose of this Leaflet equipment has been classified as Light, General Purpose; Light, Specific and Heavy, (see Chapter 501 para 4.3). In the case of Light, General and, hopefully, that for Light, Specific equipment, it is expected that the test specification based on the general information presented in Chapter 501 para 4 will be adequate for flight clearance of these categories of equipment. Such a test can be expected to be relatively severe because of the necessary generalizations made in its formulation. If such a test either causes, or is expected to cause, difficulty then acquisition of flight data could well lead to a less severe test; the trade-off being that the applications for the equipment become more specific.

4.2 DERIVATION OF VIBRATION TEST SPECTRA

4.2.1 This section covers derivation of two appropriate types of test which result from the analysis of flight data.

(i) Sine on Wide Band Random Test. Plots of ASD will provide the necessary frequency information. This will include confirmation of those frequencies obtained from the rotorcraft design authority which are associated with the various rotating elements, and also describe the frequency range of the general random vibration background excitation.

For a sine on random test, the amplitudes of the dominant sine-like components and the level of the random vibration need to be determined. The random vibration is generally idealised to the form of a number of straight line segments that envelope the measured results. The margin between the envelope and the actual measurements is a trade-off between confidence that the equipment has been adequately tested and avoidance of too severe an over-test that could lead to unrepresentative failures during the test.

Estimates of the amplitudes of the sine like components can be obtained from acceleration density information. For example, multiplying an ASD by the resolution bandwidth will provide a measure of mean square amplitude as a function of frequency. From such a plot, the mean square, and so the rms value, can be obtained within the resolution bandwidth at the various centre frequencies. By this means, an rms value can be assigned to the frequency components of interest.

There are two aspects of the above approach that should be noted. Firstly, it is implicitly assumed that the data are sinusoids, i.e., the peak to rms ratio is 1.414, and, secondly ASDs are subject to certain averaging effects that are discussed in para 3.3.1 (i) of this Annex. To avoid the possibility of undertesting that might result from these averaging effects, the above procedure can be based on peak hold spectra, see para 3.3.1 (iv), although now the possibilities of overtesting have to be considered. Re-examination of time histories may be helpful at this stage.

If either of the above qualifying remarks give cause for concern, then the more representative test (narrow band on broad band random), described below, should be considered as an alternative.

(ii) Narrow Band on Broad Band Random Test. The narrow band on broad band random test is probably the most appropriate test for the majority of equipment. The narrow band component is to accommodate excitation at the rotorcraft's blade passing frequency. The broad band background vibration caters for the remaining sources of excitation. Additional narrow bands can be centred upon harmonics of the main rotor blade passing frequency, if considered necessary. This decision should be based upon flight data and, where available, knowledge of the equipment resonances.

The production of a vibration test spectrum involves two phases. Firstly, a description of the broad band background vibration and, secondly, a description is required of the narrow band process occurring around the blade passing frequency.

To gain maximum confidence in the test specification, it would be desirable to envelope every aspect of the measured responses. This would lead to a single test spectrum which would be applicable for each of the store's axes. The penalty associated with this approach is that of possible over test. To avoid this problem, responses e.g., acceleration spectral density, can be compared by overlaying spectra. By this means it is possible to both assess the scatter in the results and to isolate which conditions or measurements are responsible for the scatter.

Two approaches for determining the level of the narrow band vibration are suggested below, viz., using spectra or amplitude probability data.

From spectra originating at appropriate locations within the equipment (see para 2.3 of this Annex), values of mean square vibration occurring around the blade passing frequency can be obtained. It can be convenient to use the cumulative power format for this purpose, (see para 3.3.1 (ii) of this Annex). From the measured mean square amplitude within the test bandwidth, the level of the narrow band vibration in terms of g^2/Hz can be calculated.

The use of amplitude probability data is suggested as an alternative to spectra as a means of establishing test levels of the narrow band component. From a plot of amplitude probability distribution for a representative mission, (see para 3.4.1 (ii) of this Annex), test amplitudes can be specified. A probability of occurrence of 1 in 500 is suggested.

The corresponding amplitude peak acceleration can then be interpreted in terms of an rms value as it is known that, for a Gaussian process, as will be generated during test, the 1 in 500 case amplitude is related to the rms by a constant, i.e. ,

$$A_{1\text{ in }500} = 2.88 \text{ x rms}$$

To obtain the most representative test, the amplitude probability distribution information should relate to band limited data, centred about the blade passing frequency and if appropriate, associated harmonics. In practice, it may be considered acceptable to use the full recommended analysis frequency range i.e., 200Hz, for this purpose. This is because, for a typical measurement, it may be expected that about 80% of the contribution to the overall mean square value originated from the blade passing frequency.

LEAFLET 501/3

VIBRATION AND INTERNAL NOISE

SPECIFICATION FOR THE EVALUATION OF HUMAN EXPOSURE TO ROTORCRAFT VIBRATION

1 INTRODUCTION

1.1 SCOPE

1.1.1 The vibration environment in the rotorcraft may impair the working efficiency of the crew member both indirectly, due to discomfort and general dissatisfaction with the machine, and by direct interference with certain visual and manipulative tasks. Prolonged exposure to high amplitudes of rotorcraft vibration may also be detrimental to the long term health of the crew member.

1.1.2 This Leaflet presents a method of evaluating the severity of crew station vibration.

1.2 LAYOUT OF LEAFLET

(i) PARA 2: describes how to evaluate the relative discomfort of the vibration and presents a dose effect relationship which may be used to give an indication of the risk to some aspects of health.

(ii) PARA 3: presents guidelines for the design of hand controls for vibration environments.

(iii) PARA 4: presents similar guidelines for the design of instruments and displays.

(iv) ANNEX A: Measurement system details and performance.

(v) ANNEX B: Factors affecting manual control performance in the rotorcraft vibration environment.

(vi) ANNEX C: Display factors affecting visual performance in the rotorcraft vibration environment.

(vii) ANNEX D: Bibliography.

1.3 GENERAL CONSIDERATIONS

1.3.1 The guidelines given in this Leaflet are based wherever possible on the results of experimental studies. The effects of vibration on comfort, health and performance are mediated by a complex interaction of a number of factors. Some simplification has been necessary in order to make the procedures described in the Leaflet simple to apply. Where other variables are likely to be important they have been identified and, in the case of the sections on hand control and visual performance, they are discussed in separate appendices. In some areas current knowledge is still incomplete. In these cases the probable relative importance of the relevant variables has been indicated, based on current scientific opinion.

1.4 MEASUREMENT OF ROTORCRAFT VIBRATION

1.4.1 Measurement Sites. The vibration shall be measured at the interface between the aircrew member and the seat and seat back in each of the orthogonal translational axes defined in Annex A to this Leaflet. Measurements shall be made on both the pilot's and co-pilot's seats and on other crew seats within the rotorcraft. Vibration measurements shall also be made on the instrument consoles and, wherever practicable, on the floor or foot pedals close to the pilot's and co-pilot's feet. The axes and methods of measurement of vibration on instruments and on the cockpit floor are also described in Annex A.

1.4.2 Recording and Analysis of Acceleration Time Histories. Where possible the acceleration time histories should be recorded in a storable format (e.g., on magnetic tape in either analogue or digital form). Performance requirements for the transducers, recording and analysis equipment are given in Annex A. When the data are analysed the acceleration time histories should each be band limited at 0.4 Hz and 100 Hz. Different frequency weightings are required for each application, as defined in later sections. The weighting filters are described in Annex A and their response is illustrated in Fig.1. The derivation of these weightings and their application are presented in Refs 4, 9, 11. The overall root mean square acceleration amplitude for each axis and weighting function shall be computed for each flight mode in a typical mission. In each case the root mean square acceleration amplitude shall be computed over more than twenty seconds, or a period appropriate to the duration of an event. (e.g., transition to or from hover).

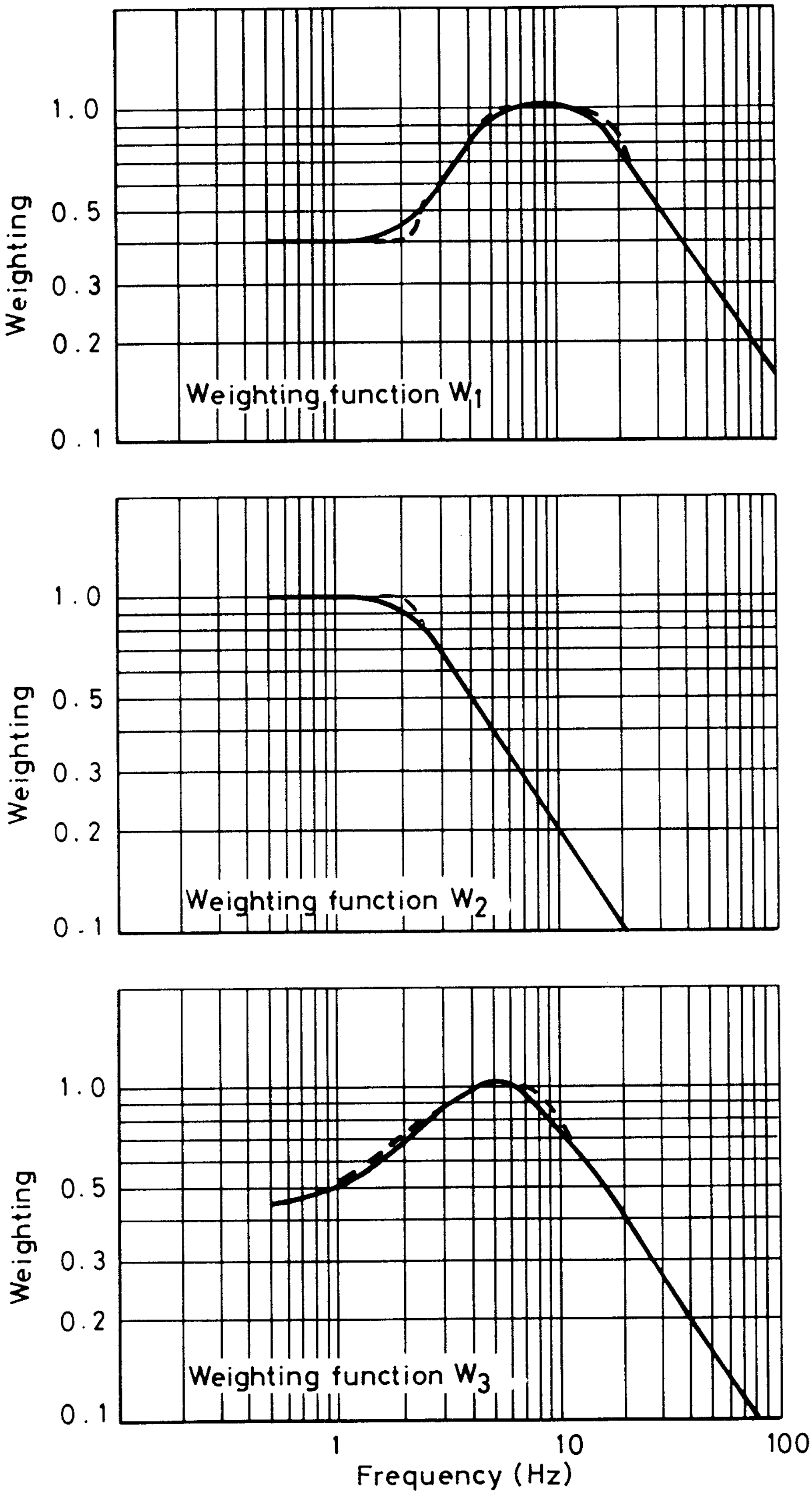


FIG.1 THE FREQUENCY WEIGHTING FUNCTIONS USED IN THE EVALUATION OF THE EFFECTS OF ROTORCRAFT VIBRATION ON AIRCREW.

(——— Frequency weighting defined in Table 1 Annex A - - - Straight line approximation: See Table 2 Annex A

2 GENERAL EFFECTS OF WHOLE BODY VIBRATION

2.1 INTRODUCTION

2.1.1 This para is concerned with the evaluation of the effects of vibration other than those which directly interfere with human activities. The degree to which a particular vibration condition is considered to cause unacceptable discomfort is dependent upon many factors, including the circumstances of the mission and the nature of the task being performed. The extent to which discomfort will interfere with the performance of a particular task is also dependent on similar factors. When the task being performed is not demanding, a moderate amount of discomfort due to vibration may enhance performance by maintaining the level of arousal of the aircrew. However, if the aircrew are well motivated and aroused discomfort may prove to be an unacceptable additional stress. This para does not, therefore, present a vibration limit for comfort, but requires that an equivalent comfort value be reported for comparative purposes (see Annex D Refs 4, 5, 9, 14).

2.1.2 There are currently no firm data to show how the risk of injury is related to the duration of exposure to vibration. A time dependency is defined in this para as a relationship between vibration amplitude and duration which appears to be consistent with current information (see Annex D Refs 8, 10, 11). No absolute limits are set for vibration exposure, as the acceptable risk, particularly in the military context, is somewhat dependent upon the nature of the task being performed. However it is required to report vibration levels which exceed the defined time dependency, as these may indicate an increased risk of injury due to long-term vibration exposures.

2.2 EVALUATION OF DISCOMFORT DUE TO VIBRATION

2.2.1 The weighted root-mean-square acceleration value in each of the three translational axes on the seat shall be computed with the following weightings:

x-axis on supporting seat surface	-	W_2
y-axis on supporting seat surface	-	W_2
z-axis on supporting seat surface	-	W_1

2.2.2 The weighted root mean square acceleration amplitudes in each axis shall be reported separately but may also be expressed in the combined form:

$$a_s = (a_{sx}^2 + a_{sy}^2 + a_{sz}^2)^2 \text{ ms}^{-2} \text{ rms}$$

where a_{sx} a_{sy} and a_{sz} are the weighted root mean square acceleration amplitudes measured in the x, y and z axis respectively.

2.2.3 The root mean square acceleration value shall also be obtained for z-axis floor vibration after weighting by W_1. Procedures also exist for evaluating accelerations in other axes, but they are not considered appropriate here (see Annex D Ref 9). The likely degree of discomfort produced by any weighted value may be estimated by reference to Table 1.

TABLE 1

FREQUENCY-WEIGHTED VIBRATION MAGNITUDE	DEGREES OF COMFORT
$<$0.315 ms^{-2} rms	not uncomfortable
0.315 - 0.63 ms^{-2} rm	a little uncomfortable
0.5 - 1.0 ms^{-2} rms	fairly uncomfortable
0.8 - 1. 6 ms^{-2} rms	uncomfortable
1.25 - 2.5 ms^{-2} rms	very uncomfortable
$>$2. 0 ms^{-2} rms	extremely uncomfortable

LIKELY DEGREES OF DISCOMFORT PRODUCED BY FREQUENCY-WEIGHTED VIBRATION MAGNITUDES (a_S).

2.2.4 The weighted root mean square acceleration amplitudes on the seat and floor shall be reported and documented for each flight mode in a typical mission.

2.3 EVALUATION OF THE RISK OF INJURY DUE TO VIBRATION

2.3.1 The evaluation of the risk of injury due to long term exposure to vibration is based on the z-axis acceleration on the supporting seat, weighted by W_1, as in the evaluation of discomfort in para 2.2. The design aim should be that the weighted root mean square acceleration amplitude should not exceed the boundary defined in Fig 2 for the expected daily exposure. The root mean square acceleration boundary is a function of time:

$$a = \frac{10.68}{t^{1/4}}$$

where t is the flight duration in seconds. The weighted z-axis root mean square acceleration values obtained in typical flight modes, and the typical flight duration which would exceed the above boundary, shall be reported to the Rotorcraft Project Director.

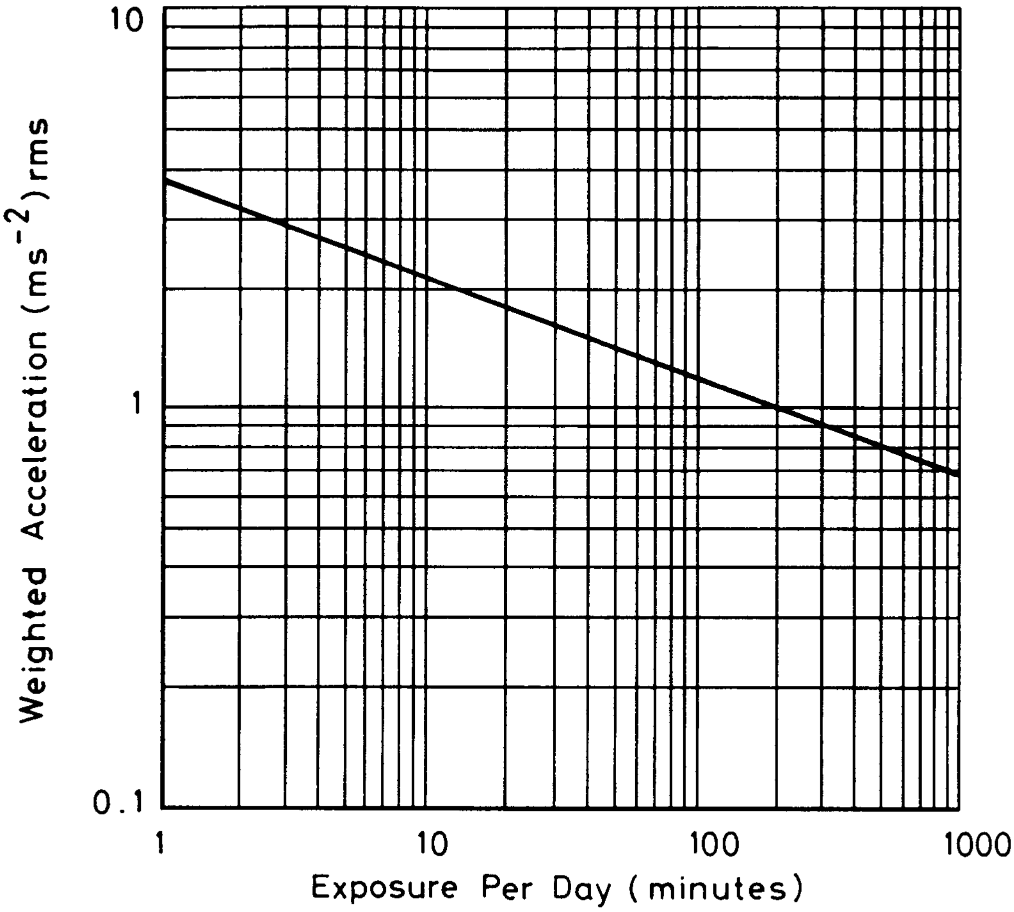


FIG.2 THE RELATIONSHIP BETWEEN WEIGHTED ROOT MEAN SQUARE ACCELERATION AMPLITUDE AND DURATION OF EXPOSURE DEFINING A BOUNDARY FOR INCREASED RISK OF VIBRATION INJURY.

3 EFFECTS OF VIBRATION ON MANIPULATION OF HAND CONTROLS

3.1 INTRODUCTION

3.1.1 Vibration produces involuntary motion and force in the limbs which can restrict the accuracy of manipulative and control actions. Tasks requiring the highest control precision are usually performed by hand controls. Foot control is not considered in this para for this reason, and because there is a paucity of experimental data concerning control by the feet.

3.1.2 The procedures outlined in this para define the maximum accuracy of hand movement or force which can normally be expected with a given vibration input. The extent to which the performance of a control task is affected by the movement or force at the hand is determined by a number of factors which are discussed in Annex B.

3.2 EVALUATION OF THE EFFECTS OF VIBRATION ON HAND CONTROL

3.2.1 The weighted root mean square acceleration amplitude in the three translational axes on the seat, and in the x-axis on the seat back, shall be computed with the following weightings:

x-axis on supporting seat surface	-	W_2
y-axis on supporting seat surface	-	W_2
z-axis on supporting seat surface	-	W_3
x-axis on seat back	-	W_3

3.2.2 Evaluation procedures used elsewhere do not take separate account of seat-back vibration (see Annex D Refs. 12, 13). It is evaluated here because x-axis vibration may occur at relatively high acceleration amplitudes in rotorcraft and there is experimental evidence to show that, when transmitted via the seat back, this can significantly affect control performance (see Annex D Ref.1).

3.2.3 The minimum position or force error between the hand and control shall be determined separately from Fig. 3 for the weighted root mean square acceleration amplitude in each of the above axes. If a particular task demands greater accuracy than that indicated by the relationship in Fig. 3, further precautions such as those outlined in Annex B of this Leaflet shall be taken to ensure that the task is not significantly degraded by vibration.

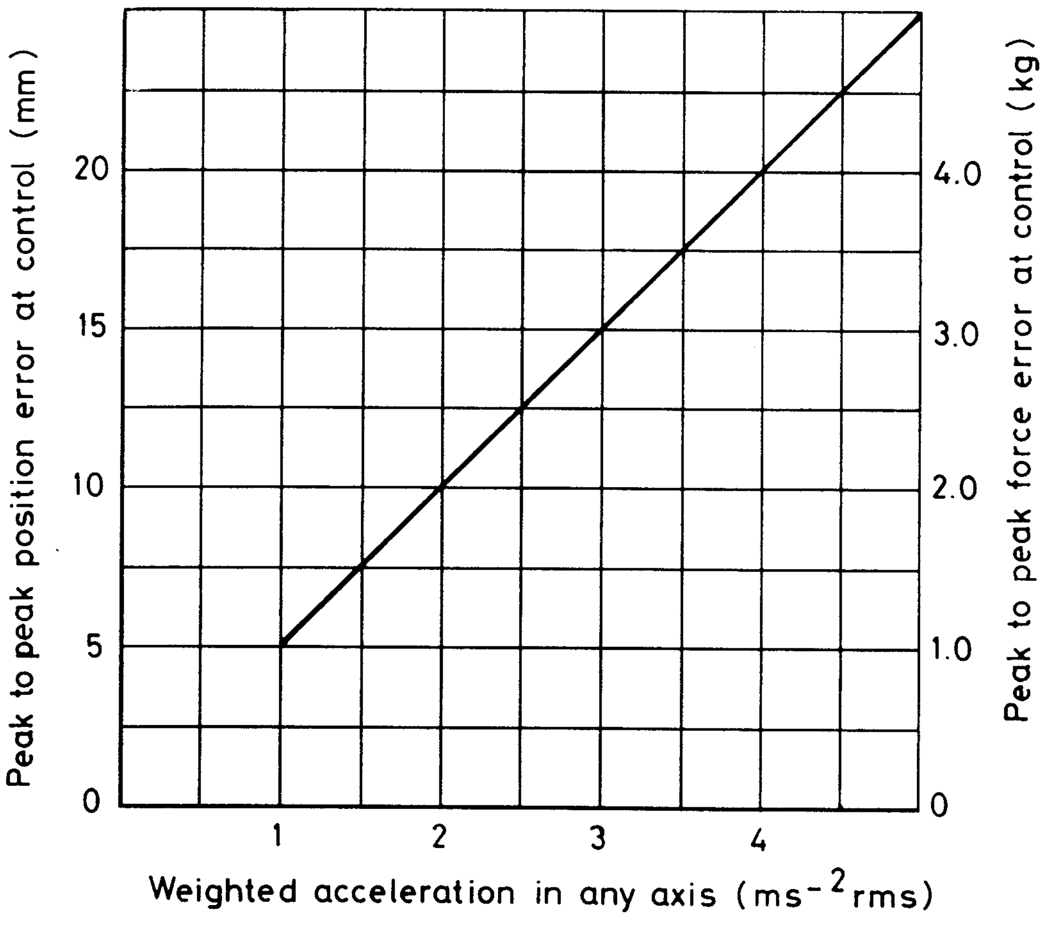


FIG.3 RELATIONSHIP BETWEEN WEIGHTED ROOT MEAN SQUARE ACCELERATION AMPLITUDE AND TRACKING ERROR

4 EFFECTS OF VIBRATION ON VISUAL PERFORMANCE

4.1 INTRODUCTION

4.1.1 As the transmission of vibration to the head and eyes is dependent on the transmissibility of both the seat and the body the eyes will not vibrate with the same amplitude and phase as the surrounding cockpit structures above 1 or 2 Hz. Moreover, vibration at the seat results in both rotational and translational head motion. Differential motion between the retinal image and the eye causes retinal blur, which degrades the ability of the aircrew member to see fine detail. The effect is worse for near objects, such as cockpit instruments, than for objects at optical infinity because the effect of translational motion decreases with distance.

4.1.2 The procedures outlined in this para define minimum values for the sizes of characters or symbols on cockpit displays. The values represent the size of the smallest character or symbol which can be read without increased risk of error due to either whole body or instrument vibration. They assume otherwise favourable viewing conditions, and the minimum character size may need to be increased if this is not the case. Factors which may affect the extent to which visual performance is degraded by vibration are discussed in Annex C.

4.2 EVALUATION OF THE EFFECTS OF WHOLE BODY VIBRATION ON VISUAL PERFORMANCE

4.2.1 The weighted root mean square acceleration amplitude in the three translational axes on the seat, and in the x-axis on the seat back, should be computed with the same weightings as defined in para 3.2, for the evaluation of the effects of vibration on hand control.

4.2.2 Evaluation procedures used elsewhere do not take account of seat-back vibration (see Annex D Refs 12, 13). It is evaluated here because x-axis vibration may occur at relatively high acceleration amplitudes in rotorcraft and there is experimental evidence to show that, when transmitted via the seat-back, this can have a significant effect on visual performance (see Annex D Ref 20).

4.2.3 Alphanumeric characters on the instruments and displays in the rotorcraft shall not be smaller than the minimum size defined by the limit in Fig.4 for the highest weighted root mean square acceleration amplitude in the above axes. If the viewing conditions are not optimum, as defined in Annex C, it may be necessary to increase the character size above the minimum limit to avoid reading difficulties.

4.2.4 Detailed non-alphanumeric symbols on displays shall be treated in the same way as alphanumeric characters, but consideration should be given to the factors discussed in Annex C.

4.2.5 If instruments are normally used during short periods of high amplitude vibration, such as in transitional flight, the symbols and characters on those instruments should be larger than those on other instruments, in accordance with Fig.4. Any character or symbol which is not necessary for the safe operation of the rotorcraft under the particular vibration conditions need not conform to these standards.

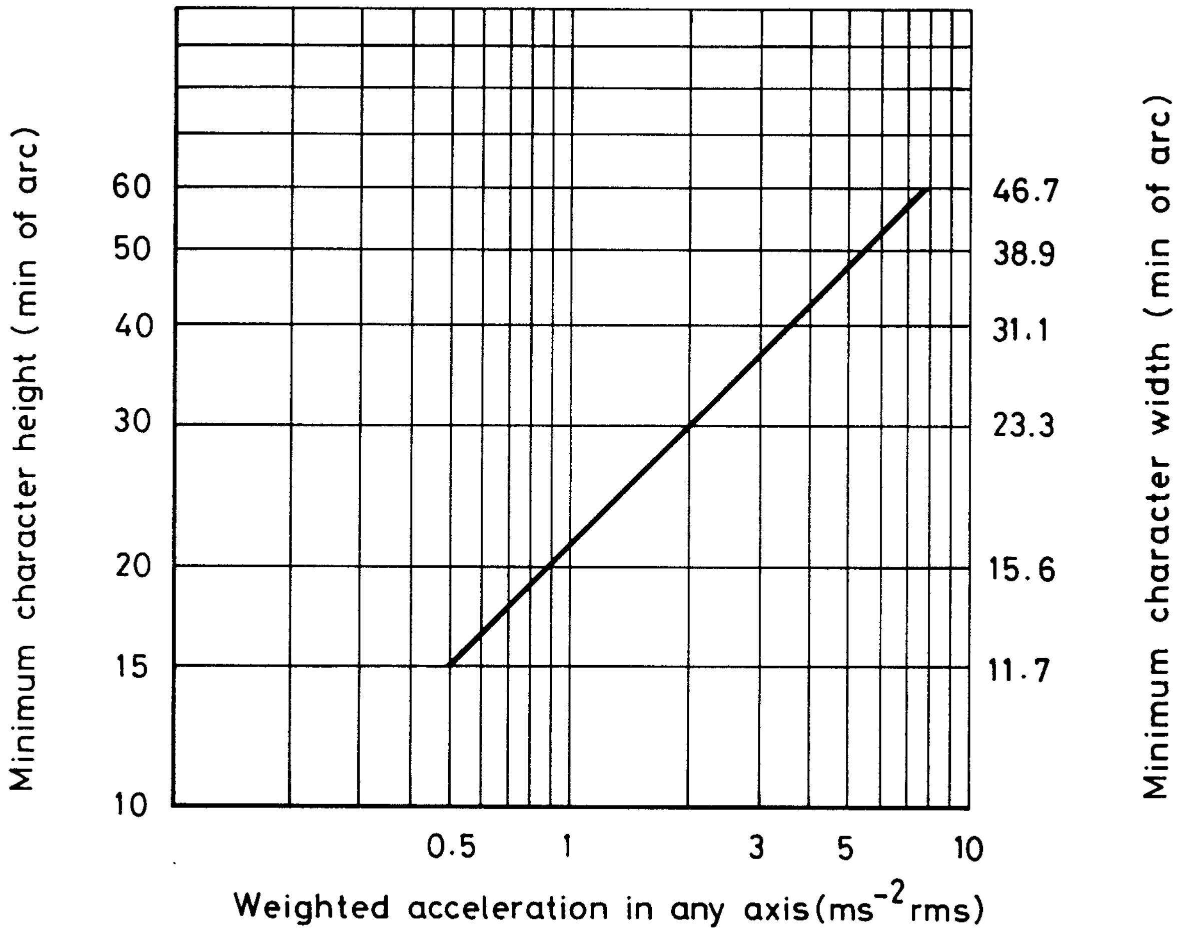


FIG.4 RELATIONSHIP BETWEEN WEIGHTED ROOT MEAN SQUARE ACCELERATION AMPLITUDE AND THE MINIMUM CHARACTER OR SYMBOL SIZES ON INSTRUMENTS AND DISPLAYS.
(The precise values are based on a 9 by 7 font)

4.3 EVALUATION OF THE EFFECTS OF INSTRUMENT VIBRATION ON VISUAL PERFORMANCE

4.3.1 The weighted root mean square acceleration amplitude in the vertical and lateral horizontal (side to side) axes on the instrument panel or instrument face shall both be computed with weighting W_2. The relationship between minimum character or symbol size and the weighted root mean square acceleration amplitude in each axis shall be determined in the same way as for whole-body vibration, as outlined in para 4.2 above.

ANNEX A

MEASUREMENT SYSTEM DETAILS AND PERFORMANCE

1 PERFORMANCE REQUIREMENTS FOR ACCELEROMETERS, SIGNAL CONDITIONING AND RECORDING APPARATUS

1.1 The accelerometers, signal conditioning and recording apparatus should all be capable of reproducing acceleration signals in the frequency range from less than 0.4 Hz to above 100 Hz with an accuracy better than 0.5 dB. The noise level in the system, including accelerometer cable noise, should be less than 0.1 ms^{-2} rms at frequencies within the measurement bandwidth and the system should be capable of capturing the largest peak accelerations without saturation or distortion.

2 CALIBRATION

2.1 Care should be taken to ensure the accurate calibration of the measurement system and if acceleration time histories are recorded, calibration signals of known acceleration amplitude should be recorded on the same medium, via the same signal conditioning system, a short time before and after the recordings are made.

3 ACCELEROMETER MOUNTING DETAILS

3.1 MEASUREMENTS ON THE SEAT AND SEAT BACK

3.1.1 Vibration measurements on the seat and seat back should be made by accelerometers mounted in an SAE pad or SITBAR (see Annex D Refs 24, 28). Details of the SAE pad and SITBAR are given in Fig. 6. For the vibration measurements on the seat the pad should be placed between the crew member and the seat with the accelerometers orientated in the x-, y- and z-axes as defined in Fig.5. The pad should be placed such that the ischial tuberosities of the crew member are located centrally on the pad. For the vibration measurements on the seat back the pad should be placed between the highest part of the back of the crew member in contact with the seat and the seat back with the accelerometer orientated in the x-axis as defined in Fig.5

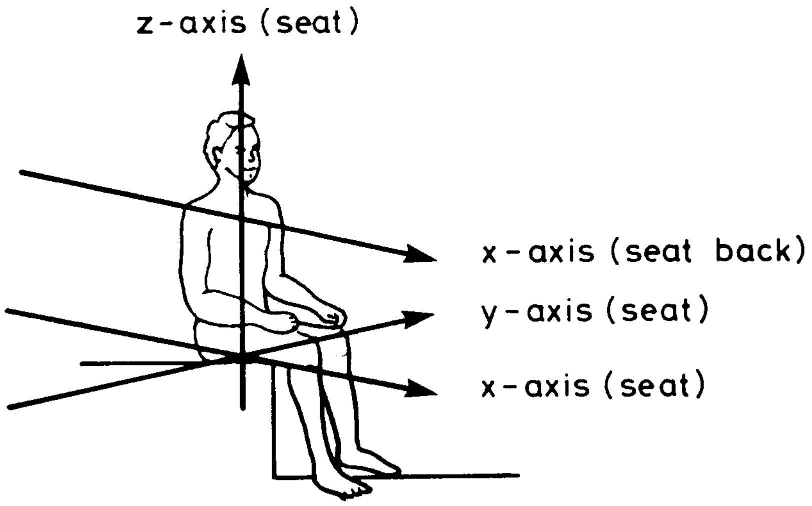
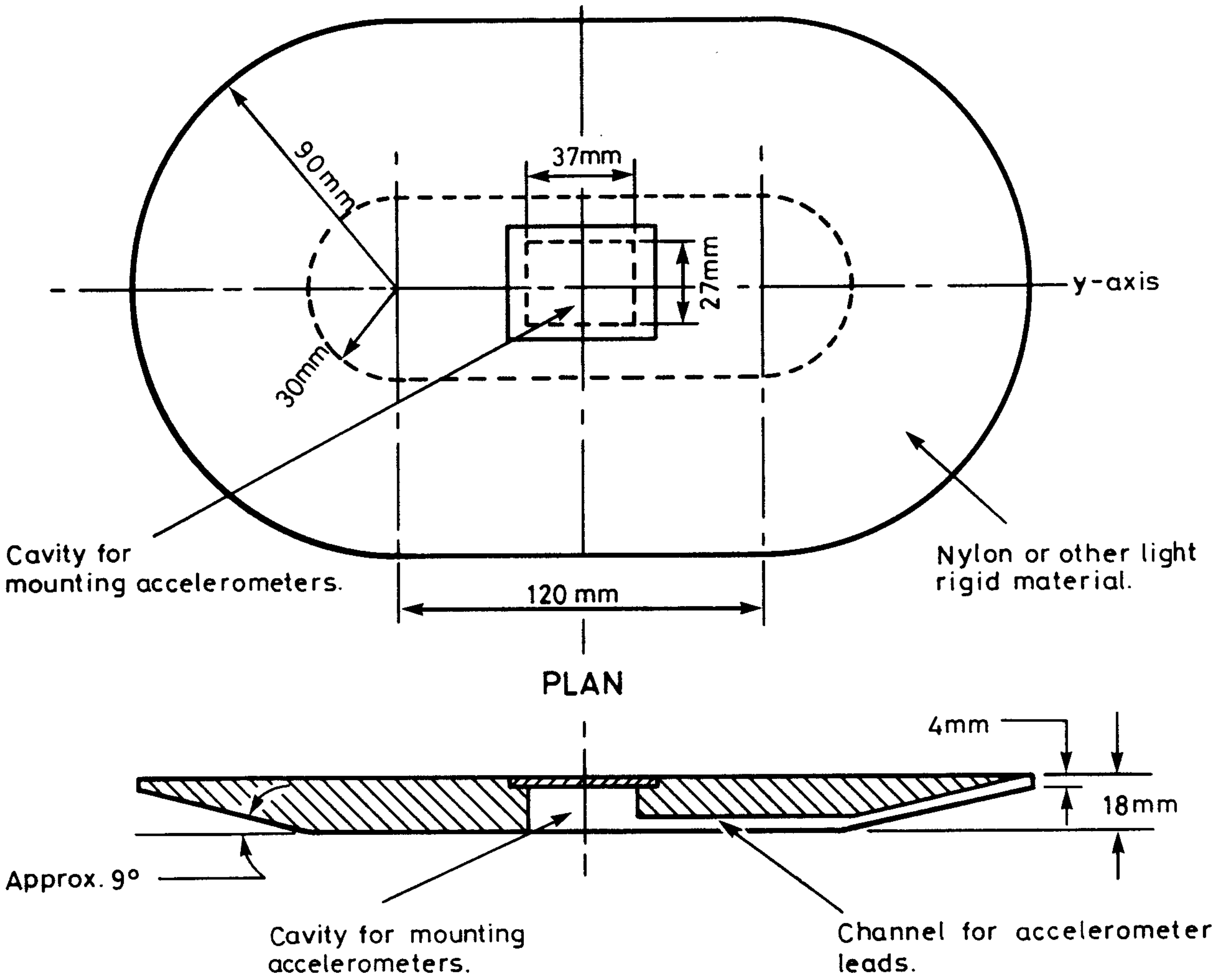
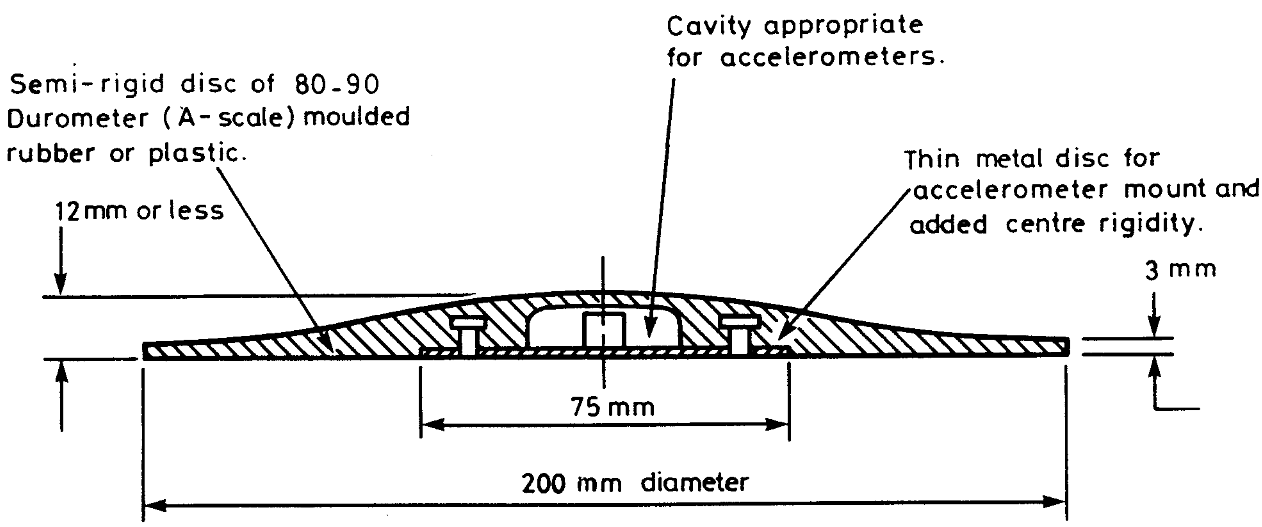


FIG.5 MEASUREMENT AXES FOR SEAT ACCELERATION



(i) SIT - BAR



(ii) SAE PAD

FIG.6 SUGGESTED DESIGNS FOR ACCELEROMETER MOUNTS FOR MAKING VIBRATION MEASUREMENTS AT THE SEAT AND SEAT-BACK INTERFACE

3.2 MEASUREMENTS ON THE FLOOR

3.2.1 For measurements of floor vibration an accelerometer shall be firmly attached either to the cockpit floor near to the feet of the aircrew or on the floor near the pedal controls on which the feet are placed. The accelerometer shall be orientated in an axis approximately parallel to the z-axis at the seat.

3.3 MEASUREMENTS ON INSTRUMENTS AND DISPLAYS

3.3.1 For measurements of display vibration the accelerometers shall be rigidly attached to the instrument console, or to the face of an instrument. The accelerometers shall be orientated in axes approximately parallel to the z- and y-axes at the seat.

4 BAND LIMITING FILTERS

4.1 The recorded signals should be passed through band limiting filters to give an attenuation of -3 dB at 0.4 Hz and 100 Hz and an attenuation rate of at least ± 12 dB per octave outside these limits. The response at frequencies between 1 Hz and 80 Hz should vary by less than ± 1 dB.

5 FREQUENCY WEIGHTING PROCEDURES

5.1 TIME DOMAIN ANALYSIS PROCEDURE

5.1.1 The frequency weighting may be accomplished either by filtering the signal, using analogue or digital circuitry, before computing the root mean square acceleration amplitudes, or by frequency domain techniques (see para 5.2). If a filter is used the modulus of the filter transfer function should not depart from that defined in Table 1 by more than ± 1 dB within the frequency range from 0.5 Hz to 80 Hz. The root mean square amplitude of the filtered acceleration time history should preferably be computed by means of true integration over the entire length of the time history (for statistical accuracy this should be more than 20 secs duration or for the period of each flight mode). An alternative method would be to use an instrument giving a running rms indication. If the latter technique is used the averaging time constant should be at least 10 secs and the duration of the acceleration time history should be long enough for the output to reach a steady state.

5.2 FREQUENCY DOMAIN ANALYSIS PROCEDURE

5.2.1 If suitable frequency analysis equipment is available the weighted root mean square acceleration amplitude may be computed from the following relationship:

$$a_w(rms) = \left(\int_{0.5\,Hz}^{80\,Hz} Gaa(f) \left| H(f) \right|^2 df \right)^{\frac{1}{2}}$$

where $G_{aa}(f)$ is the power spectral density of the acceleration time history and H(f) is the frequency weighting function. In this case the idealised frequency weighting function defined in Leaflet 501/3 Fig.1 and Table 2 may be used. The resolution of the frequency analysis should be 1 Hz or less.

TABLE 1
DEFINITION OF FREQUENCY WEIGHTING FUNCTIONS

Frequency weighting function W1 is the equivalent to a four pole filter with the transfer function:

$$H(s) = \frac{K.\left(1+s\tau_1\right)\left(1+\frac{s.\tau_3}{Q_2}+s^2\tau_3^2\right)}{\left(1+\frac{s.\tau_2}{Q_1}+s^2\tau_2^2\right)\left(1+\frac{s.\tau_4}{Q_3}+s^2\tau_4^2\right)} \quad *$$

Frequency weighting functions W_2 and W_3 are equivalent to two pole filters with the transfer functions:

$$H(s) = \frac{K.\left(1+s\tau_1\right)}{1+\frac{s.\tau_2}{Q_1}+s^2\tau_2^2} \quad *$$

* s is the Laplace operator and is equivalent to jω

where $j = \sqrt{-1}$ and ω is the forcing frequency in rad. s^{-1}

The coefficients of the above expressions are tabulated below:-

	W_1	W_2	W_3
K	0.040	1.00	0.42
τ_1 (seconds)	0.00995	0.0796	0.107
τ_2 (seconds)	0.00995	0.0796	0.0300
τ_3 (seconds)	0.0633	-	-
τ_4 (seconds)	0.0400	-	-
Q_1	0.63	0.63	0.68
Q_2	0.80	-	-
Q_3	0.80	-	-

TABLE 2: DEFINITION OF STRAIGHT LINE APPROXIMATIONS TO WEIGHTING FUNCTIONS

Frequency (Hz)	Weighting Factors (dB)		
	Weighting W_1	Weighting W_2	Weighting W_3
0.5	-8	0	-6
1.0	-8	0	-6
1.25	-8	0	-5
1.6	-8	0	-4
2.0	-8	0	-3
2.5	-6	-2	-2
3.15	-4	-4	-1
4.0	-2	-6	0
5.0	0	-8	0
6.3	0	-10	0
8.0	0	-12	0
10.0	0	-14	-2
12.5	0	-16	-4
16.0	0	-18	-6
20.0	-2	-20	-8
25.0	-4	-22	-10
31.5	-6	-24	-12
40.0	-8	-26	-14
50.0	-10	-28	-16
63.0	-12	-30	-18
80.0	-14	-32	-20

ANNEX B

FACTORS AFFECTING MANUAL CONTROL PERFORMANCE IN THE ROTORCRAFT VIBRATION ENVIRONMENT

1 DYNAMICS OF THE CONTROLLED SYSTEM

1.1 A portion of the error in position or force at the hand which is described in Leaflet 501/3 para 3.2 consists of involuntary control activity directly induced by the vibration. Where the dynamics of the controlled system involve first or higher order lags a large proportion of this higher frequency activity will be filtered out (see Annex D Refs 1, 15, 17, 19, 27). The overall effect of vibration on a manual control system may therefore be less in higher order systems, than in simple positioning tasks. In some circumstances it may be advantageous to introduce lags into position control tasks (such as cursor controls) in order to reduce the effects of the vibration. However, it should be borne in mind that such measures will tend to reduce overall tracking accuracy, particularly of fast moving targets (see Annex D Ref 17).

2 CONTROL FORCE AND RESISTANCE

2.1 Stick type controls should always incorporate some resistance proportional to applied force. The control gain or control sensitivity should then be specified as control output per applied force. There is generally an optimum range of control sensitivities for the performance of a particular task (see Annex D Ref 6, 16). Decreasing the control sensitivity will tend to reduce the level of vibration-induced control activity and so reduce the effect of vibration on control performance. Decreasing the control sensitivity beyond the optimum range will also reduce the overall control performance, however the optimum range of control sensitivity will probably be a little lower in moderate to severe vibration environments. The optimum control sensitivity will depend on the bandwidth of the control task, higher control gains being more appropriate when higher frequency responses are necessary.

2.2 Non-linear control resistance such as statistic and coulomb friction having a damping effect on the control, which tends to reduce vibration induced control activity. However, when the friction is large the overall control performance is likely to be significantly degraded. Pure force controls, which have no frictional damping, perform particularly well when no vibration is present, but tend to be susceptible to vibration effects (see Annex D Refs 1, 15).

2.3 For rate control applications in the vibration environment it may also be advantageous to implement or increase breakout forces or backlash in a control, but care should be taken to ensure that the control efficiency is not impaired.

3 CONTROL TYPE AND LIMB RESTRAINTS

3.1 Stick controls may be operated by the whole arm, hand, finger or thumb. Which is appropriate depends on such factors as the bandwidth of the task, the control forces and what is being controlled. In each case the designer should ensure that adequate support is given to the controlling limb. If the control is a hand operated side arm stick, a suitable armrest should be provided. In the case of a thumb operated control, a hand support may be provided to be gripped by the fingers. In each case considerable care should be taken in the design of the support to ensure that the transmission of vibration from the rotorcraft structure to the controlling limb is minimised (see Annex D Refs 17, 26).

3.2 The advent of fly by wire control systems has made it possible to implement primary flight controls by miniature, side arm sticks. A miniature side arm stick may perform as well as a conventional centre stick provided that adequate limb support is provided (see Annex D Ref 25, 27).

3.3 Rolling ball controls may be particularly suitable for positioning tasks (such as cursor control) in vibration environments provided that the response speed is not critical. They are not, however, suited to continuous tracking tasks. Other forms of rotary controls, such as knobs and wheels, do not necessarily provide advantages over sticks in the vibration environment (see Annex D Ref 14).

4 FOOT CONTROLS

4.1 There are some data which indicate that the above principle based on studies of the effect of vibration on hand control also apply to the effects of vibration on foot control movements.

ANNEX C

DISPLAY FACTORS AFFECTING VISUAL PERFORMANCE IN THE ROTORCRAFT VIBRATION ENVIRONMENT

1 VIEWING DISTANCE

1.1 Under no circumstances should the viewing distance be less than 0.25m. Where large translational motions of the head may occur, greater viewing distances will be advantageous provided that the angular size of the display is not reduced (see Leaflet 501/3 para 4.2). A viewing distance of 0.5 to 0.75m is advisable for optimum visual performance, particularly where the aircrew are frequently looking out of the rotorcraft. The angular size of the characters should be determined for the maximum possible viewing distance which can occur under operational conditions (see Annex D Refs 2, 7, 18).

2 CHARACTER SEPARATION

2.1 The vertical separation between characters and symbols should be not less than half the recommended height for alphanumeric characters. (See Leaflet 501/3 para 4.2). It is recommended that characters and symbols should be separated horizontally by a distance of greater than quarter of the maximum width of the characters. Large horizontal separations (i.e., greater than 3/4 of the maximum character width) between associated alphanumeric characters may not be conducive to good legibility.

3 FONT SHAPE

3.1 The alphanumeric font should be such that confusion between the most important and most frequently used characters is minimised for optimal visibility under vibration conditions. On dot matrix displays the characters should be at least 9 elements high by 7 elements wide. The Huddleston and Lincoln-Mitre fonts both produce acceptable results under vibration (see Annex D Ref 21). Cursive displays may require separate consideration: there is currently little experimental data to define a recommended form.

4 LUMINANCE AND CONTRAST

4.1 The optimum luminance of characters on an electronic display will depend on the ambient illumination. For optimum visibility under vibration conditions the luminance should be such as to produce a contrast ratio for the characters over the background of between 4:1 and 16:1 although higher contrast ratios may be acceptable in some applications. It may be necessary to vary the display luminance over a wide range to maintain acceptable contrast levels where the display face can be illuminated by direct sunlight or where other bright objects may cause reflection or glare problems (see Annex D Refs 3, 22).

ANNEX D

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PART 5 APPENDIX No. 2
AERO-ELASTICITY AND VIBRATION
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 500: AERO-ELASTICITY

500	MIL-H-8501	HELICOPTER FLYING AND GROUND HANDLING QUALITIES; GENERAL REQUIREMENTS FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	AFGS-87221	AIRCRAFT STRUCTURES GENERAL SPECIFICATION FOR
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

500 1.

2. STRUCTURAL DISTORTION

500 2.	MIL-S-8698	PARA: 3.1.3.2

3. FLUTTER

500 3.	MIL-T-8679	PARA: 3.5
	MIL-S-8698	PARA: 3.6.2, 3.6.3
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4. GROUND AND AIR RESONANCE

500 4.	MIL-H-8501A	PARA: 3.7.3
	MIL-T-8679	PARA: 3.5.2
	MIL-S-8698	PARA: 3.6.4, 3.6.5.2.2
	MIL-F-83300	PARA: 3.8.11

CHAPTER 501: VIBRATION AND INTERNAL NOISE

501	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-STD-2164	ENVIRONMENTAL STRESS SCREENING PROCESS FOR ELECTRONIC EQUIPMENT
	MIL-H-8501	HELICOPTER FLYING AND GROUND HANDLING QUALITIES; GENERAL REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II-ROTARY WING AIRCRAFT
	MIL-STD-1789	SOUND PRESSURE LEVELS IN AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION

501 1.

2. VIBRATION DESIGN REQUIREMENTS FOR ROTORCRAFT

501 2.	MIL-H-8501A	PARA: 3.7
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4. VIBRATION REQUIREMENTS FOR ROTORCRAFT EQUIPMENT

501 4.	MIL-STD-810D	PARA: 4.5.7, 5
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PART 6

FLIGHT AND GROUND HANDLING QUALITIES

CONTENTS

APPENDIX No 1 **FLIGHT AND GROUND HANDLING QUALITIES FOR MILITARY DERIVATIVES OF CIVIL ROTORCRAFT***
(Note: See relevant para of this Appendix for military derivative requirements relating to particular chapters of Part 6)

APPENDIX No 2 **U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS**

***** In Preparation

CHAPTER 600

GENERAL REQUIREMENTS AND DEFINITIONS

1 INTRODUCTION

1.1 Chapter 600 introduces the terms of reference against which handling quality requirements can be established. A number of handling requirements which are generally applicable to all channels are also included. Requirements which are more specifically related to particular channels are dealt with in subsequent chapters.

1.2 In many cases handling requirements are inevitably subjective in nature, and in others adequate data on which to quantify them is not currently available. Consequently, the mandatory requirements, stated in the Chapters of Part 6, are often expressed in qualitative terms. Additional explanatory and advisory material on how compliance with the requirements might be demonstrated is included in Leaflets associated with each Chapter.

2 SCOPE AND APPLICABILITY

2.1 See Leaflet 600/1.

2.2 GENERAL APPLICABILITY
2.2.1 The requirements of Part 6 shall apply to:

(i) All types of rotorcraft.

(ii) All relevant configurations of a particular type, where this is variable.

(iii) Any rotorcraft of a particular type that has been manufactured and rigged within the most critical combination of allowable design tolerances.

(iv) All ranges of loading, flight conditions, ambient environmental conditions, and functional states of the rotorcraft encountered in executing the missions called for by the Rotorcraft Specification, and any other usage, reasonably envisaged, for the rotorcraft.

2.3 PENALTIES
2.3.1 The Contractor shall advise the Rotorcraft Project Director of any special penalties of cost, delivery, performance or maintenance which might arise from meeting any particular requirement of Part 6.

3 ROTORCRAFT STATES

3.1 The overall operating state of the rotorcraft is defined by the selected configuration, loading situation and the functional status of relevant sub-systems, see Leaflet 600/1.

3.2 CONFIGURATIONS
3.2.1 The requirements of Part 6 shall apply to any configuration associated with a particular mission flight phase and, where relevant, to the transition from one configuration to another.

3.3 LOADING CASES

3.3.1 The contractor shall declare a loading envelope for each role of the rotorcraft, defining the locations of the centre of gravity over the range of permitted all-up-weight, including the effects of fuel usage, disposable loads and internally and externally carried stores.

3.3.2 Assessments of handling characteristics shall embrace the most critical combinations of all-up-weight and centre of gravity location for the critical configuration.

3.4 NORMAL OPERATING STATE

3.4.1 The Normal Operating State is defined as that in which all rotorcraft systems are functioning normally, with no component failures, and any modes of automatic control introduced to achieve particular mission flight phases are engaged.

3.4.2 Unless otherwise stated the requirements of Part 6 refer to the Normal Operating State.

3.5 FAILURE STATES

3.5.1 The contractor shall, to the satisfaction of the Rotorcraft Project Director, identify the rotorcraft system or component failures that significantly affect the handling qualities in terms of the transient response of the rotorcraft to the failure, and any impact on completion of the sortie with that system failed. Consideration shall be given to failure propagation.

3.5.2 The degree of failure survival to be incorporated will be defined by the Rotorcraft Specification.

3.5.3 Certain components, systems, or combinations thereof may have extremely low probability of failure during a given flight. Failures of this type are classified as Special Failure States and may be very difficult to predict with accuracy. Special Failure States of this type need not be considered in complying with the requirements of Part 6 if justification for considering the Failure States as Special is submitted by the Contractor and approved by the Rotorcraft Project Director.

4 OPERATIONAL PHASES

4.1 It is necessary to consider the handling qualities of the rotorcraft over the whole spectrum of operations, including both non-flight and in-flight phases. The latter can be further categorized as Active, Attentive or Passive, according to the degree of pilot involvement in actually flying the rotorcraft. See Leaflet 600/2.

4.2 For any particular rotorcraft different handling quality characteristics assume differing degrees of relevance in these various phases, while others are commonly applicable. The contractor shall ensure that adequate handling qualities are achieved in those phases relevant to each particular rotorcraft operation.

5 FLIGHT ENVELOPES

5.1 The requirements of Part 6 make reference to the various envelopes defined in paras 5.2 to 5.5. Unless otherwise stated this assumes Normal Operating State of the rotorcraft. Under certain Failure States more restrictive envelopes may be prescribed, subject to agreement with the Rotorcraft Project Director.

5.2 OPERATIONAL FLIGHT ENVELOPE

5.2.1 An Operational Flight Envelope refers to the combination of flight condition parameters, (e.g., airspeed, altitude, temperature, weight, load factor), encountered in accomplishing any particular mission or role called for in the Rotorcraft Specification.

5.3 SERVICE FLIGHT ENVELOPE

5.3.1 The Service Flight Envelope encompasses the flight conditions over which the rotorcraft can be continuously operated without risk of exceeding the Permissible Flight Envelope.

5.3.2 The Service Flight Envelope is based on rotorcraft design considerations and will embrace all operational flight envelopes, in terms of sustained flight conditions and manoeuvring from one flight condition to another.

5.3.3 The rotorcraft shall not be deliberately operated outside of the Service Flight Envelope.

5.3.4 In establishing the overall Service Flight Envelope the Contractor will define the following constituent envelopes, for all relevant configurations and loading conditions (Leaflet 600/3, para 2):

(i) Low speed envelope.

(ii) Forward flight envelope.

(iii) Sideslip envelope.

5.4 PERMISSIBLE FLIGHT ENVELOPE

5.4.1 The Permissible Flight Envelope is the outermost boundary of flight conditions to which the rotorcraft may be flown. From all points in the Permissible Flight Envelope, it shall be possible to return to the Service Flight Envelope without exceptional skill or technique on the part of the pilot.

5.5 ROTOR SPEED LIMITATIONS

5.5.1 The contractor shall define the maximum and minimum, steady and transient, rotor speeds allowable in power-on and power-off conditions throughout the Permissible Flight Envelope.

6 AMBIENT ENVIRONMENTAL CONDITIONS

6.1 PREVAILING ATMOSPHERIC CONDITIONS

6.1.1 The Rotorcraft Specification will define the ranges of altitude, temperature, levels of precipitation, potential icing and wind speeds in which the rotorcraft will be required to operate.

6.2 ATMOSPHERIC DISTURBANCES

6.2.1 Definition of Conditions. The Rotorcraft Specification will define the atmospheric disturbances and turbulence levels in which the rotorcraft will be required to operate.

6.2.2 Compliance. Unless otherwise stated, quantitative requirements of Part 6 assume steady atmospheric conditions and ideally the rotorcraft should be assessed against these requirements in conditions of no more than light turbulence. Continuous turbulence and discrete gust disturbances to be considered in the design process are described in Leaflet 600/4.

6.3 SEA STATE CONSIDERATIONS

6.3.1 Shipborne Operations. In addition to the critical wind vector envelope (Leaflet 600/3, para 2.1), the Rotorcraft Specification will, where relevant, define the limits of deck motion under which the rotorcraft is to be operated.

6.3.2 Airborne Operations. Some automatic flight control system sensors measure displacement of velocity relative to the sea surface and the adverse effects of surface characteristics on overall flight behaviour must be guarded against, (para 10.2.5).

6.4 GROUND SURFACE CONDITIONS

6.4.1 Rotorcraft on Ground

(i) Unless otherwise stated any requirement of Part 6 relating to taxiing, running take off or run-on landing assumes a dry, firm, substantially flat and level surface.

(ii) The consequences of any repetitive irregularities to be expected in such surfaces shall be taken into account.

(iii) The Rotorcraft Specification will define any slope of the ground surface (together with rotorcraft orientation relative to the slope) from which, or upon which, the rotorcraft is required to operate.

7 LEVELS OF HANDLING QUALITIES

See Leaflet 600/5 for factors affecting perceived handling qualities**.**

7.1 Definition of Levels. The requirements of Part 6 are in general expressed in terms of the minimum standard of handling quality parameter necessary top attain particular levels of acceptability, defined as:

LEVEL 1: Handling qualities clearly satisfactory for the mission flight phase being undertaken.

LEVEL 2: Handling qualities acceptable to accomplish the mission flight phase, but with some degradation in mission effectiveness and/or an increase in pilot workload.

LEVEL 3: Handling qualities such that the rotorcraft can be controlled but mission effectiveness is clearly unsatisfactory and/or total workload approaching the limit of pilot capability.

These LEVELS 1, 2 and 3 are associated with Cooper-Harper Ratings of 1 to 3, 4 to 6 and 7 to 8 respectively. (See Leaflet 900/3).

7.2 APPLICATIONS OF LEVELS

7.2.1 Normal Operating States and no more than "Light" Turbulence.

(i) Handling qualities shall not fall below LEVEL 1 in the Operational Flight Envelope referred to in para 5.2.

(ii) Handling qualities shall not fall below LEVEL 2 throughout the Service Flight Envelope defined in para 5. 3.

7.2.2 Failure Operating States. Degradations in levels of handling qualities following failures may be allowed, by agreement with the Rotorcraft Project Director, if the probability of reaching levels lower than those given in para 7.2.1 is sufficiently low.

7.2.3 Operation in Atmospheric Turbulence. The Rotorcraft Specification will define the higher intensities of turbulence at which degraded levels of handling qualities will be allowed.

8 BASIC OPERATIONAL REQUIREMENTS

8.1 MISSIONWORTHINESS

8.1.1 With the rotorcraft in its Normal Operating State, or such Failure States as may be defined in the Rotorcraft Specification, the ability to execute the roles or missions called for by the Rotorcraft Specification shall not be curtailed by any handling deficiency.

8.1.2 Execution of the missions called for in the Rotorcraft Specification shall not cause the pilot undue fatigue, related to the duration or nature of the sortie, or require exceptional skill or physical strength.

8.1.3 For all cases of applicability given in para 2.2 it shall be possible for the pilot:

(i) To achieve any flight condition within the Service Flight Envelope.

(ii) To fly the rotorcraft from any condition to any other condition in the Service Flight Envelope without requiring undue skill or physical strength.

(iii) To readily return within the Service Flight Envelope from any condition in the Permissible Flight Envelope without exceeding any rotorcraft design limit or using exceptional skill or strength.

8.2 MANOEUVRABILITY AND STABILITY

8.2.1 If stability and control augmentation devices are incorporated in order to meet requirements of Part 6 these devices shall have acceptable reliability and failure mode characteristics.

8.2.2 A high degree of stability, imparted for good regulation under adverse operating conditions, (e.g., atmospheric turbulence, or IMC), shall not prevent the required manoeuvre capability being achieved.

8.2.3 Control augmentation, introduced to satisfy para 8.2.2, shall not lead to any difficulties in control due to over-sensitiveness, or to the generation of dangerous flight conditions referred to in para 8.3.

8.2.4 Where the inherent stability and manoeuvre characteristics of the rotorcraft are augmented by artificial means, the desired changes in overall behaviour between one flight phase and another shall be achieved with an absolute minimum of action by the pilot to select the appropriate mode.

8.3 AVOIDANCE OF DANGEROUS FLIGHT CONDITIONS

8.3.1 When operating close to boundaries of the Service Flight Envelope, the rotorcraft behaviour in response to control inputs, outside disturbances or change in operating state, shall be sufficiently progressive in nature and give unambiguous indications of any approach to dangerous flight conditions, to allow adequate time for the pilot to take corrective action.

8.3.2 Combinations of flying control characteristics and rotorcraft response characteristics shall not result in the generation of dangerously high structural loads due to mishandling without the pilot being aware of an excessive control input or the proximity to dangerous flight conditions.

8.4 AUTOROTATIVE FLIGHT

8.4.1 In the event of a total loss of engine power or failure of the transmission system, at any steady or transient condition within the Service Flight Envelope, it shall be possible to establish an autorotative flight condition in which:

(i) Rotor speed can be maintained within the design limitations.

(ii) Adequate control margins remain in each channel.

(iii) The rate of descent and rotorcraft design characteristics permit a safe power-off landing to be made.

8.5 AERODYNAMIC INTERFERENCE EFFECTS

8.5.1 The rotorcraft shall be designed so that any changes of flow condition or rotorcraft configuration, whilst operating in the Service Flight Envelope, shall not give rise to any unexpectedly large or abrupt forces or moments on the rotorcraft, or saturation of autostabilisation systems, that require undue skill or activity in any control channel on the part of the pilot, to prevent excessive changes of attitude, angular rate or acceleration about any rotorcraft axis (see Leaflet 600/3 para 3).

8.6 MINIMUM COCKPIT CREW

8.6.1 Although normal operation of the rotorcraft may employ two pilots, it shall be possible for the rotorcraft to be flown by one pilot, with the overall workload and the rotorcraft characteristics allowing the handing qualities to be rated at LEVEL 3, or better.

8.6.2 Where two piloting stations are provided it shall be possible for the rotorcraft to be flown by one pilot from either station.

8.7 DELEGATED FLIGHT CONTROL

8.7.1 During certain mission phases, for example underslung load manipulation, and winch operation in near-hovering flight conditions, it may be expedient for the pilot to delegate a limited measure of control over the positioning of the rotorcraft relative to external constraints or obstructions, to an authorised crewman stationed to have a better view of the immediate situation than is possible for the pilot. A remotely situated control station to achieve these ends shall meet the following requirements:

(i) The crewman's control system shall be enabled only by the pilot.

(ii) It shall be possible for the crewman's control system to be completely overridden by instinctive action of the pilot through his flying controls, at any time.

(iii) The crewman's control (or controls) shall be self-centring to a datum position. The initial datum shall be commensurate with the rotorcraft flight condition and/or position at the time control is handed over by the pilot. Subsequently, the control may be trimmed to a new datum flight condition established by the crewman.

(iv) The crewman's control(s) shall operate in an instinctive sense to achieve rotorcraft response, with minimal break out forces and a smooth progressive build up in force feel with application of control away from datum.

9 CONTROL CHARACTERISTICS

9.1 The requirements of paras 9.2 to 9.5 relate to the means available to the pilot for the direct control of translational motion along, and rotation about, the pitch, roll and yaw axes of the rotorcraft. In the conventional configuration these controls comprise a centrally mounted cyclic stick, a collective pitch lever, and foot operated rudder pedals. However, this is not intended to preclude alternative flying control configurations, (such as side arm controllers), or controls associated with alternative rotorcraft configurations. Unless otherwise stated the requirements of paras 9.2 to 9.5 refer to any form of controller, although the relative significance of displacement and force characteristics may differ for conventional and alternative forms of flying controls. (See Leaflet 600/6, para 2).

9.2 GENERAL OPERATION OF PRIMARY FLIGHT CONTROLS

9.2.1 Under all operating conditions the controls shall be free of objectionable vibration and discontinuities or irregularities in the forces required to hold a particular control position, or move precisely to another. The forces required to operate the flying controls shall also be well harmonised.

9.2.2 When released by the pilot in any steady flight condition, any control shall retain its position, unless acted upon by a recentering device or a parallel actuator driven by an automatic flight control system.

9.2.3 The flying controls shall be sufficiently well mass balanced, statically and dynamically, so that:

(i) Any residual imbalance does not lead to any limit cycle situation.

(ii) The achievement of the breakout force requirements, particularly those in para 9.3.1 to 9.3.3, is not compromised.

(iii) Following any significant unsolicited disturbance to the rotorcraft, with the controls released as in para 9.2.2, any tendency for the flying controls to move under inertial loading shall be in the sense to counteract the disturbance rather than amplify its effects.

See also Leaflet 600/6, para 3.2.2.

9.2.4 There shall be no objectionable control force felt, or tendency for a flying control to move, when any other control is operated, as a result of control interference or designed control mixing intended to be irreversible.

9.2.5 There shall be no objectionable force felt by the pilot through any flying control, or unacceptable degradation in system performance due to that flying control moving, as a result of any automatic flight control system functioning through a series actuation device in that channel.

9.3 BREAKOUT FORCES

9.3.1 Breakout forces shall not cause piloting difficulties incompatible with the required flight levels (see Leaflet 600/6, para 3).

9.3.2 In the rotorcraft Normal Operating State the breakout force in each primary flying control channel shall be within the limits given by Leaflet 600/6 Table 1.

9.3.3 The breakout force for each control shall be symmetrical about the static point, within a tolerance of 10%, i.e. , the allowable breakout force in one direction shall be in the range 90% - 110% of the breakout force in the other direction.

9.3.4 Following a first failure, or disengagement of one lane, of a blade pitch powered control system the maximum breakout force in any control channel shall not be greater than that shown in Leaflet 600/6 Table 2, and the minimum shall not be less than the minimum in Table 1.

9.3.5 Following the failure or disengagement of any trim, automatic flight control or artificial feel system, the breakout force on any control shall not be less than the minimum shown in Table 1.

9.3.6 Following a flying control jam, or failure of an artificial feel system, the force at the pilot's control to initially overcome the jam shall not exceed the levels shown in Leaflet 600/6 Table 3. During subsequent operation of a control channel affected in this way, the breakout force shall not exceed the levels shown in Leaflet 600/6 Table 4.

9.4 CONTROL TRIM AND ARTIFICIAL FEEL

9.4.1 The requirements of this paragraph relate primarily to conventional control configuration, (para 9.1), in which basic operation of the control surfaces is associated with displacement of the flying controls by the pilot.

(i) A means shall be provided for positive self centring of the longitudinal and lateral flying controls to datum positions determined by the pilot as a result of trimming the control forces to zero, in any flight condition.

(ii) Positive self centring may be applied to other control channels if this is expedient to the execution of the roles required of the rotorcraft.

(iii) All control trimming devices shall be easily operated by the pilot, in an instinctive sense, without having, to release the relevant flying control. For trimming in the collective channel when relevant, the instinctive sense shall be considered to be a rearward (towards pilot) movement to trim to a higher collective setting.

(iv) In addition to maintaining the control in its trimmed position, in accordance with para 9.2.2, the trim system shall also retain the zero force datum while the control is displaced from that position against spring feel, e.g., to manoeuvre the rotorcraft.

(v) If released from an offset position, when held against a self centring spring force, the control shall return to datum crisply, without objectionable overshoot or residual motion.

(vi) The control force trim system shall operate sufficiently rapidly that, during the most extensive, aggressive changes of flight condition dictated by the rotorcraft role requirements, the control forces can be contained, allowing comfortable and precise control of the rotorcraft to be maintained.

(vii) The trim system operation shall not be sufficiently rapid to make precise trimming difficult, through over-sensitivity of the trim control, or to cause unacceptably rapid build up of flying control force or displacement in the event of a trim system runaway.

(viii) The combined effects of breakout and initial force gradient characteristics shall not result in any objectionable discontinuities in feel leading to difficulty in maintaining precise control of the rotorcraft. (See Leaflet 600/6 para 3.3).

(ix) For a progressively increasing displacement from the trimmed position, against a spring feel device, the control force shall continue to increase substantially linearly. (Leaflet 600/6 Table 5).

(x) There shall be some means of readily engaging and disengaging the trim /feel system. There shall be no objectionable uncommanded disturbances to the flying controls upon engagement or disengagement of the trim system.

(xi) Any failure of the trim /feel system shall not result in a restriction of flying control range available.

9.4.2 For alternative control configurations in which control surface operation responds to a pilot-applied force rather than a displacement:

(i) It shall be possible to trim the control force to zero at any steady flight condition.

(ii) Para 9.4.1 (iii) is applicable.

(iii) The control surface setting datum, established as a result of para 9.4.2 (i), shall be retained while a control force is applied to manoeuvre the rotorcraft.

(iv) If a control surface is suddenly released, the associated control surfaces shall return to datum setting without causing any unexpected or objectionable rotorcraft response behaviour.

(v) Paras 9.4.1 (vi), (vii), (viii) and (ix) shall also be applicable.

9.5 STRUCTURAL PROTECTION

9.5.1 If necessary, in order to comply with para 8.3 design consideration shall be given to physically deterring the pilot from applying excessive control inputs. This may be achieved by means of additional force constraints superimposed on the normal feel characteristics referred to in paras 9.3 and 9.4

9.5.2 If an automated sub-system is introduced to satisfy para 9.5.1, that sub-system must have adequate integrity. In particular no failure shall lead directly to a dangerously high rotorcraft loading situation, or result in any objectionable reduction in flying control authority below that necessary for normal operation, manoeuvres and landing under all operating conditions called for by the Rotorcraft Specification.

9.5.3 Non-functioning of a system such as that referred to in para 9.5.2, due to any failure or malfunction, shall be indicated to the pilot.

9.6 SECONDARY CONTROLS

9.6.1 The requirements of para 9.6 refer to those controls, in addition to the primary flying controls, the operation of which involves progressive application by the pilot over a range of travel. This may include for example: engine controls, rotor brake applications, and possible means of effecting rotorcraft configuration changes.

9.6.2 When released by the pilot these controls shall retain their position, unless acted upon by self centring or scheduled actuation devices.

9.6.3 Combined breakout, friction ,restoring or inertial characteristics shall not result in any irregularities in the forces required to operate the controls which make small precise adjustments difficult or time consuming to achieve.

9.6.4 During progressive applications in one direction the operating force shall not reverse, and should not reduce.

10 RESPONSE TO CONTROL INPUTS

10.1 PRIMARY FLYING CONTROLS

10.1.1 The requirements of para 10.1 are applicable to any form of controller, whether or not any automatic flight control system functions are engaged.

10.1.2 The controls shall be operated in an instinctive sense in order to generate motion of the rotorcraft, under all operating states.

10.1.3 Control inputs shall produce rotorcraft displacement, rate or acceleration type of responses along or about each rotorcraft axis, whichever is appropriate to particular flight phases associated with the execution of the roles called for in the Rotorcraft Specification (Leaflet 600/7, para 2.2).

10.1.4 The rotorcraft shall be sufficiently responsive to enable the manoeuvres associated with the missions called for in the Rotorcraft Specification, and safe operation of the rotorcraft, to be executed without excessive control applications in any channel. (Leaflet 600/7, para 3).

10.1.5 The flying controls shall not be over sensitive, to an extent that leads directly to difficulty in establishing or maintaining any desired flight condition, or that promotes pilot-induced oscillations. (Leaflet 600/7, para 3).

10.1.6 Rotorcraft responses to control inputs in all channels shall be well harmonised throughout the Permissible Flight envelope. (Leaflet 600/7, para 4).

10.1.7 Control inputs in a particular channel or axis shall not result in objectionable rotorcraft responses being generated in other channels or axes, that cannot be contained by relatively small (instinctive) corrections through the appropriate control channel (Leaflet 600/7, para 4).

10.1.8 Rotorcraft response to control inputs in all channels shall be continuous, progressive and predictable as the rotorcraft is manoeuvred within, and passes into and out of ground effect.

10.2 FLIGHT UNDER AUTOMATIC CONTROL

10.2.1 The Rotorcraft Specification will define the autostabilisation, autopilot mode and control facilities to be available in the rotorcraft.

10.2.2 When flying, with an automatic flight control system mode engaged, the pilot shall be able to override the system and fly the rotorcraft through the primary flight controls without any conscious switching action to disengage the automatic mode.

10.2.3 In certain automatic flight control system modes of operation, the rotorcraft flight conditions may be adjusted or demanded by control inputs through means, other than the primary flying controls, (e.g., trimmers) which may for example modify stored datum values of relevant flight parameters.

Under these circumstances:

(i) The pilot trim input to produce the required rotorcraft response shall be in an instinctive sense.

(ii) The new value of the flight condition parameter demanded shall be indicated to the pilot unless the response of the rotorcraft is sufficiently prompt and unambiguous, that the pilot has no difficulty in anticipating the appropriate input required to establish the desired flight condition with the minimum delay and/or overshoot.

10.2.4 Any manoeuvre carried out under direct autopilot control shall be repeatable in all normal operating conditions, and predictable throughout the anticipated range of operating conditions.

10.2.5 In the case of the autopilot modes which may be engaged for prolonged operation under agitated environmental conditions (para 6.2.1 and para 6.3.2), system performance design aims shall be such that they do not result in undue rotorcraft motion or system demands which, in conjunction with rotorcraft design characteristics, are likely to lead to:

(i) Rotorcraft design limits being reached.

(ii) Discomfort to occupants.

(iii) Reduction in crew mission effectiveness.

10.2.6 Where rotorcraft response to series actuator runaway, pilot intervention criteria, or system authority requirements dictate that series actuators operate close to their mid positions during sustained flight conditions, the system design shall be such that the actuators are self trimming, or sufficiently close trimming is possible without imposing an objectionable or time consuming work load on the pilot. A warning of actuator offset during non-manoeuvring flights shall also be provided.

10.3 RATE OF CONTROL APPLICATION

10.3.1 The ability to safely operate the rotorcraft and to perform the manoeuvres necessary to meet the role requirements of the Rotorcraft Specification shall not be limited by the rate at which control inputs can be supplied to the blades, or other control surfaces or devices.

10.3.2 The rates of operation of primary flying controls, powered servos and auxiliary series and parallel actuation devices shall not be excessively high or low, that they lead to objectionable rotorcraft response characteristics.

10.4 LOST MOTION
10.4.1 Overall lost motion in any control channel shall not be sufficient to result in any perceptible indeterminacy in control of the rotorcraft or objectionable response characteristics including any tendency to limit cycle oscillations. (Leaflet 600/7 para 5).

10.5 ENGINE CONTROL
10.5.1 In any operating condition within the Ground and Service Flight Envelopes it shall be possible for the pilot to select power and/or drive speed settings, appropriate to the operation of the rotorcraft via the engine controls.

10.5.2 For multi-engine installations it shall be possible for the pilot to readily adjust and match the outputs of all operating engines, in any steady ground operating or flight condition.

10.5.3 The engine control system shall contain adequate means of protection against catastrophic damage due to overspeed, over temperature, and over torque in the event of excessive power demands or failures within the engine system.

10.5.4 The protection devices referred to in para 10.5.3 shall not prevent "emergency" power being obtained when necessary.

10.5.5 Engine response to rapid changes in power demanded as a result of manoeuvre or flying control inputs shall be adequate for the roles of the rotorcraft, and the execution of emergency recovery manoeuvres. The Rotorcraft Specification will define minimum time intervals over which specified increases in engine output shall be achieved. For some tactical roles the rate at which engine power can be shed may also be significant.

10.5.6 The engine control system shall provide adequate stability margins when operating under low power conditions associated with driving electrical, hydraulic or any other form of power generators required by the rotorcraft subsystems in either:

(i) An "Accessory Drive" mode selected on the ground.

(ii) Power-off autorotative flight conditions.

There shall be no need for the pilot to take any specific action to prevent undue fluctuations in the outputs of these generating systems, under these conditions.

11 STABILITY

11.1 Specific stability characteristics in individual axes are dealt with in Chapters 601 to 605, based on transient response criteria described in Leaflets 600/7 and 600/8.

11.2 STATIC STABILITY

11.2.1 The general requirement for positive static stability is that the final control displacement or force to achieve a change in flight condition shall be in the same direction as that required to initiate the change, and this direction shall always be in an instinctive sense.

11.2.2 For LEVEL 1 handling qualities the rotorcraft shall have positive static stability in all axes, for all flight condition throughout the Service Flight Envelope, with the exception of an concessions identified in Chapters 601 to 604 inclusive.

11.3 SHORT TERM DYNAMIC STABILITY

11.3.1 The recovery from a small discrete disturbance to the flight condition shall be characterised by a positive initial return towards datum, followed by short times to pass through datum with small overshoot, and to settle within a final tolerance.

11.3.2 Criteria for this initial recovery are described in Leaflet 600/7, para 6.2. The flight parameters to which these criteria apply will depend on the particular rotorcraft or flight mission phase under consideration, as covered in Chapters 601 to 605 inclusive.

11.4 LONG TERM STABILITY CHARACTERISTICS (APERIODIC)

11.4.1 Where the rotorcraft flight control philosophy calls for small adjustments to the flight conditions to be achieved through single control or trim inputs, any effective time constant of the relevant response parameter shall not be sufficiently long that difficulty in judging the suitability of any particular input leads to repeated corrective inputs in order to achieve the desired flight condition.

11.4.2 Leaflet 600/8 para 3 describes a means of quantifying critical long time constants, in the context of para 11.4.1. Following an appropriate single step control or trim input, the ratio of increments in the relevant response parameter over two consecutive equal time intervals shall be in accordance with values to be specified, where applicable, in Chapters 601 to 605.

11.5 TRANSLATIONAL VELOCITY AND POSITIONAL CONTROL

11.5.1 To exercise precise control over the translational velocity and positioning of the rotorcraft in the Low Speed Envelope, the relevant parameters must respond to logically conceived control inputs with the minimum delay. (Leaflet 600/8, para 4.1).

11.5.2 There shall be a consistent and predictable relationship between a discrete control input from trim and the associated steady state translational parameter increment, over the relevant range of initial flight conditions. (Leaflet 600/8, para 4.5).

1.5.3 Within an acceptably short period of completing a discrete control input the relevant response parameter shall have reached an acceptable value which is subsequently retained within specified tolerance. (Leaflet 600/8, para 4.6).

11.6 LONG TERM STABILITY CHARACTERISTICS (OSCILLATORY)

11.6.1 For LEVEL 1 handling qualities the frequency and damping characteristics of rotorcraft motion in any axis shall not degrade the ability to execute the mission required, or reduce aircrew effectiveness through fatigue or discomfort.

11.6.2 For LEVEL 3 handling qualities, under IFR no obtrusive oscillatory mode with a period less than 20 seconds shall be divergent.

11.7 RESIDUAL OSCILLATIONS
11.7.1 There shall be no tendency for undamped small oscillations to persist to the extent that they interfere with the execution of the roles required of the rotorcraft. Unless stated otherwise in Chapters 601 to 605, for LEVELS 1 and 2, the amplitude of persistent oscillation about any axis, in calm air conditions, shall not exceed ± 0.5 degrees.

11.8 ATMOSPHERIC DISTURBANCES
11.8.1 Following a major discrete atmospheric disturbance which may cause temporary saturation of an engaged automatic flight control system facility, that system shall re-establish its normal control function upon removal of the disturbance without itself having been disoriented or generating excessive overshoot in the rotorcraft response, or giving rise to the need for undue action or skill by the pilot.

12 CONTROL MARGINS

12.1 The background to the general requirements in paras 12.2 and 12.3 is given in Leaflet 600/9 with individual axes treated as appropriate in Chapters 601 to 605.

12.2 RESTORATION OF CONTROL
12.2.1 Having reached any operating point in the Permissible Flight Envelope, there shall be a sufficient margin of control available in each channel to enable the rotorcraft to be safely restored to the appropriate Service Flight Envelope without any design limitations being exceeded.

12.3 MANOEUVRABILITY AND RECOVERY
12.3.1 For any approved loading condition, at any steady flight condition within the Service Flight Envelope, there shall be sufficient margins of control available in the critical sense in each channel:

(i) To generate the angular responses within the given time interval (Leaflet 600/9 para 2).

(ii) To reverse the angular rate resulting from a discrete gust in the critical direction, without design limit loads being, exceeded (Leaflet 600/9, para 3.1).

(iii) To reverse the angular rate resulting from the critical rotorcraft system failure.

whichever is the greatest, having regard to the particular flight phase, pilot intervention time, extent of stability augmentation remaining available.

12.3.2 There shall also be adequate margins available:

(i) To meet the Rotorcraft Specification requirements for Vertical agility in the Low Speed Flight Envelope under critical rotorcraft loading and local environmental conditions.

(ii) To reverse the rate of descent resulting from critical rotorcraft system or engine failures while operating at low height above ground or sea surfaces, or local obstructions.

(iii) To ensure that safe autorotational flight with rotor speed within design limits can be established and maintained from any point in the Service Flight Envelope which is dependent upon a lifting rotor, and at the critical rotorcraft loading condition.

12.4 OPERATION WITH AFCS

12.4.1 In those rotorcraft which depend on automatic flight control systems to meet requirements in Part 6 and:

(i) These systems function through limited authority series actuators in each channel.

(ii) The actuator trimming requirement of para 10.2.6 is met by providing a means of applying separate trim inputs to the series actuators.

the requirements of paras 12.4.2 and 12.4.3 are applicable.

12.4.2 The series actuator trim facility shall have sufficient authority to cope with the most adverse combinations of flight condition throughout the Service Flight Envelope, centre of gravity location and rotorcraft and system tolerances.

12.4.3 In the event of any failure resulting in a hardover in the series actuation system, the requirements of para 12 shall still be applicable.

12.5 FAILURE STATES

12.5.1 No single failure in a control system actuator or servo, trim or feel sub-system or energy supply on which they depend, shall result in a critical reduction in the effective control margin in any channel.

12.6 CONTROL MIXING

12.6.1 No mixing or the interaction on any particular control channel by any other channel shall result in the control margins required by paras 12.1 to 12.5 not being available in the critical flight operation conditions.

13 CARRIAGE AND RELEASE OF EXTERNAL STORES AND LOADS

13.1 EXTERNAL STORES. The requirements of para 13.1 relate to those items of role equipment, armaments etc., which are carried externally to the rotorcraft, on attachments provided specifically for that purpose.

13.1.1 Carriage of Stores. Throughout the Service Flight Envelope the presence of any approved external store shall not result in:

(i) Unacceptable levels of buffeting, vibration or degraded control margins and dynamic response characteristics.

(ii) Interference effects on sensors providing rotorcraft flight condition status information to the pilot through cockpit instrumentation, or to any automatic flight control system.

13.1.2 Release of Stores. At any steady flight condition or manoeuvre condition approved for release:

(i) All stores shall positively separate from the rotorcraft, without contacting any part of the rotorcraft.

(ii) In a multi-store fit, the release of stores in any sequence shall not result in the loss of adequate control margin in any axis to retrim the rotorcraft following the release of any store.

(iii) The transient response of the rotorcraft following the release of any store shall not be excessive or lead to a dangerous flight condition if pilot corrective action is delayed by an appropriate time interval.

13.2 EXTERNAL LOADS

13.2.1 The requirements of para 13.2 relate to loads suspended from the rotorcraft by means of cargo hook or winch, and include recoverable sensors and detectors deployed from the rotorcraft for the execution of tactical operational missions. In addition to the requirements of para 13.1 which are generally applicable, those in para 13.2.2 shall also be met.

13.2.2 Carriage of Suspended Loads.

(i) Because of immense variety in aerodynamic and inertial characteristics of loads which rotorcraft may conceivably be required to carry, each case should be considered individually.

(ii) The contractor shall establish envelopes of flight speed and manoeuvres including methods of attachment and trail lengths, within which specific loads may be safely carried in a controlled manner.

(iii) The overall flight dynamic characteristics of the load and its means of suspension shall be such that the pilot (see also para 8.7) does not have undue difficulty in positioning the load relative to an external reference when this is applicable, in curtailing any tendency for the load to execute large amplitude motion likely to bring it or its suspension into close proximity to critical components of the rotorcraft, and in controlling any feedback effects of forces or moments applied to the rotorcraft by the suspended load.

LEAFLET 600/1

GENERAL REQUIREMENTS AND DEFINITIONS

SCOPE AND APPLICABILITY

1 OBJECTIVES OF PART 6

1.1 The Chapters and associated Leaflets of Part 6 are intended to provide a coherent basis on which to determine the handling qualities required in any rotorcraft, and to provide guidance to the design, procurement and acceptance processes.

1.2 The requirements are based on the needs for the execution of particular roles or mission task elements. Frequent reference is made to the Rotorcraft Specification, as the source definition of the roles or mission tasks to be carried out by any particular project rotorcraft.

1.3 The requirements primarily address the quality of the rotorcraft flying characteristics, and the pilot's interface with the rotorcraft necessary for mission accomplishment, in the most generalised way, without suggesting any particular design solution. The rotorcraft configuration and degree of system complexity selected by the designer are viewed as merely means to an end, namely satisfying the mission-oriented requirements, and particular configurations or types of system do not warrant individual sets of basic requirements.

1.4 A number of handling issues are inherently subjective by nature and this leads to the expression of requirements in qualitative terms. In a number of other cases adequate data with which to quantify specific requirements is not currently available. This also leads to qualitative requirements sometimes backed up by provisional data for initial guidance, in the leaflets. Qualitative requirements are more open to variable interpretation than quantitative ones, but do have a value in drawing attention to issues that need to be addressed even though full supporting data is not available. In the absence of well substantiated data, there is a danger of generating misleading requirements if inappropriate data is applied uncritically. This should be avoided.

1.5 While the fundamental qualitative requirements are less likely to change significantly, the detailed numerical values ascribed to criteria for various levels of handling quality are likely to be refined as a result of accumulated operational experience and continued advances in technology and the application of techniques and research. It is envisaged that these changes will be accommodated by the structure of Part 6, primarily through the issue of updated Leaflets.

2 APPLICABILITY

2.1 ROTORCRAFT TYPES

2.1.1 The requirements of Part 6 are intended to apply to all those aircraft that derive the major part of their lifting capability, over part at least of their operating envelope, from aerodynamic lifting surfaces rotating about a shaft axis.

2.1.2 Para 2.1.1 does not preclude rotorcraft in which lift is derived in some flight regimes from non-rotating wings to partially or completely support the rotorcraft.

2.1.3 The requirements of Part 6 apply to all rotorcraft irrespective of the number or orientation of rotors employed for lifting, propulsive or control purposes.

2.1.4 Rotorcraft that employ auxiliary means of propulsion, in addition to the rotors referred to previously, are not precluded.

2.2 VARIABLE CONFIGURATIONS

2.2.1 The requirements of Part 6 are intended to apply, where relevant to any configuration of a particular rotorcraft in which the orientation of the rotor shaft axis or axes, non rotating aerodynamic surfaces or auxiliary propulsive devices, can be varied relative to the major body axes of the rotorcraft.

2.2.2 Alternative configurations also embrace the retraction and extension of an undercarriage, opening or removal of significantly sized or located doors or hatches, and the re-orientation relative to the rotorcraft body axes, of equipment associated with the execution of specific roles or missions.

2.3 SUB SYSTEM FUNCTIONAL STATUS

2.3.1 Handling of the rotorcraft is directly dependent upon a number of major rotorcraft sub systems such as flying control systems, automatic flight control systems and engine and transmission systems. These in turn are dependent on the sources of energy (electrical, hydraulic, pneumatic etc.,) by means of which they function and are controlled.

2.3.2 The requirements of Part 6 cater for the normal and failure states of these relevant systems through reference to the Rotorcraft Operating State and application of the LEVELS of handling quality to the various criteria.

LEAFLET 600/2

GENERAL REQUIREMENTS AND DEFINITIONS

OPERATIONAL PHASES

1 INTRODUCTION

1.1 This Leaflet categorizes the operational phases identified in Chapter 600 para 4.1, for which rotorcraft handling quality requirements have to be laid down.

2 NON FLIGHT PHASES

2.1 Pre-flight and post-flight operations of the rotorcraft include handling by the flight crew on:

2.1.1 Fixed ground surfaces which may include elevated/restricted clearance platforms.

2.1.2 Ship decks.

2.1.3 Water Surfaces.

2.2 GROUND OPERATIONS

2.2.1 Handling operations by the flight crew with the rotorcraft on the ground include engine start, rotor engagement and stopping, taxiing and running landings and take-offs.

2.3 SHIPBORNE OPERATIONS

2.3.1 Handling operations by the flight crew with rotorcraft on a ship's deck include engine start, rotor engagement and stopping, possible operation of rotorcraft/deck restraint system, and limited taxi operations.

2.4 WATERBORNE OPERATIONS

2.4.1 If normal operation of the rotorcraft involves take-off and landing on water surfaces, this will be stated in the Rotorcraft Specification, together with sea state conditions under which this is expected to be carried out.

2.4.2 Even though take-off and landing from a water surface may not be normal operations, emergency alighting on water, and subsequent control of the rotorcraft may have to be addressed in the clearance of over-water operation of the rotorcraft.

3 BASIC FLIGHT PHASES

3.1 Each flight, or mission, is to be considered as a number of continuous flight phases, each one characterised by varying degrees of emphasis on:

3.1.1 The division of the pilot's attention between actually flying the rotorcraft through the flying controls and other tasks such as navigation, communications and involvement in the current tactical situation.

3.1.2 The needs for highly manoeuvrable or for highly stabilised rotorcraft flight characteristics.

3.1.3 The urgency with which the pilot needs to take action to recover the rotorcraft safely from system failure situations or major upsets.

3.2 For the purpose of ascribing levels of acceptability to the handling quality parameters of rotorcraft over the flight spectrum outlined above, the flight phases are categorised as in paras 3.2.1 to 3.2.3. This initial categorisation is compatible with that associated with pilot response times presented in Chapter 604 'Automatic Flight Control Systems'.

3.2.1 Active Flight Phase.

(i) Pilot Involvement in Flying - High:
Continuously flying rotorcraft through the flying controls.

(ii) Major Rotorcraft Handling Considerations:
Short term stability and response characteristics;
Manoeuvrability; Precise transient flight path control.

3.2.2 Attentive Flight Phase.

(i) Pilot Involvement in Flying Medium.
Monitoring automatic flight control system;
Gentle manoeuvres to change flight condition;
Short periods of instrument flight.

(ii) Major Rotorcraft Handling Considerations.
Long and short term stability characteristics;
Ease of retrimming to new flight condition;
Response to atmosphere disturbance and system failure modes.

3.2.3 Passive Flight Phase.

(i) Pilot Involvement in Flying - Low.
Monitoring automatic flight control system behaviour;
Occasional retrimming; Protracted flight periods under autopilot control.

(ii) Major Rotorcraft Handling Consideration.
Long term flight path stability; Response to major atmospheric disturbances and system failure modes.

4 OPERATIONAL FLIGHT PHASE CATEGORISATION

4.1 The basic flight phase categories under para 3 are further refined in paras 4.1.1 to 4.1.3, reflecting flight operating regimes.

4.1.1 Active Flight Phases.

(i) Up and Away Manoeuvring: well away from any ground restraints or obstructions in which the manoeuvring may be further classified as either "Moderate" or "Aggressive".

(ii) Low Level, Low Speed Manoeuvring: in proximity to the ground, ship deck, rig platform or other obstruction.

(iii) Low Level, Moderate/High Speed; including more aggressive nap of the earth flight.

(iv) Low Level, Low Visibility: in which compensating for the effects of poor visibility increases pilot workload or rotorcraft system complexity already associated with paras 4.1.1(ii) and (iii) and may call for the provision of pilot vision aids such as night vision goggles, thermal imaging, etc.

4.1.2 Attentive Flight Phases.

(i) Up and Away Flight: involving manoeuvring not exceeding the "Moderate" level of para 4.1.1 (i) .

(ii) Low Level: mainly associated with coupled automatic hover modes and preparedness to intervene in the event of control system or engine failures.

4.1.3 Passive Flight Phases.

(i) Up and Away Flight Conditions associated with protracted periods of "hand-off" operation under automatic control.

LEAFLET 600/3

GENERAL REQUIREMENTS AND DEFINITIONS

FLIGHT ENVELOPES

1 INTRODUCTION

1.1 This Leaflet describes the scope of the constituent envelopes referred to in Chapter 600, para 5 and aerodynamic effects encountered in them.

2 SERVICE FLIGHT ENVELOPE

2.1 LOW SPEED ENVELOPE

2.1.1 The Low Speed Envelope is primarily concerned with the definition of maximum allowable relative air velocity approaching the rotorcraft from any direction over the full 360 degrees azimuth range, for sustained operation in the low speed regime. This is intended to cover:

(i) Stationary hover and vertical manoeuvring, in winds up to the limiting velocity vector conditions.

(ii) Operations in and out of ground effect.

(iii) Translational manoeuvres in any direction up to the limiting relative velocity conditions.

(iv) Take off and landing, and transitions between the hover and the forward flight envelope.

2.1.2 The Rotorcraft Specification will define the critical velocity vectors to be achievable in the low speed regime.

2.1.3 There should be no gap between the forward speed edge of the Low Speed Envelope and the lower limit of the Forward Flight Envelope (para 2.2.1).

2.1.4 In the absence of a specific on-board low airspeed measuring system, an alternative means of determining the relative air speed and direction will have to be arranged, for the demonstration of compliance. This could, for example, make use of doppler ground speed signals (if available) and heading, corrected for reported windspeed and direction, or, formating on a pace vehicle.

2.1.5 The contractor will define the maximum yaw rate limitations on the rotorcraft.

2.2 FORWARD FLIGHT ENVELOPE

2.2.1 The Forward Flight Envelope covers the forward airspeed range from the edge of the Low Speed Envelope up to the maximum normal operating speed. The defined minimum airspeed of the forward flight envelope should be greater than the minimum at which consistent behaviour of the indicated airspeed system can be assured.

2.2.2 The forward flight envelope will also define the altitude, ambient temperature, load factor and sustained bank angle limits up to which the rotorcraft can be continuously operated, including any restrictions due to total weight of the rotorcraft or engine/transmission limitations.

2.3 SIDESLIP ENVELOPE

2.3.1 The Sideslip Envelope defines the aerodynamic sideslip angle limitations over the Forward Flight Envelope.

2.3.2 Insofar as an aerodynamic sideslip indicator may not be included in the rotorcraft standard flight instrumentation fit, a calibrated vane or equivalent device may be required for the purposes of demonstrating compliance.

3 AERODYNAMIC INTERFERENCE EFFECTS

3.1 With reference to Chapter 600, para 8.5, attention should be given to reducing any adverse effects due to:

3.1.1 Variations in rotor wake interactions with the airframe or other aerodynamic surfaces as the rotorcraft translates to or from the hover, and manoeuvres throughout the Low Speed Envelope.

3.1.2 Considerations in para 3.1.1 should also be applied over the full power range in forward flight.

3.1.3 Airframe flow separation effects resulting from incidence or sideslip variations as the rotorcraft changes flight condition within the Service Flight Envelope.

3.1.4 Variations in ground effects on rotor or airframe components as the height and speed of the rotorcraft relative to the local "ground" surface is varied.

3.1.5 Changes in rotorcraft configuration such as orientation of rotor axes or aerodynamic surfaces relative to rotorcraft datum axes, retraction or extension of undercarriage, deployment of tactical system elements, etc., where applicable.

LEAFLET 600/4

GENERAL REQUIREMENTS AND DEFINITIONS

ATMOSPHERIC DISTURBANCES

1 INTRODUCTION

1.1 This Leaflet provides guidance on the representation of atmospheric turbulence and gusts which should be used in the design process, and in analytic or simulation work in support of certification of the rotorcraft.

1.2 The subjective assessment of Atmospheric Turbulence is covered quantitatively in Leaflet 900/3.

2 AVAILABLE MATERIAL

2.1 Reference is made to DEF STAN 00-970, Volume 1, Leaflet 600/5. Pending alternative data this should be used, with caution, where applicable to rotorcraft operations. Because of a number of editorial changes yet to be completed, and also a few areas of lack of clarity, it has not just been uncritically transposed into this Leaflet, at this stage.

2.2 Some extracts are presented below, for initial guidance.

3 TURBULENCE INTENSITIES AND SCALE LENGTHS

3.1 REFERENCE INTENSITIES

3.1.1 A reference intensity is defined and an overall measure of atmospheric disturbance, relating to random turbulence and discrete gust models to be used. It is related to, but not necessarily equal to, the rms values of gust velocity components. Table 1 below associates particular reference intensities with the subjective grades by means of which turbulence is generally qualitatively described:

TABLE 1

Nominal Grade of Turbulence	Reference Intensity (ft/sec)
Light	0 - 3
Moderate	3 - 6
Severe	6 - 12
Extreme	12 - 24

3.2 PROBABILITIES

3.2.1 Fig 1 of the Volume 1 Leaflet referenced above shows the probability of equalling or exceeding given reference intensities at a number of heights above different forms of terrain. This is not reproduced here, pending some clarification of the method which should be used in actually applying this probability data.

3.2.2 The altitudes referred to, which are relevant to rotorcraft operation are: 250 ft, 1,000 ft and 10,000 ft. For rotorcraft operations within this band of heights the atmosphere may be represented in terms of a suitably chosen mean height. Below 250 ft simple generalisations become inadequate for realistic turbulence modelling.

3.3 SCALE LENGTHS

3.3.1 The conventional spatial reference for atmospheric turbulence is the scale length, applied to each of the three mutually perpendicular horizontal and vertical, components. Table 2 below ascribes values to the three component scale lengths, depending on whether height (H) is less than or greater than 2,500 ft.

TABLE 2 - SCALE LENGTHS OF TURBULENCE COMPONENTS

Height above Terrain (H ft)	H ≤2,500 ft	H >2,500 ft
Vertical Component	H	2,500
Fore/Aft Component	184 x (H) $^{1/3}$	2,500
Lateral Component	184 x (H) $^{1/3}$	2,500

3.4 RMS TURBULENCE INTENSITIES

3.4.1 The rms intensity of each turbulence velocity component is related to the reference intensity, ref para 3.1 above, and the corresponding scale length:

RMS Intensity = Reference Intensity x (Scale Length/2500)$^{1/3}$

4 TURBULENCE MODELS

4.1 Major properties of atmospheric turbulence are covered here by two complementary types of model. Investigations of general handling qualities in continuous turbulence should be based on the power-spectrum models. The discrete-gust models should be used for assessments of the response to, and recovery from, the larger forms of disturbance.

4.2 POWER-SPECTRUM MODELS

4.2.1 Volume 1 Leaflet 600/5 para 3.2.1 defines the von Karman representation of turbulence velocity components as functions of the appropriate rms intensity and scale length and a spatial frequency, to be used. While reasonably applied in linear analyses, the difficulties of implementing the von Karman form in practical simulations are recognised, and the Dryden form of approximation is allowed, providing the achieved spectrum fits the von Karman form over the range of frequencies of interest.

4.2.2 A well documented shortcoming of these classical power spectrum models applied to aircraft response to atmospheric turbulence studies, is their inability to provide a sufficiently high probability of occurrence of relatively large disturbances. Turbulence models have been devised which exhibit this "intermittency" feature together with the appropriate power spectrum characteristics. Where computationally feasible, this type of model should be used for rotorcraft response to turbulence studies and simulation.

4.2.3 The basic power spectrum models described above are functions of a spatial frequency. For application in time domain studies the traditional approach is to transform them into functions of temporal frequency by the frozen-field concept. This assumes a distance traversed in a given time interval to be inversely proportional to the mean true airspeed. This presents no problem for forward flight conditions, but for rotorcraft in near-hovering conditions some consideration is needed on the appropriate definition of velocity to be used. One approach would be to associate an appropriate mean wind speed, (ref para 5.1), with the turbulence intensity and use the resulting relative airspeed of the rotorcraft for the transformation referred to above.

4.3 DISCRETE GUST MODELS

4.3.1 Volume 1 Leaflet 600/5 defines families of discrete gusts of equal probability in which the energy considerations are compatible with the von Karman form of power spectrum. For a given atmospheric condition expressed in terms of reference intensity and scale length the family of possible gusts, derived by varying the gradient distance parameter, should be explored to determine which results in the most critical rotorcraft response, in the particular context involved. Fig 1 illustrates the gust profiles and parameters involved. The same transformation to the time domain as described in para 4.2.3 above is also involved.

4.3.2 Single Ramp Gust (Fig 1a). This is the fundamental discrete gust model, applicable to all 3 components of turbulence with the family of gust amplitudes for each component given by the equation shown.

4.3.3 Double Ramp Gust (Fig 1b). This double ramp form should be used to seek out any resonant build up in particular rotorcraft response parameters of interest. Each single constituent ramp is a member of the same family, but a factor of 0.85 may be applied to each magnitude to allow for the reduced probability of actually encountering this particular form of gust profile. The individual gradient distances and the separation distance should be varied in search of the most critical response.

4.3.4 Linear Approximation to Ramp Gust (Fig 1c). The straight line approximation to the ramp profile is an acceptable alternative where rigid body modes only are being considered. Where the rotorcraft models used for gust studies include more refined representations of rotor dynamics, the linear ramp approximation should only be used when the contractor has established that this simplification does not overlook any significant aspects of the gust responses under critical conditions. The approach is equally applicable to single and double ramp gust profiles.

5 STEADY WIND EFFECTS

5.1 MEAN WIND AND TURBULENCE

5.1.1 It is commonly assumed that at a height of 33 ft above typical airfield environments the reference intensity is approximately ¼ of the mean wind speed. This relationship may need revision in the context of nap of the earth rotorcraft operations.

5.2 WIND SHEAR

5.2.1 In the context of typical airfield environments it is commonly assumed that between heights of 10 ft and 300 ft the wind speed, u, at height Z ft is given by:

$$u = (0.43 \log_{10} (Z) + 0.345) W$$

where W is the wind speed at a height of 33 ft.

5.2.2 This relationship may need revision in the context of typical low level operational environments for rotorcraft.

6 COMPLIANCE

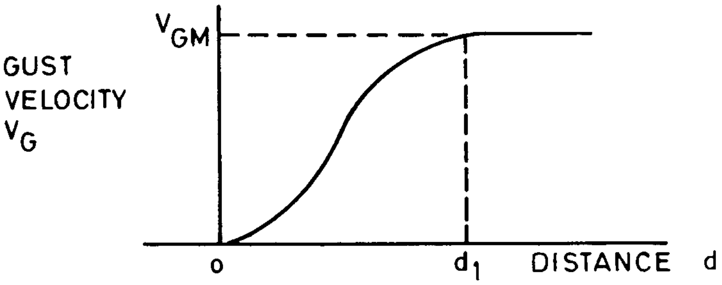
6.1 There will always be difficulty in demonstrating formal compliance with requirements involving specific turbulence levels due to:

(i) Practical problems of quantifying the actual level of turbulence experienced during any particular flight test.

(ii) Likely non-availability of the required levels of turbulence in the atmosphere at the appropriate points in the rotorcraft certification programme.

6.2 Unless otherwise stated, quantitative requirements of Part 6 assume relatively mild levels of atmospheric turbulence and ideally the rotorcraft should be assessed against those requirements in conditions of no more than light turbulence. Levels of acceptability of the handling qualities in turbulent conditions up to the Specification levels should then be subjectively assessed in flight, as and when suitable conditions prevail. This should be supported by analytic and/or simulation studies of rotorcraft response to atmospheric disturbances, modelled using the guidance in this Leaflet.

(a) SINGLE RAMP GUST



FOR $0 \leq d \leq d_1$ $\quad V_G = \frac{V_{GM}}{2} \left(1 - \cos \frac{\pi d}{d_1} \right)$

FAMILY OF EQUIPROBABLE GUSTS FOR VARIOUS d_1 GIVEN BY :–

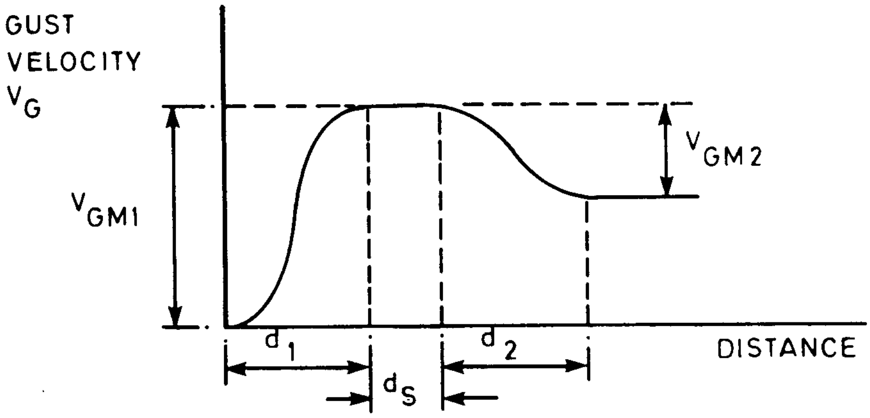
$$V_{GM} = 5 \bar{\sigma} \left(\frac{d_1}{2500} \right)^{1/3} \quad d_1 \leq L_v$$

$$V_{GM} = 5 \bar{\sigma} \left(\frac{L_v}{2500} \right)^{1/3} \quad d_1 \geq L_v$$

IN WHICH:- $\bar{\sigma}$ REFERENCE INTENSITY

L_v (ft) TURBULENCE COMPONENT SCALE LENGTH

(b) DOUBLE RAMP GUST



(c) APPROXIMATE RAMP

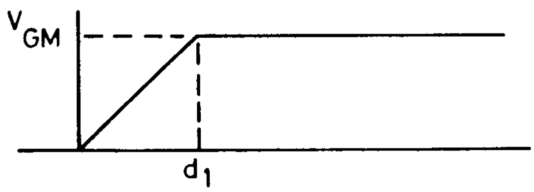


FIG.1 DISCRETE GUST MODELS

LEAFLET 600/5

GENERAL REQUIREMENTS AND DEFINITIONS

LEVELS OF HANDLING QUALITIES

1 INTRODUCTION

1.1 This Leaflet summarises the major factors affecting the overall handling qualities of the rotorcraft, as perceived by the pilot. A number of these topics are the subject of specific requirements in Part 6, while others are dealt with in other parts of DEF STAN 00-970 Volume 2 as design or installation issues, but are included here as a reminder of their impact on the pilots' rating of the rotorcraft handling qualities.

2 OVERALL ROTORCRAFT BEHAVIOUR

2.1 FLIGHT CHARACTERISTICS

2.1.1 The balance between the responsiveness of the rotorcraft to deliberate control applications and its stability or insensitiveness to external atmospheric disturbances or other unscheduled upsets.

2.1.2 The progressiveness of any changes in stability and control response characteristics over the Permissible Flight Envelope, and the ease with which the pilot is able to recognise the approach to design limitations or dangerous flight conditions.

2.2 GROUND HANDLING

2.2.1 The ease with which the rotorcraft can be operated from start up to shut down, including the execution of pre-flight checks, taxied and manoeuvred on the ground by the aircrew with the minimum of dependence on external ground personnel or services, other than essential guidance facilities.

3 CONTROL CHARACTERISTICS

3.1 ACCESSIBILITY

3.1.1 The ease with which all primary flight controls, engine controls, rotorcraft system controls, selectors switches and buttons can be comfortably reached and operated by the pilot when in his seat in its normally adjusted position and with harness locked.

3.2 OPERATION

3.2.1 Harmonisation of flying controls, with smoothness and logicality of control forces allowing precise control of the rotorcraft to be easily maintained.

3.3 FAILURE TOLERANCE

3.3.1 Level of rotorcraft system failure survivability compatible with mission/roles required by the Rotorcraft Specification.

4 INSTRUMENTATION AND FIELD OF VIEW

4.1 INSTRUMENTATION

4.1.1 Presentation of all basic flight condition data describing rotorcraft speed, attitudes and heading, together with rotor speed and critical system monitoring information, in an ambiguous form that is readily scanned under operating conditions ranging from bright sunlight to darkness, by either pilot where relevant.

4.1.2 All system indicators, advisory and warning devices to be readily observable by the aircrew under all operating conditions.

4.2 FIELD OF VIEW (See Chapter 104)

4.2.1 Adequate fields of external view are available to the pilot, and co-pilot where applicable, in his normal seated position, as defined by the Rotorcraft Specification, especially for critical approach and landing phases of flight, steeply banked manoeuvres when appropriate to the usage envisaged for the rotorcraft, and flight operations close to the ground or external obstructions.

4.2.2 No objectionable reflections in the cockpit glazing under all conditions of internal and external lighting.

4.2.3 Means of preventing outside visual reference being unacceptably obscured as a result of atmospheric precipitation, low temperature, or deposits on the glazing.

4.2.4 Provision of weapon sights or vision enhancement systems to facilitate execution of roles required by the Rotorcraft Specification under adverse operating conditions.

5 COCKPIT ENVIRONMENT

5.1 Levels of vibration, noise, seat comfort, temperature or ingress of undesirable fluids or fumes to the cockpit not to cause undue pilot fatigue, distraction or reduction in mission effectiveness during sorties of extended duration.

6 WORKLOAD

6.1 Division of Pilot's Attention between:

6.1.1 Tasks specifically related to flying the rotorcraft:

(i) Interpretation of rotorcraft flight condition status from cockpit instrumentation and motion, visual or aural cues.

(ii) Decision making processes.

(iii) Timely execution of appropriate control strategies.

and

6.1.2 Secondary tasks such as navigation, communication, system status monitoring, general airmanship and tactical mission-oriented tasks.

6.2 Dependence on:

6.2.1 Role or mission flight phase being undertaken.

6.2.2 Normal or Failure Operating state of the rotorcraft.

6.2.3 Prevailing ambient environmental conditions.

LEAFLET 600/6

GENERAL REQUIREMENTS AND DEFINITIONS

CONTROL CHARACTERISTICS

1 INTRODUCTION

1.1 This Leaflet provides explanatory and quantitative support to the requirements contained in Chapter 600, para 9.

1.2 Those requirements for the primary flying controls are presented in a general form which can be applied irrespective of the form of the controllers used.

2 CONTROL CONFIGURATIONS

2.1 Primary flying controls fall into two broad categories as outlined in paras 2.2 and 2.3, but there is still a significant degree of commonality in the requirements for handling qualities.

2.2 DISPLACEMENT SYSTEMS

2.2.1 Displacement systems operate the control surfaces in response to movement of the flying controls by the pilot, or an appropriate automatic flight control system. This embraces the conventional rotorcraft configuration consisting of a centrally located cyclic stick, collective lever and rudder pedals. In this there is generally a fairly direct mechanical relationship between flying control position and rotor blade angle applied. However, variations within this category are possible, separately or in combination:

(i) Controllers can be relocated, ranges of travel reduced, and several control functions combined in one controller, (e.g., side arm controllers).

(ii) The direct mechanical link between cockpit and control surfaces can be replaced to varying degrees of completeness, (i.e. fly-by-alternative media).

2.3 FORCE SENSITIVE SYSTEMS

2.3.1 Force sensitive systems cause the control surfaces to be fundamentally operated in response to forces applied by the pilot. In practice the generation of that control force may be associated with some small deflection of the controller, but the force, rather than the deflection, is the relevant parameter.

2.3.2 The processing and transmission of the pilot's control force signal to the operating control surfaces invariably involves the alternative media referred to in para 2.2.1(ii). The reduced relevance of a manual reversion mode with this control philosophy also makes the approach compatible with the more compact controller configurations referred to in para 2.2.1(i), although this is not a necessity.

2.4 CONTROL INPUTS

2.4.1 Unless it is stated otherwise, or is obvious in the immediate context, the term 'control input' used in Part 6 should be interpreted as meaning either a displacement or a force, whichever is to the particular rotorcraft control system.

2.5 COMPATIBILITY

2.5.1 In a particular rotorcraft, it would generally be considered bad practice to mix the displacement and force sensitive control philosophies, certainly within individual rotor systems.

3 BREAKOUT

3.1 Breakout force is here defined as the static force which has to be overcome by the pilot in order to initiate a control input. Unless stated otherwise, throughout Part 6 'breakout force' refers to the force applied at the normal point of contact of the pilot with the particular flying control.

3.2 ORIGINS OF BREAKOUT FORCES

3.2.1 In displacement control systems breakout forces basically stem from control run bearing friction, servo seal friction etc., and detents associated with self-centring, where this is applicable.

3.2.2 Inadequate balancing of a flying control run is likely to lead to increased, and/or asymmetric breakout force levels necessary to retain the trimmed control position. The basic control run design should incorporate adequate static and dynamic mass balancing. This may be supplemented by the careful application of spring compensation, provided that this is based, for example, on low rate springs and geometrical arrangements which ensure that the additional spring forces involved:

(i) are insensitive to flying control position;

(ii) do not have any unacceptable effects on the control feel characteristics about the datum position selected by the pilot.

3.2.3 The factors referred to in para 3.2.1 do not arise in the case of the force-sensitive type of control system. Nevertheless, for the reason in para 3.3.3 (iii) a force deadspace, effectively a breakout characteristic, may be necessary.

3.3 RELATIONSHIP TO CONTROL UTILISATION

3.3.1 Acceptable breakout force characteristics are a compromise between:

(i) Making them as small as possible, to minimise interference with the smooth execution of precise control inputs around the datum point.

(ii) The need for finite values to accommodate specific aspects of rotorcraft system operation, (para 3.3.3).

3.3.2 The level of breakout force which is acceptable is related to the general levels of forces and gradients involved in operation of any particular type of system. There should not be any marked discontinuity in control force following breaking out of the control system. In particular the net control force associated with 5% full travel deflection, from the trimmed datum position should not be less than the breakout force.

3.3.3 Operational aspects which depend on the presence of definite breakout force characteristics include:

(i) Positive self-centring of certain displacement type controls to specific datum positions.

(ii) Provision of an adequate grounding point for those auto-stabilisation systems which function through series actuation devices incorporated within displacement type control runs between the cockpit controls and the control surfaces or associated powered actuators. In order for the extension of such series actuator links to move the control surface by the appropriate amount, rather than back-off the cockpit control, the net breakout force on the cockpit side of the series actuator must be greater than on the control surface side. Clearly, the excess flying control breakout can be reduced by locating the series actuation subsystem as far downstream, towards the control surface as is practicable.

(iii) The presence of a finite breakout characteristic also has some beneficial effect in reducing the likelihood of inadvertent control inputs, with the pilot lightly holding the controls in a relaxed manner.

3.4 BREAKOUT FORCE CHARACTERISTICS

3.4.1 Normal operating state. Table 1 below indicates ranges of acceptable breakout force in each primary flight control channel for various levels of handling qualities.

(i) Handling qualities should be interpreted as being at LEVEL 2 if the breakout force is less than the minimum shown for LEVEL 1 but not less than that shown for LEVEL 2, or between the maximum values shown for LEVELS 1 and 2. Similarly LEVEL 3 applies within the limits given for LEVEL 3 but outside of those given for LEVEL 2.

(ii) Minimum values of breakout force are generally set to avoid poor self centring and datum position holding in displacement type systems, and to reduce the likelihood of inadvertent control inputs in any system. However, attention is again drawn to the need to satisfy para 3.3.3 (ii) when this is applicable.

(iii) Maximum values of breakout force are set to avoid impairing the application of small, precise control inputs and corrective reversals.

(iv) Table 1(a) relates to the conventional displacement type of system (para 2.2). Table 1(b), etc., are intended to cater for alternative control configurations, as and when suitable data can be specified.

(v) If a system includes adjustable friction devices, the minimum limits should be achieved with that friction at its minimum setting.

TABLE 1

BREAKOUT FORCE LIMITS (lb) - NORMAL OPERATING STATE

(a) Conventional Control Configuration			
CHANNEL	LEVEL 1	LEVEL 2	LEVEL 3
Longitudinal	0.5 - 1.5	0.5 - 2.0	0 - 3.0
Lateral	0.5 - 1.5	0.5 - 2.0	0 - 3.0
Directional	3.0 - 5.0	2.0 - 7.0	1.0 - 9.0
Collective	1.0 - 3.0	0.5 - 4.0	0 - 5.0

(b) Alternative Configuration			
CHANNEL	LEVEL 1	LEVEL 2	LEVEL 3
Longitudinal Lateral Directional Lifting	(TBA)	(TBA)	(TBA)

3.4.2 Failure States

(i) Following a servo failure, and possible introduction of a bypass mode there is likely to be an increase in the effective seal friction etc., degrading the breakout force characteristics. The proposed maximum values given in Table 2 (a) show an increase for LEVELS 2 and 3, compared with the normal operating state, but LEVEL 1 is unchanged since by definition this should be a function of mission worthiness rather than rotorcraft operating state.

TABLE 2

MAXIMUM BREAKOUT FORCE (1b) - FOLLOWING FIRST FAILURE

(a) Conventional Control Configuration			
CHANNEL	LEVEL 1	LEVEL 2	LEVEL 3
Longitudinal	1.5	2.2	3.5
Lateral	1.5	2.2	3.5
Directional	5.0	8.0	10.0
Collective	3.0	5.0	7.0

(ii) Following a jam in a (displacement type) flying control system the maximum control force to first overcome the jam is given in Table 3. For subsequent continuous operation of the system maximum breakout forces are given by Table 4.

TABLE 3

MAXIMUM CONTROL FORCE (1b) TO INITIALLY FREE A FLYING CONTROL SYSTEM JAM

(a) Conventional Control Configuration			
CHANNEL	LEVEL 1	LEVEL 2	LEVEL 3
Longitudinal	5	30	50
Lateral	3	12	20
Directional	10	40	70
Collective	7	25	40

TABLE 4

MAXIMUM BREAKOUT FORCE (1b) FOLLOWING FREEING OF CONTROL SYSTEM JAM

(a) Conventional Control Configuration			
CHANNEL	LEVEL 1	LEVEL 2	LEVEL 3
Longitudinal	1.5	5	10
Lateral	1.5	5	10
Directional	5.0	15	30
Collective	3.0	5.0	20

4 TRIM AND SPRING FEEL

4.1 In the context of conventional flying control configurations there is a universal requirement to introduce spring feel for displacement of the longitudinal and lateral controls from their trimmed datum position. Proposed ranges are included in Table 5 below. The lateral gradient should be in the range 40% - 60% of the gradient selected for the longitudinal channel.

4.2 Opinion is more divided on the merits of spring feel applied to the directional and collective channels. Consequently no values are put forward for these channels in Table 5, at this time. On the other hand, for certain mission flight phases there may be merit in having a facility for trimming these controls to particular datum positions, with transient manoeuvring accommodated by a spring device performing more of a return-to-datum function than having any very specific feel characteristic.

TABLE 5

SPRING FEEL GRADIENTS (lb/in)

CHANNEL	MIN	MAX
Longitudinal	0.5	1.5
Lateral	0.25	1.0
Directional	-	-
Collective	-	-

LEAFLET 600/7

GENERAL REQUIREMENTS AND DEFINITIONS

ROTORCRAFT TRANSIENT RESPONSE CHARACTERISTICS

1 INTRODUCTION

1.1 This Leaflet introduces criteria describing the net transient response characteristics, and indicates how the initial control responsiveness and short term dynamic stability requirements of Chapter 600, paras 10.1 and 11.3 might be quantified for the purposes of demonstrating compliance. Specific quantitative data for guidance will be given in the appropriate Leaflets associated with Chapters 601 to 605.

2 BASIC APPROACH

2.1 To achieve repeatability and allow comparison with predictions the rotorcraft characteristics are described with reference to the time history of the response of relevant parameters to finite length pulse control inputs, as shown in generalised form in Fig. 1. These inputs are to be applied in steady trimmed flight in steady atmospheric conditions.

2.2 RESPONSE TYPE

2.2.1 The flight parameters which are relevant will depend on the particular Flight Phase involved, and the rotorcraft axis under consideration.

2.2.2 For Active flight phases involving Aggressive manoeuvring, angular rate responses about each axis should be closely related to inputs in the appropriate primary flight control channel. That is, a sensibly small control input, held constant should result in a smooth, rapid build up to an angular rate, predictably related to the magnitude of the input. When the control is returned to datum, the angular rate response should also smoothly and rapidly return to zero.

2.2.3 In other Active flight phases, more concerned with moderate manoeuvring, precise flight path control, flying close to the ground or other obstacles, and also in the majority of Attentive phases, the additional stability associated with an attitude type of response being closely related to the control input makes this latter preferable to the rate type outlined in para 2.2.2.

2.2.4 Some rotorcraft roles may involve Active phases demanding very precise control over positioning of the rotorcraft relative to external references or obstacles. For these purposes an attitude type of response characteristic may not be sufficient, and translational velocity, or even position, may have to be considered as the relevant parameter (see also Leaflet 600/8).

2.2.5 Under adverse operating conditions the external visual reference cues inherent in the tasks referred to in para 2.2.4, are likely to be severely degraded. Compensation for this would lead to a shift in emphasis towards a translational velocity or positional type of response characteristic.

2.2.6 Modification of the inherent response characteristics of the basic rotorcraft, to achieve the various response types outlined above, will require stability and control augmentation systems involving the feedback of various response parameters. The extent to which the individual optimisation for each of the various phases of flight operation is pursued, in covering the multi-role potential of a particular rotorcraft, will have an impact on system complexity, reliability and overall cost-effectiveness of the rotorcraft. These aspects should be carefully considered in the formulation of the Rotorcraft Specification, and its interpretation by the Contractor, to avoid any unnecessary over-design.

2.3 PULSE INPUTS

2.3.1 The input consists of a sharp control application, held constant for a predetermined time interval, with the control then returned to, and held at its original setting for as long as necessary for the particular assessment.

2.3.2 A pulse input is chosen, so that the final datum flight condition to which the rotorcraft should ideally return is fully identified, being the same as the initial condition prior to the input. (This would not be the case for sustained step inputs).

2.4 CONTROL APPLICATION

2.4.1 In some cases it may be appropriate to vary the magnitude and duration of the control input. Unless otherwise stated the pulse is assumed to consist of a 10% full travel or one inch displacement, whichever is the least, in the case of conventional displacement type flying controls; or 10% of the full force range in the case of alternative force-sensitive controls, held for one second in each case.

2.4.2 For control responsiveness assessments the input should be applied through the appropriate flying control. However for stability assessments, particularly those with an automatic flight control system involvement, it is generally expedient to generate these test inputs (electronically), through the system.

2.5 PEAK RESPONSES

2.5.1 The peak value of the parameter response to the control input is a measure of control responsiveness and also a basis for quantifying the stability of the rotorcraft in its return to datum.

2.5.2 In this context it is proposed to adopt the maximum value reached by the relevant parameters, or the value reached at the end of the pulse, whichever occurs later, as the nominal peak response.

2.5.3 When an angular displacement of the rotorcraft is the relevant response parameter its first peak will be reached after the end of the pulse input, and so can be adopted with no question, as indicated by the full line in Fig.1.

2.5.4 When an angular rate is the relevant rotorcraft response parameter, it may reach its first maximum value before the end of the pulse, as shown dotted in Fig.1. In this case it is proposed to adopt the value coinciding with the end of the pulse on the grounds that this is a readily identifiable point meaningful to both control sensitivity and as a starting point for the return of the parameter to datum.

3 INITIAL RESPONSE TO CONTROL INPUT

3.1 RESPONSE CRITERIA

3.1.1 For the control input described in paras 2.3 and 2.4 the response of the relevant flight parameter can be characterised in terms of responsiveness, initial delay and sensitivity, quantified by peak values and intermediate values shown in Fig. 1.

3.2 CONTROL RESPONSIVENESS

3.2.1. The responsiveness of the rotorcraft to control inputs is indicated by the peak value, (para 2.5).

LEVEL 1:	Peak value must be greater than some minimum for adequate responsiveness, and not exceed some maximum to avoid over sensitivity. These limits will depend on whether Aggressive or Moderate manoeuvring requirements have to be met.
LEVELS 2 & 3:	Peak values within maximum and minimum limits wider than for LEVEL 1 would be accepted.

3.3 INITIAL DELAY

3.3.1 The cumulative effects of control system lags and delays, and vehicle response characteristics should be such that within a specified finite time of the initiation of the control input the relevant parameter must have achieved a minimum percentage of the peak value.

3.3.2 The time referred to in para 3.3.1, (T_I on Fig. 1), remains to be optimised in terms of practicality of timing and resolution between acceptable and unacceptable response characteristics. It is anticipated that 0.5 sec may be suitable.

3.3.3 With reference to Fig. 1, for LEVEL 1 handling qualities, at T_1 (TBA) sec y_1 should not be less than 30%.

3.3.4 For LEVELS 2 and 3 the minimum value of y_1 at T_1 could be lower than for Level 1 handling qualities. It is not proposed to distinguish between Aggressive and Moderate manoeuvring for this particular criterion.

3.4 SENSITIVITY

3.4.1 To avoid oversensitivity the parameter response y_1% at time T_1 (see Fig. 1) should not exceed some percentage (TBA) of the peak value.

3.4.2 With reference to Fig. 1, prior to the end of the control input, the response parameter should not exceed the peak value by more than y_2% (TBA).

3.4.3 There should be no obtrusive hesitation in the rate of build up of the response parameter to its first maximum value.

3.4.4 Further criteria, probably based on bandwidth concepts remain to be established to give guidance on avoiding pilot induced oscillations, especially in the case of high control power, highly augmented rotorcraft and in high gain piloting tasks.

4 CONTROL HARMONISATION

4.1 The responsiveness and sensitivity of the rotorcraft response to control units in each axis, and also the associated control force characteristics, need to be well matched to each other so that:

4.1.1 An intentional input in one channel does not result in inadvertent inputs in other channels, while at the same time not inhibiting the pilot from making instinctive control adjustments in other channels to contain any cross coupled rotorcraft responses to the primary input.

4.1.2 Manoeuvres which require multi-axis control inputs can be executed in a smoothly, co-ordinated fashion without undue skill or effort on the part of the pilot.

4.2 These considerations warrant particular attention in two main areas:

4.2.1 Where multi axis control functions are combined in a single controller. The more axis involved, (e.g. , some side arm controller configurations), the more care needs to be applied.

4.2.2 Control mixing is often introduced into rotorcraft to minimise the cross coupling of responses to a particular intentional control input. However, the degree of mixing required to satisfy quasi-steady changes of flight condition, and rapid transient changes, generally differ. Therefore, care is needed to reach the best design compromise without undue system complication.

5 LOST MOTION

5.1 PRIMARY FLYING CONTROLS

5.1.1 In the context of the primary flying controls overall lost motion refers to an input at the controller which does not result in corresponding movement of the associated rotor blade or control surface. Under all operating conditions, for LEVEL 1 qualities this input should not exceed ± 1% of the full range of control displacement or force. Acceptable magnitudes of lost motion for the lower levels of handling qualities remain to be determined.

5.2 AUTOMATIC FLIGHT CONTROL SYSTEM INTERFACE

5.2.1 Lost motion at pick-offs, sensors, actuators, etc., comprising closed loop control systems can potentially introduce a variety of amplitude-dependent non linearities which degrade the phase and gain relationships governing the behaviour of the system, and hence the rotorcraft, in the presence of small amplitude disturbances (e.g., light atmospheric turbulence).

5.2.2 The contractor should ensure that the physical characteristics of the components installed in the control loop do not result in objectionable behaviour of the rotorcraft.

6 DYNAMIC STABILITY CHARACTERISTICS

6.1 RESPONSE MODES

6.1.1 The response of a particular flight parameter to any disturbance is the summation of a number of periodic and aperiodic modes embracing a range of frequency and damping characteristics. The number of such modes is invariably increased with the incorporation of sophisticated stability and control augmentation systems. This mixing of often well damped periodic and aperiodic modes makes it difficult to observe individual modes and meaningfully apply classical frequency and damping factor criteria in stability analyses. Consequently the approach described in para 6.2 has been adopted.

6.2 SHORT-TERM STABILITY CRITERIA

6.2.1 For LEVELS 1 and 2 the criteria are directly related to the time history of the recovery of the relevant flight parameter from a disturbed condition. For the purposes of demonstrating compliance with requirements, that disturbed condition is conveniently taken as the peak response to a pulse input, as defined in para 2.5, with criteria time-referenced from the point at which that peak occurs.

6.2.2 With reference to Fig. 1, unless otherwise stated in the requirements or associated Leaflets, recommended criteria for LEVEL 1 qualities are:

(i) Maximum time (T_{30}) to return within 30% peak disturbance from datum.

(ii) Minimum and maximum limits on the time (T_{01}) to first pass through datum.

(iii) During the first return to datum there should be no obtrusive hesitation in the rate of return.

(iv) Maximum percentage of peak distribution for first peak overshoot (x_1).

(v) Minimum time (T_{02}) for any second pass through datum (in same sense as initial disturbance) from any overshoot x_1, greater than 5%.

(vi) Maximum percentage of peak disturbance for any second peak (x_2) in the same sense as the initial disturbance.

(vii) Maximum time (T_F) to return and remain within ± X_F% of peak disturbance about datum.

6.2.3 Unless otherwise stated in the requirements or their associated Leaflets, and pending alternative data, the following values for the criteria of para 6.2.2 are recommended for guidance:

T_{30}	1 sec
T_{01}	greater than 1 sec, less than 2 sec
x_1	15%
T_{02}	more than 2 sec
x_2	10%
T_F	3 sec
x_F	10%

6.2.4 For LEVEL 2 qualities criteria similar to those in para 6.2.2 above are relevant, with some quantitative relaxation in their application. In addition, the LEVEL 1 requirement for a definite initial overshoot, ideally very small, which is implicit in 6.2.2 (ii), can be waived in favour of limiting the time (T_{11}) for the response to first fall below 10% of the peak value. Unless otherwise stated in the requirements of their associated Leaflets and pending alternative data, the following values are recommended for guidance.

T_{30}	1 sec
T_{11}	greater than 1 sec, less than 2 sec
x_1	20%
T_{02}	more than 2 sec
x_2	15%
T_F	5 sec
x_F	10%

6.2.5 For LEVEL 3 qualities any dominant short period oscillatory modes should be damped. Unless otherwise stated in the requirements or their associated Leaflets, and pending alternative data, the following criteria are recommended for guidance.

(i) Where flight under IFR is required, oscillations having a period of 5 seconds or less should halve amplitude in less than one cycle, and those with a periods greater than 5 seconds in less than 2 cycles.

(ii) For flight under VFR, oscillations having a period of 5 seconds or less should halve amplitude in less than 2 cycles, and those with periods between 5 and 15 seconds should not be divergent.

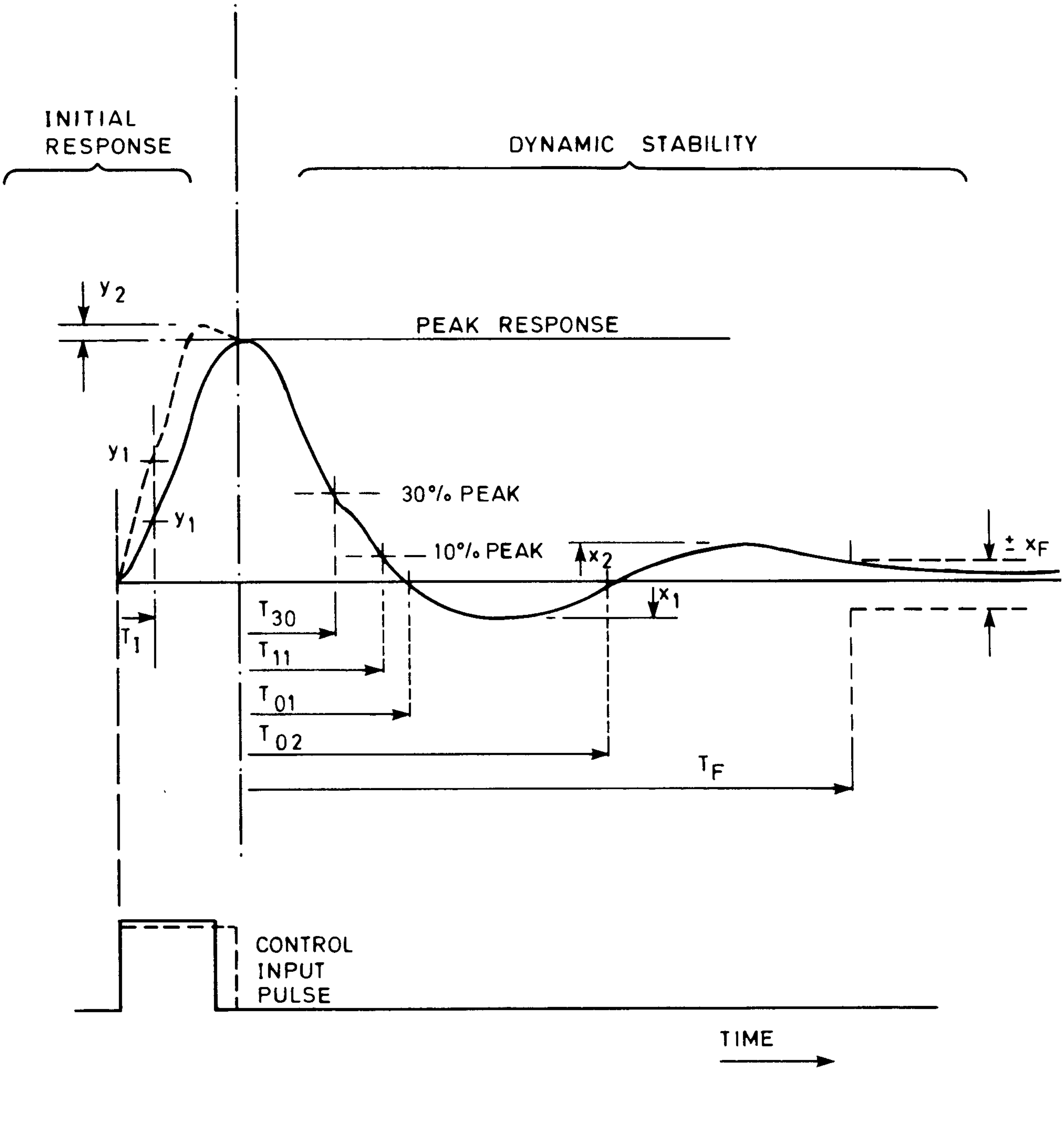


FIG.1 TRANSIENT RESPONSE CHARACTERISTICS

LEAFLET 600/8

GENERAL REQUIREMENTS AND DEFINITIONS

LONG TIME CONSTANT RESPONSE MODES

1 INTRODUCTION

1.1 This Leaflet relates to the longer term stability characteristics introduced in Chapter 600, paras 11.4 and 11.5. An example of the relevance of para 11.4.1 is its application to the task of refined retrimming of a conventional rotorcraft to an intended airspeed in forward flight. The parameters involved in the low speed positioning task addressed in Chapter 600, para 11.5 are also basically governed by relatively long aperiodic time constants. In this case however the handling qualities objective is to significantly shorten the net effective response time so that the pilot can comfortably anticipate the appropriate control inputs required to maintain precise control of the rotorcraft.

2 TRANSLATIONAL MOTION

2.1 CONVENTIONAL ROTORCRAFT CONFIGURATIONS

2.1.1 Translational velocity changes in any flight regime are fundamentally brought about by re-orientation of the main rotor thrust vector. This is inevitably closely related to rotorcraft attitude changes. While attitude dynamics can generally be made well damped and short period in character, speed changes are dominated by relatively long time constants. Long response time constants make it difficult to judge at any time following a control input:

(i) Whether the control input was in fact appropriate to achieve the intended change in condition.

(ii) The proximity to that final condition.

2.1.2 If the time constant is excessively long such doubts can prompt corrective action, which can then degenerate into a repetitive, over-correcting situation.

2.2 ALTERNATIVE ROTORCRAFT CONFIGURATIONS

2.2.1 Alternative configurations may incorporate elements providing controllable propulsive forces directly along the axes of translational motion. Assuming these elements themselves are intrinsically sufficiently responsive, this would serve to reduce the basic long time constant referred to in para 2.1.

3 RESPONSE TO A SINGLE INPUT

3.1 INPUT

3.1.1 In this present context "input" refers to a single pilot action. This may result in step change in a particular control or trim inceptor, but is not meant to preclude the effects of any relatively short time scale scheduling of variation in the appropriate control surface setting or trim variable, initiated by that single pilot input.

3.2 TIME HISTORY CONSIDERATIONS

3.2.1 Fig. 1 depicts the long term response to a single pilot input, (para 3.1.1), of a relevant parameter governed by a long time constant, once the other initial transients have decayed. If left undisturbed this would continue to build up asymptotically towards its final value. The protractedness of the exponential response to a step input is highlighted by the following table of times, expressed as multiples of the time constant, taken to reach the given percentages of the final value:

No. of Time Constants:	1	2	3	4
% Final Value Reached:	63. 2	86.5	95. 0	98. 2

As another example, having reached a particular value after an elapsed time of two time constants, the response will only increase by a further 10% over the next interval of one time constant.

3.2.2 In the real world, leaving the response alone to reach its final value lacks practicality in determining the effective time constant of such a response parameter, when this may be of the order of several tens of seconds.

3.2.3 Alternatively, consider the increments in the response over two successive equal time intervals. It can be shown that the ratio of these increments is related to the ratio between the time interval and the response time constant, as shown in Fig. 2.

3.3 ASSESSMENT OF APERIODIC RESPONSES

3.3.1 Having decided upon the maximum time constant which would be acceptable on the basis of say the time (3 time constants) within which the response shall have achieved 95% of its final value, a suitable time interval can be chosen, and the ratio of response increments over successive intervals determined from the time history. This ratio should lie above the curve shown in Fig. 2.

3.3.2 It should be noted that this process for assessing the effective time constant of an aperiodic response to a single input does not rely on knowledge of either:

(i) The final value of the response (difficult to establish without extremely lengthy recording in ideal conditions because of the slow convergence basic to the problem), or

(ii) The precise point of initiation of the exponential rise, (generally obscured by other transient modes with varying rates of decay).

4 TRANSLATIONAL AND POSITIONAL CONTROL

4.1 THE BASIC TASK

4.1.1 The fundamental flight mechanics issues were introduced in para 2.1. To meet the requirements of Chapter 600, para 11.5.1, the net response timescale has to be compressed.

4.1.2 The basic manual flying technique achieves this by initially applying an excessive input, to give impetus to the response, followed by a reduction to the final setting required by the new flight condition. The extent and phasing of this overcontrolling demands judgement on the part of the pilot, while the necessary anticipation itself is dependent on the availability of adequate cues, external references etc. Which cues are of paramount importance will depend on the particular flight phase, and the local environmental conditions under which it is being undertaken.

4.1.3 The rotorcraft control system can only fundamentally orientate the rotor thrust vector relative to the rotorcraft body axes. In order to achieve appropriate orientation in space, through the application of well ordered control inputs, the rotorcraft axes themselves must also be suitably stabilised in space.

4.2 WORKLOAD REDUCTION

4.2.1 The pilot workload described in para 4.1 can be alleviated by appropriate feedback control augmentation system integration, thereby simplifying the pilot's control input required to achieve a particular flight condition to a relatively simple form. To meet stringent requirements for precise manoeuvring under adverse operating conditions at LEVEL 1, automatic control augmentation will generally be needed.

4.2.2 Further, to achieve adequate translational acceleration without excessive rotorcraft attitude variations, auxiliary propulsive elements referred to in para 2.2 may also be necessary.

4.3 DISCRETE CONTROL INPUT

4.3.1 In the present context "discrete" refers to a relatively simple form of control input (para 4.2.1) which may contain an element of overshoot and correction, (as illustrated in Fig. 4), but which is still completed in time (T_0 in Fig.4) which is short compared with the effective rise time (T_2 in Fig. 4) of the associated response.

4.3.2 In an ideal case, the single step input can be thought of as the perfect, limiting, discrete input.

4.4 FRAMEWORK OF REQUIREMENTS

4.4.1 Chapter 600, paras 11.5.2 and 11.5.3 lay down a qualitative framework of requirements in an area where definitive quantitative data remains to be established.

4.4.2 The concept of a 'discrete' control input (para 4.3) has been introduced to allow greater flexibility in formulating meaningful requirements for changes in translational velocity or position, than would be possible with simple single step inputs. It remains to establish the practicality of this approach.

4.5 TRANSLATIONAL TRIM CHANGES

4.5.1 Chapter 600 para 11.5.2 addresses the parameter change per unit control input. In this context the "final" increment from any steady initial flight condition is dependent on the "final" magnitude of the control input, irrespective of any over-control during the initial interval (T_0) in Fig. 4. (This later only affects the time taken to approach the final response level).

4.5.2 The major consideration in para 11.5.2 centres on the predictability of this final response, and an idealised requirement that the response per unit input should not vary significantly at any initial flight condition within a useful working envelope of relative wind vectors, within the Low Speed Envelope. Over the remainder of the Low Speed Envelope any gain variations should remain orderly.

4.5.3 The extent of the working envelope referred to in para 4.5.2 will depend on the missions/roles called for in the Rotorcraft Specification. The wider this envelope, the more sophisticated the control system augmentation will have to be, in order to counter the non-linearities of changing aerodynamic flow states, etc., over the operating envelope.

4.6 TRANSIENT RESPONSE CHARACTERISTICS

4.6.1 The requirement in Chapter 600, para 11.5.3 avoids any difficulty of quantitatively recognising a final value of the response, by making the value reached after a specified time interval the reference quantity, and then limiting subsequent deviation from that value (see Figs. 3 and 4).

4.6.2 In practical terms of precision of control over the rotorcraft it would often be more relevant to call for required velocity changes being achieved within a specialised displacement of the rotorcraft, rather than an elapsed time. However, in terms of demonstrating compliance, a time criteria is more practicable.

4.6.3 Fig. 3 refers to LEVEL 1 handling qualities, and indicates a single step control input. This is the ultimate, ideal, discrete control of input. It remains to be established whether or not sensible requirements can be based on such an input.

4.6.4 It also remains to develop the quantitative criteria, but it is anticipated that for precise control tasks, time intervals (T_1 in Fig. 3) less than 5 seconds may be involved. It should also be noted that initial system response lags and delays, and the decay of other transient modes immediately following the input, are likely to occupy a significant proportion of these specified time intervals, and this precludes their treatment in terms of simple exponential time constants. The holding tolerance of ± 10% of the reference value is provisionally offered for guidance.

4.6.5 Fig. 4 refers to lower LEVELS of handling qualities. This recognises an increased pilot workload in the form of the discrete control input, involving some degree of initial overshoot and correction, before entering the settling period. T_2 may also be longer than T_1 when finally quantified.

4.6.6 It remains to establish meaningful restraints on the control overshoot which should be allowed in the present context. It is anticipated that this overshoot could not realistically exceed 50%-100% of the final control input. It would probably also be relevant to specify that no more than two reversals of control be allowed after the initial application, before reaching the final setting.

4.6.7 The setting time (T_2 in Fig. 4) should still be defined from the completion of the discrete control input, as this then relates to the pilot's ability to judge the suitability and effectiveness of his control activity.

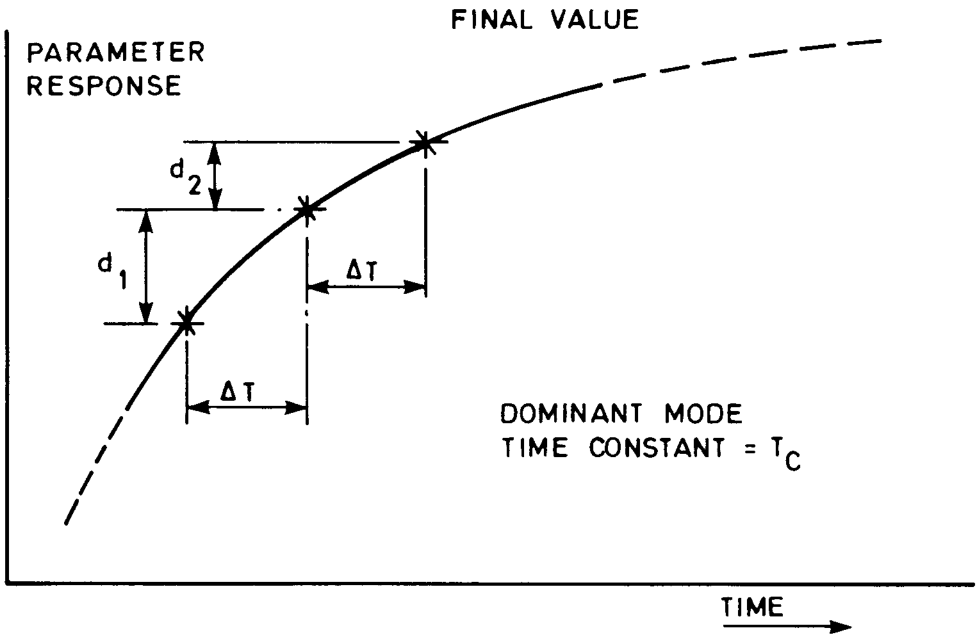


FIG.1 TYPICAL APERIODIC RESPONSE TO STEP INPUT

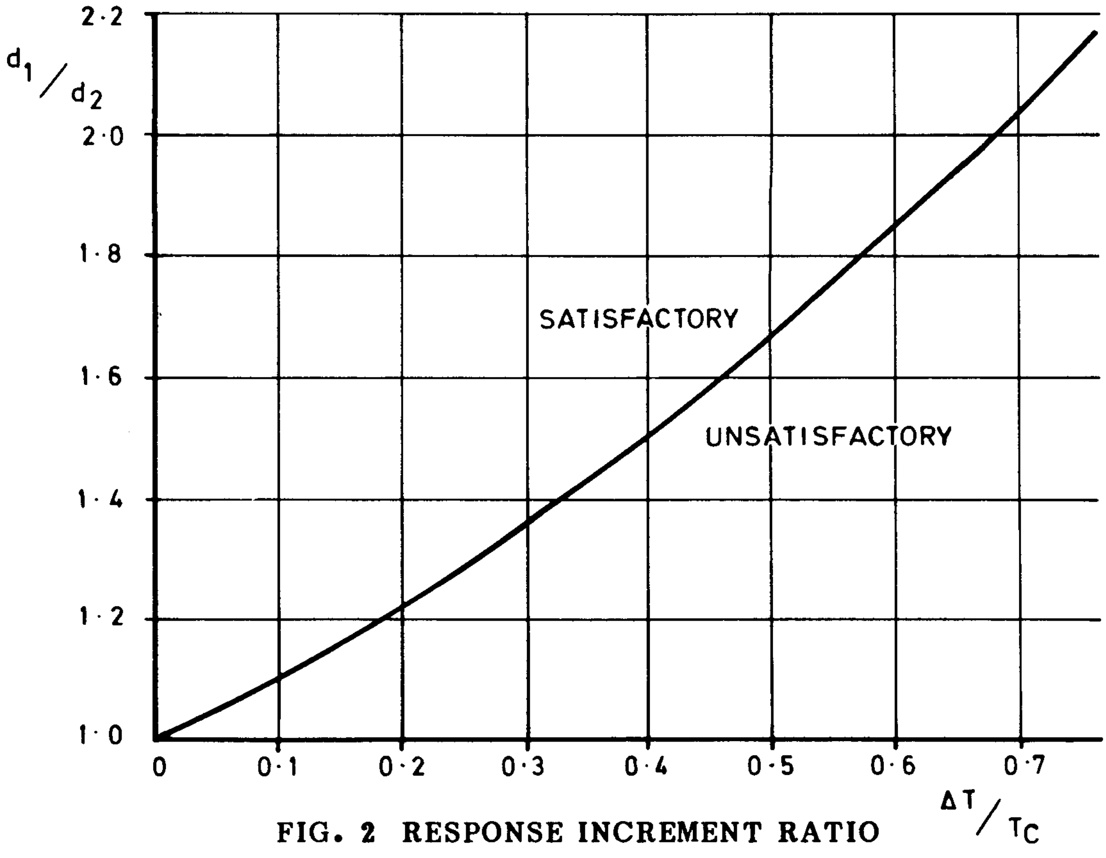


FIG. 2 RESPONSE INCREMENT RATIO

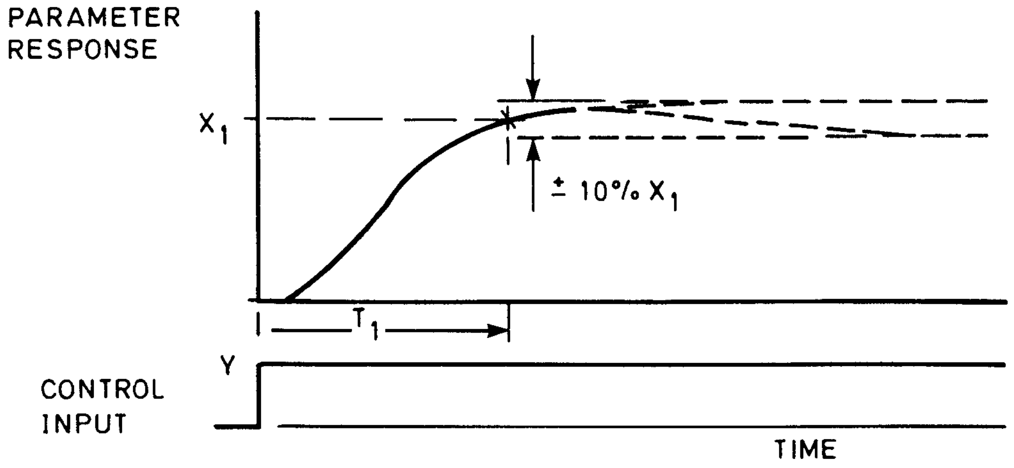


FIG.3 TRANSLATIONAL RESPONSE - LEVEL 1

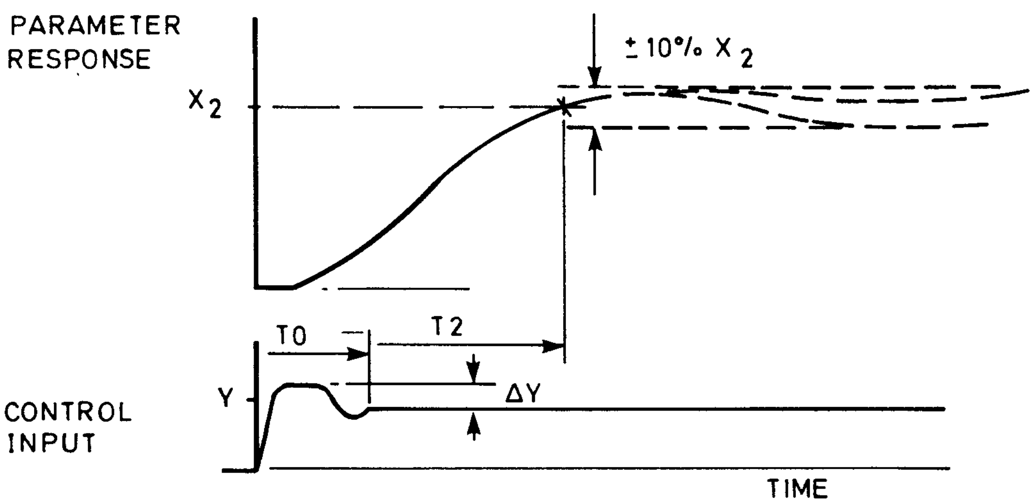


FIG. 4 TRANSLATIONAL RESPONSE - LEVEL 2 AND 3

LEAFLET 600/9

GENERAL REQUIREMENTS AND DEFINITIONS

CONTROL MARGINS

1 INTRODUCTION

1.1 This leaflet relates to the general requirements for control margins introduced in Chapter 600 para 12. The objective is to ensure that adequate control range is built into the rotorcraft to allow specific response targets to be met in manoeuvring and recovery from upsets or failure situations, rather than to merely leave an arbitrarily chosen percentage of the full control ranges available at the most critical steady state flight conditions.

1.2 This purpose should generally be met by the angular rate and vertical axis considerations summarised in Chapter 600 paras 12.3.1 and 12.3.2, respectively. In some critical low level operating conditions, translational motion considerations relevant to obstacle avoidance may be necessary, but remain to be identified.

2 MARGINS FOR MANOEUVRE

2.1 At the operational conditions defined in Chapter 600, para 12.3.1 there should be sufficient control available in each channel to generate responses within 1.5 seconds, in the critical sense, which are not less than those shown in paras 2.2 and 2.3.

2.2 ACTIVE FLIGHT PHASES

ANG RATE (DEG/SEC)	AGGRESSIVE	MODERATE
Pitch	15	10
Roll	15	10
Yaw (Forward Flight)	15	10

Pending alternative data, and unless contradicted by the Rotorcraft Specification, at any steady flight condition in the Low Speed Envelope, vertical agility should not be less than:

	AGGRESSIVE	MODERATE
Increment in Normal Accn (g)	±0.25 g	±0.12 g

2.3 ATTENTIVE FLIGHT PHASES

ANGULAR DISPLACEMENT	(DEG)
Pitch	10
Roll	10
Heading	6

3 MARGINS FOR RECOVERY

3.1 GUST DISTURBANCES

3.1.1 The rotorcraft design process will include theoretical assessments of the effects of atmospheric disturbances. These studies should lead to estimates of the control margins required to deal with large discrete gusts. Leaflet 600/4 para 4.3 describes the forms of discrete gust models which should be used. Leaflet 600/4 para 5 also relates gusts to steady wind conditions and may be relevant in considering disturbed conditions in the Low Speed Envelope.

3.1.2 The severity of gusts to be considered should be agreed between the contractor and the Rotorcraft Project Director. Gusts used for any handling studies should be consistent with those used in structural strength analyses.

3.1.3 Pending alternative data the following is given for initial guidance:

(i) Forward Flight Envelope. Linear ramp model, based on reference intensities in the 'Extreme' range given in Leaflet 600/4, para 3.1 coming from any direction. In general symmetric response characteristics tend to be most sensitive to vertical (up) gusts at high speed. Lateral/directional responses will have a critical azimuth direction for the horizontal gust, and attention should be focused on that situation, once identified.

(ii) Low Speed Envelope. Linear ramp model based on Leaflet 600/4, para 5 and the maximum steady wind of the envelope. There is likely to be a different critical quartering azimuth for the horizontal wind direction for each of the longitudinal, lateral and directional control channels.

3.2 CONTROL STRATEGY

3.2.1 Corrective control inputs are a very subjective matter. Pending further advice, simple ramp inputs to various levels are suggested as a means of theoretically investigating control inputs required to contain gust disturbances. This can also be employed in the context of recovering from major upsets due to flight control system failures etc.

3.2.2 There should be some delay, from the onset of the gust or the system failure to the initiation of the control input. This will be dependent on the state of attentiveness of the pilot, flight phase, etc., and remains to be detailed.

CHAPTER 601

LONGITUDINAL FLIGHT HANDLING QUALITIES

1 INTRODUCTION

1.1 The requirements in Chapter 601 relate to those specific aspects of operation of the longitudinal flying controls, associated rotorcraft response, and translational and pitching motion in the nominal plane of symmetry, not already dealt with in Chapter 600.

2 PITCH ATTITUDE BEHAVIOUR (See also Leaflet 601/1 para 2)

2.1 CHANGES OF FLIGHT CONDITION

2.1.1 The rotorcraft shall be designed to minimise, as far as is practicable, the variation of pitch attitude when trimmed at a particular airspeed in the Forward Flight Envelope, as power is varied over the full range applicable. (See also Leaflet 601/2, para 4.1).

2.1.2 In cases where the Rotorcraft Specification calls for aggressive approaches to the hover, the rotorcraft design shall be such that the nose up attitudes involved are not sufficiently high to result in:

(i) Unacceptable degradation of the pilot's external field of view in critical directions.

(ii) Inadequate clearance between critical rotorcraft components and external surfaces or obstacles.

2.1.3 If an active aerodynamic surface is introduced to modify pitch attitude characteristics in any of the operating regimes referred to in para 2.1.1, no single failure of the actuation/control sub-system shall result in:

(i) A dangerous flight condition developing if the pilot's reaction is delayed for an appropriate intervention time.

(ii) The rotorcraft being prevented from achieving, a safe, sustained flight condition, and subsequently landing safely.

2.2 LONG TERM LONGITUDINAL DYNAMIC STABILITY

2.2.1 The requirements in para 2.2 refer primarily to the non-manoeuvring, near-steady state conditions prevailing in the Attentive and Passive flight phases, with flying control settings retained as described in Chapter 600 para 9.2.2. Unless otherwise stated steady atmospheric conditions are assumed. (See Leaflet 601/1 para 3.1).

2.2.2 Attentive Flight Phases. The rotorcraft shall not deviate unacceptably in pitch from a trimmed flight condition if the flying controls are left unattended for short periods of time. (See Leaflet 601/1 para 3.2.2).

2.2.3 Passive Flight Phases. The rotorcraft shall be well stabilised against the effects of external disturbances allowing the pilot's attention to be diverted from flying the rotorcraft, for protracted periods. (See Leaflet 601/1 para 3.2.3).

2.3 SHORT TERM LONGITUDINAL RESPONSE CHARACTERISTICS

2.3.1 The requirements of para 2.3 relate primarily to the up and away Active Flight Phases, but also have some relevance to the Attentive Passive Phases. (See Leaflet 601/1 para 4.1).

2.3.2 Active Flight Phases

(i) Aggressive Manoeuvring Tasks. Pitch rate response shall be in accordance with Leaflet 601/1 para 4.2.4.

(ii) Moderate Manoeuvring Tasks. Pitch rate and attitude response characteristics shall be in accordance with Leaflet 601/1 para 4.2.5.

(iii) For those rotorcraft which have response characteristics enabling aggressive manoeuvring tasks to be carried out at LEVEL 1, the pilot shall also have no difficulty (e.g., due to any oversensitivity) in the precise execution of gentle manoeuvres or establishing steady flight conditions when required. (See Leaflet 601/1 para 4.2.2).

2.3.3 Attentive Flight Phases. Pitch attitude response shall be in accordance with Leaflet 601/1 para 4.3.

2.3.4 Passive Flight Phase. The operation of any autopilot modes available in the rotorcraft shall not result in any unacceptable short term pitch characteristics (see Leaflet 601/1 para 4.4).

3 AIRSPEED CONTROL

3.1 The requirements of this paragraph primarily relate to indicated airspeed along the flight path, and shall be met for all critical configuration and loading cases.

3.2 LONGITUDINAL STATIC STABILITY

3.2.1 A forward longitudinal control displacement or force shall be required in order to initiate and maintain an increased forward airspeed. Similarly an aft control input shall be required to initiate and maintain a reduced forward airspeed.

3.2.2 The requirements in para 3.2.1 shall be met from any trimmed flight condition within the Forward Flight Envelope:

(i) Above the minimum power speed or minimum speed for IFR (if defined), whichever is the lower, up to maximum normal operating speed.

(ii) With the collective pitch or power control, whichever is appropriate, fixed at the trimmed setting.

(iii) See also Leaflet 601/2 paras 2.1 and 2.2.

3.2.3 The requirement in para 3.2.1 shall also be met at trimmed conditions in the Low Speed Envelope:

(i) Over the limited speed range about the hover defined in Leaflet 601/2 para 2.3.1.

(ii) With the rotorcraft height maintained constant.

(iii) See also Leaflet 601/2 para 2.3.

3.3 TRIMMING TO AN AIRSPEED

3.3.1 At any trimmed flight condition in the Forward Flight Envelope the pilot shall be able to readily retrim, in either direction, to a slightly different desired airspeed without having to exercise undue skill or make repeated adjustments. In particular:

(i) The means available to the pilot for longitudinal trimming shall not be over-sensitive.

(ii) The effective overall time constant of the airspeed response to a single trim input by the pilot shall not be excessive. (Leaflet 601/2 paras 3.2 and 3.3).

3.4 MAINTAINING AIRSPEED IN CLIMB AND DESCENT

3.4.1 From any steady level trimmed flight condition at airspeeds between the minimum power speed or minimum speed for IFR, whichever is the least, and normal maximum operating speed minimal longitudinal control adjustments by the pilot shall be required to maintain airspeed substantially constant at the initial trimmed value following moderate power adjustments, for example those associated with acquiring and maintaining specific flight paths in controlled airspace or in similar manoeuvres. See also Leaflet 601/2 para 4.

3.4.2 Rotorcraft characteristics shall not result in pilot induced excursions in airspeed or vertical velocity while attempting to follow a specific flight path dictated by air traffic control requirements or the execution of particular mission tasks.

3.5 AIRSPEED INDICATION SYSTEM BEHAVIOUR

3.5.1 The requirements of Part 6 assume that the system shall provide consistent indications of relative airspeed along the flight path for all angles of incidence and sideslip associated with normal manoeuvres required of the rotorcraft throughout the Forward Flight and Sideslip Envelopes.

3.5.2 The minimum airspeed above which consistent indications are produced during acceleration from low speed, and the minimum speed down to which consistent indications are produced during decelerations, shall be repeatable for each rotorcraft loading and configuration.

4 OPERATIONS IN THE LOW SPEED ENVELOPE

4.1 The requirements of this paragraph relate primarily to low level operation and low speed manoeuvring during Active Flight Phases, but also refer to attentive flight phases under autopilot control. The major requirements centre around the precision of translational velocity and positioning of the rotorcraft relative to external references, fixed or moving. (See also Leaflet 600/8, para 4).

4.2 ATTITUDE CONTROL

4.2.1 Where control augmentation systems involving feedback of translational velocity and/or positional data are introduced to achieve high levels of positional control as referred to in para 4.3, overall system characteristics shall be such that:

(i) Adequate dynamic attitude stabilisation is retained (Leaflet 600/8, para 4.1.3).

(ii) Attitude excursions are not sufficiently large or rapid to be disconcerting or impede critical outside visual referencing.

4.2.2 Where the form of control augmentation indicated in para 4.2.1 is not adopted the pitching response characteristics shall meet the requirements in para 2.3.2.

4.3 TRANSLATIONAL AND POSITIONAL CONTROL

4.3.1 For these Active flight phases which demand precise control over the fore and aft positioning of the rotorcraft relative to an external reference, fixed or moving, the requirements in para 4.3.2 to 4.3.4 shall be met under the following conditions.

(i) In steady winds up to the limits of the Low Speed Envelope.

(ii) With rotorcraft height maintained substantially constant by means of appropriate adjustments to the power available, and torque compensation where relevant.

(iii) Assuming the pilot has adequate visual cues relevant to the particular external reference.

4.3.2 The change in fore and aft relative air velocity per unit longitudinal control input shall satisfy the general requirements in Chapter 600, para 11.5.2 and be in accordance with the sensitivity guidelines in Leaflet 601/3, para 2.3.

4.3.3 The transient response in fore and aft relative air velocity to a discrete longitudinal control input shall be in accordance with the general requirements in Chapter 600, para 11.5.3 and Leaflet 601/3, para 2.4.

4.3.4 Following discrete longitudinal control input (as in para 4.3.3), with the lateral control retained at its trimmed setting and heading maintained substantially constant by appropriate directional control input, the change in the rotorcraft lateral velocity component shall not be excessive (See Leaflet 601/3, para 2.5).

4.4 FLIGHT UNDER AUTOPILOT CONTROL

4.4.1 Where the Rotorcraft Specification calls for particular mission tasks to be carried out under automatic control either to achieve a repeatable degree of positional accuracy, or because the absence of adequate external reference cues precludes active pilot execution of the tasks:

(i) The required level of response and/or positional accuracy will be defined by the Rotorcraft Specification or the autopilot system specification.

(ii) The requirements of para 4.2 are still applicable.

4.5 RECOVERY FOLLOWING A FAILURE

4.5.1 The rotorcraft design shall be such that, in the event of a critical engine or flight control subsystem failure, resulting in loss of height while the rotorcraft is in near-hovering flight close to the ground or similar surfaces, the pilot can readily initiate forward motion of the rotorcraft through instinctive use of the flying controls in order to carry out either a fly-away manoeuvre to a safe flight condition or a safe touch down, in accordance with recommended emergency procedures.

4.5.2 The requirements in para 4.5.1 shall be applicable to Active Flight Phases and also Attentive Flight Phases with the rotorcraft under autopilot control but monitored by the pilot, having regard to appropriate pilot intervention delay times.

5 MANOEUVRES IN THE FORWARD FLIGHT ENVELOPE

5.1 SYMMETRIC MANOEUVRES

5.1.1 The requirements of para 5.1 are concerned with changes in flight condition involving airspeed, pitch attitude, incidence and normal load factor. These requirements complement a number of those associated with the short term response characteristics of para 2.3 and Leaflet 600/7. Unless otherwise stated these requirements refer to rotorcraft responses to longitudinal flying control inputs with other controls retained at their initial trimmed settings.

5.1.2 Pull-Up Manoeuvres

(i) Following a single aft control step input, resulting in a maximum increment in normal acceleration in the range ½g to ¾g, the time history of the increment in normal acceleration shall always increase positively, without undue delay, and in a progressive manner, until the maximum is reached.

(ii) Following a longitudinal control step input falling within "Moderate" manoeuvre category, at a given airspeed, the peak increment in normal acceleration shall be substantially proportional to the magnitude of the control input.

(iii) For a given control input as described in para 5.1.2 (ii) the peak increment in normal acceleration shall increase progressively with increasing initial trimmed airspeed, subject to rotor overspeed limits not being exceeded.

(iv) During "Aggressive" manoeuvres there shall be no unexpectedly rapid increase in the rate of pitch, rate of increase in normal acceleration, or rotorspeed which is not preceded by unambiguous tactile or aural cues to the pilot, that the lifting system is approaching an operating state involving aerodynamic non-linearities and/or potentially critical loading conditions. (See Leaflet 601/4, para 2).

(v) In cases where the basic rotorcraft characteristics provide insufficiently acute cues, these may be augmented by artificial warning devices. (Leaflet 601/4, para 3).

(vi) In the event of any indication of the severity of the manoeuvre increasing, as in (iv), it shall be possible for the trend to be rapidly checked by an instinctive relaxation of the control input generating the manoeuvre.

(vii) In pull up manoeuvres generated by combinations of longitudinal and lifting flying control inputs, there shall be a harmonious and predictable interchange of effects on the rotorcraft response due to each control channel, and the general qualities referred to in the remainder of para 5.1.2 shall be retained.

5.1.3 Push-Over Manoeuvres

(i) In push-over manoeuvres, with load factors approaching, the low or negative design limits, instinctive recovery control inputs shall produce a nose up pitch rate, with no tendency, however transient, to promote further nose down motion. Leaflet 601/4, para 4.

5.1.4 Accelerations and Decelerations

(i) The requirements of para 5.1.4 relate to larger, more aggressive speed changes than those associated with the airspeed trim adjustments addressed in para 3.3.

(ii) To carry out any quick stop manoeuvre necessary in mission tasks called for by the Rotorcraft Specification, the rotorcraft design shall be such that sufficient retarding force can be generated without involving a nose up attitude large enough to lead to rotor torque off-loading resulting in rotor overspeed limits being exceeded.

5.1.5 Harmonisation of Controls

(i) The pilot shall not be required to exercise undue levels of skill or effort in applying corrective inputs through the lateral and directional flying controls in order to maintain the required degree of symmetry when carrying out the manoeuvres referred to in paras 5.1.2 to 5.1.4, with the aggressiveness necessary to achieve the mission task requirements of the Rotorcraft Specification.

5.2 ASYMMETRIC MANOEUVRES

5.2.1 The requirements of para 5.2 relate to longitudinal control and response aspects of manoeuvres initiated through the lateral or directional flying controls.

5.2.2 Turning Flight

(i) In manually flown banked turn manoeuvres, initiated from steady straight and level flight conditions, an aft longitudinal control displacement or force shall be required to maintain the initial airspeed and an increased load factor.

(ii) In aggressive turning manoeuvres, rapid roll initiation or reversal shall not require unacceptably large transient longitudinal control forces or displacements to retain proper orientation and control over the flight path of the rotorcraft and rotor speed.

(iii) The pitch up avoidance requirement of para 5.1.2 (iv) shall also be satisfied in aggressive turning manoeuvres.

5.2.3 Sideslipping Flight

(i) There shall be no objectionable changes in rotorcraft pitch attitude, or longitudinal control required to contain any pitching tendencies, during any transient or sustained operation within the sideslip envelope.

(ii) If steady sideslip is progressively increased, in either direction, at any given airspeed, any associated pitch attitude or longitudinal control adjustments shall also be progressive with no unexpected discontinuities or reversals.

6 CONTROL MARGINS

6.1 There shall be adequate longitudinal control available in the rotorcraft to meet the general requirements of Chapter 600 para 12 with reference to Leaflet 600/9.

6.2 LOW SPEED ENVELOPE

6.2.1 There shall be an adequate rearward margin of longitudinal control to meet Chapter 600 para 12.3.1 when the rotorcraft is:

(i) Loaded to the critical total weight with the maximum forward, combined with any adverse lateral centre of gravity location.

(ii) In steady trimmed flight with the critical combination of relative wind speed and azimuth direction called for by the Rotorcraft Specification.

6.2.2 In those cases where the critical flight condition in para 6.2.1 (ii) arises with the rotorcraft stationary relative to some external reference, there shall be adequate longitudinal control available to allow the pilot to fly the rotorcraft to that condition, from any other condition in the Service Flight Envelope without exercising undue skill or anticipation in initiating the manoeuvre to avoid overshooting, the external reference.

6.3 FORWARD FLIGHT ENVELOPE

6.3.1 There shall be an adequate forward margin of longitudinal control to meet Chapter 600 para 12.3.1 when the rotorcraft is:

(i) Loaded to the critical total weight with the maximum rearward, combined with any adverse lateral centre of gravity location.

(ii) In steady trimmed flight at the appropriate maximum normal operating speed.

6.3.2 The requirements of para 6.3.1 shall also be met throughout the Sideslip Envelope.

6.4 AFCS TRIMMING

6.4.1 For those rotorcraft in which the inherent pitch instability required stability augmentation to meet the requirements of Part 6, with series actuators operating close to null, (Chapter 600 para 10.2.6), any associated actuator trimming device shall have sufficient authority to allow null offsets to be kept small enough to ensure:

(i) Adequate remaining actuator authority for stabilisation of the rotorcraft against external disturbances.

(ii) Acceptable pilot intervention times in the event of an actuator runaway in the adverse sense from any such null offset.

6.4.2 The requirements of para 6.4.1 shall be met for the most adverse combinations of flight condition and loading, including, centre of gravity shifts due to the release of stores, and system design tolerances.

LEAFLET 601/1

LONGITUDINAL FLIGHT HANDLING QUALITIES

PITCH RESPONSE CHARACTERISTICS

1 INTRODUCTION

1.1 This leaflet provides quantitative guidance on the initial response to longitudinal control inputs and pitch dynamic stability characteristics referred to in Chapter 601, para 2.

2 SIGNIFICANCE OF PITCH ATTITUDE AS A PARAMETER

2.1 By virtue of the ease with which variations in them can be recognised by the pilot, instrumented and applied in feedback control systems, pitch attitude and its derivatives with respect to time are prime indicators of longitudinal stability and control characteristics.

2.2 Although pitching behaviour is fundamental to the flight mechanics of conventional rotorcraft, (e.g., the generation and control of fore/aft motion), the necessity for rigid quantitative holding of a particular datum attitude, per se, under varying transient flight conditions is actually quite limited. Strict attitude retention may be required for specific mission phases involving specialised aiming or tracking tasks, or where the rotorcraft is required to act as a highly stabilised platform, but more often rotorcraft are actually operated with the intention of maintaining a particular airspeed or groundspeed, rather than a pitch attitude in its own right.

2.3 It should be noted that, as a corollary to Leaflet 601/2, para 4.1, under conditions of rigid attitude control in the presence of power changes etc., variations in airspeed are likely to be induced. The acceptability or otherwise of such variations must be considered in deciding upon the control strategy for the rotorcraft.

3 LONG TERM DYNAMIC STABILITY

3.1 RATIONALE

3.1.1 The requirements of Chapter 601 para 2.2 are expressed qualitatively, in line with the general rationale that rotorcraft handling characteristics should not prevent any specific mission accomplishment, whatever that implies.

3.1.2 Para 3.2 provides quantitative guidance applicable to general operation in Attentive and Passive flight phases. For those rotorcraft that are required to carry out specialised tasks demanding closer control over long term pitch attitude behaviour, appropriately stricter limitations will be given in the Rotorcraft Specification.

3.1.3 No specific requirement is laid down for Active flight phases since the virtually continuous control applications inherent in these phases would render any demonstration of compliance impossible. Nevertheless for satisfactory operations in the active category long period stability characteristics should be reasonably damped, no worse than those described in para 3.2.2(iii).

3.1.4 The overall objective of the attitude limitations in para 3.2 is to contain long term drift, or the amplitude of any long period response modes. The pitch rate criteria is intended to prevent noticeable changes in attitude occurring from time to time while the controls are unattended, as a result of automatic control system operation in any axis.

3.1.5 The quantitative motion criteria used in para 3.2.1 are virtually threshold levels and it is appropriate to use the same values for each flight phase, and also for LEVEL 1 and 2 handling qualities, where the differing time intervals over which the criteria can be met reflect the associated changes in pilot workload.

3.2 QUANTITATIVE GUIDANCE

3.2.1 Following release of the flying controls, pitch attitude should not vary by more than ±1 deg and pitch rate should not exceed ±3 deg/sec during the intervals given in paras 3.2.2(i), (ii) and 3.2.3 (i) and (ii).

3.2.2 Attentive Flight Phases

(i) LEVEL 1: 20 seconds

(ii) LEVEL 2: 10 seconds

(iii) LEVEL 3: Any oscillatory mode with a period greater than 5 sec but less than 10 sec should halve its amplitude in no more than 2 cycles. Oscillations with periods up to 20 sec should be damped, those with periods greater than 20 sec should not double amplitude in less than 20 sec. Any aperiodic mode should not double amplitude in less than 9 sec.

3.2.3 Passive Flight Phases

(i) LEVEL 1: 3 minutes

(ii) LEVEL 2: 1 minute

(iii) LEVEL 3: is not addressed here, as it is compatible with passive phases of flight.

4 SHORT TERM RESPONSE CHARACTERISTICS

4.1 RATIONALE

4.1.1 For those operational phases away from ground restraints or obstacles, or not requiring precise positional control of the rotorcraft, short term longitudinal handling is dominated by pitch response characteristics. This includes both the initial responsiveness to pilot control inputs and dynamic stability characterised by the transient recovery from a disturbance, as dealt with in general terms by Chapter 600, paras 10.1 and 11.3.

4.1.2 Generalized criteria, identified in Leaflet 600/7 are here related to pitch rate or pitch attitude, depending on the particular flight phase or level of handling quality involved.

4.1.3 Quantitative data provided in para 4 is provisional, pending confirmation or the derivation of more meaningful data.

4.1.4 The time at which the initial response should be monitored for adequate responsiveness without oversensitivity (T_I in Fig. 1 of Leaflet 600/7) remains to be optimised in terms of practicality of timing and resolution between acceptable and unacceptable response characteristics. It is anticipated that 0.5 sec may be suitable and this is provisionally proposed for all cases.

4.2 ACTIVE FLIGHT PHASES

4.2.1 Continuous flight of the rotorcraft through the pilot's flying controls has been further categorised into Aggressive and Moderate manoeuvring tasks with the intention of specifying minimum requirements for each, where relevant.

4.2.2 The requirement in Chapter 601, para 2.3.2 (iii) is included to highlight the fact that, although an aggressive manoeuvre capability may be called for by the Rotorcraft Specification, this can only represent part of the total operational spectrum of the rotorcraft, and that rotorcraft designed to meet exacting requirements for aggressive manoeuvring must also be easy to fly in the gentle manoeuvre and near-steady conditions comprising the remainder of the service envelope.

4.2.3 Table 1 ascribes provisional numerical values to the initial response and stability criteria identified in Leaflet 600/7, paras 3 and 6, and Fig.1.

4.2.4 Aggressive Manoeuvres

(i) For LEVEL 1 handling characteristics a pulse input through the longitudinal flying control should produce a pitch rate type of rotorcraft response in accordance with the first column of Table 1.

(ii) Reduced handling qualities in terms of less responsiveness, greater sensitivity, larger overshoot and longer settling time are reflected in the wider parameter ranges quoted for LEVEL 2 compared with LEVEL 1.

(iii) LEVEL 3 is currently not addressed in Table 1. It would be inappropriate for the pilot to embark upon deliberately aggressive manoeuvres with the rotorcraft in a sufficiently degraded operating state that led to the workload in controlling the rotorcraft approaching the limits of the pilot's capability.

4.2.5 Moderate Manoeuvres

(i) For LEVEL 1 handling characteristics a longitudinal control input should also produce a pitch rate type of rotorcraft response. As shown in Table 1 the peak response to the standard control input does not have to be as high as for aggressive tasks, but no distinction is made for the dynamic stability criteria.

(ii) If the longitudinal control input generates a pitch attitude, rather than a pitch rate, type of response the additional pilot anticipation required in accurately executing manoeuvres is likely to lead to LEVEL 2 or 3 qualities, depending on control sensitivity or dynamic stability characteristics expressed in terms of pitch attitude as in Table 1.

(iii) For LEVEL 3 handling characteristics any short period oscillatory modes should be damped. Where flight under IFR is required, oscillations having a period of 5 sec or less should halve amplitude in less than 1 cycle, and those with a period greater than 5 sec in less than 2 cycles. For flight under VFR, oscillations with a period of 5 sec or less should halve amplitude in less than 2 cycles.

TABLE 1

SHORT TERM RESPONSE - ACTIVE FLIGHT PHASES

Man.Clas.	**Aggressive**			**Moderate**		
LEVEL	**1**	**2**	**3**	**1**	**2**	**3**
Response Parameter	Pitch Rate	Pitch Rate		Pitch Rate	Pitch Att.	Pitch Att.
Peak Response	10-15°/sec	7-20°/sec	N/A	5-10°/sec	5-10°	3-15°
T_I(sec)	0.5	0.5		0.5	0.5	0.5
Y_1%	30-TBA	30-TBA		30-TBA	30-TBA	30-TBA
Y_2%	5	10		5	0	
T_{30}(sec)	<1	<1	N/A	<1	<1.5	See Para 4.2.5 (iii)
T_{11}(sec)	-	1-2		-	1.5-3	
T_{01}(sec)	1-2	-		1-2	-	
X_1%	15	20		15	25	
T_{02}(sec)	>2	>2		>2	>2.5	
X_2%	10	15		10	15	
T_F(sec)	3	5		3	5	
X_F%	10	10		10	10	

4.3 ATTENTIVE FLIGHT PHASES

4.3.1 For the satisfactory execution of attentive flight phases the rotorcraft should have the stability characteristics associated with a pitch attitude type of response. Following the standard pulse input through the longitudinal flying control, the pitch attitude response should be in accordance with the criteria identified in Leaflet 600/7 and quantified in Table 2.

TABLE 2

SHORT TERM RESPONSE - ATTENTIVE FLIGHT PHASES

LEVEL	**1**	**2**	
Peak Response	4.8°	4-12°	
T_1(sec)	0.5	0.5	
Y_1%	30-TBA	30-TBA	
Y_2%	0	0	
T_{30}(sec)	<1	<1.5	
T_{11}(sec)	-	1.5-3	
T_{01}(sec)	1-2	-	
X_1%	10	15	
T_F(sec)	3	5	
X_F%	10	10	

4.4 PASSIVE FLIGHT PHASES

4.4.1 Pilot applied control inputs are not relevant to Passive Flight Phases, but short term dynamic stability characteristics are.

4.4.2 Rotorcraft pitch attitude behaviour should be well stabilised against external disturbances (e.g., atmospheric turbulence). Also the functioning of any autopilot modes (e.g., controlling speed or height of the rotorcraft) should not result in any untoward excursions in pitch attitude or the setting up of any short period oscillatory coupling with pitch attitude.

4.4.3 Pending the derivation of any alternative criteria, the transient recovery of pitch attitude from a single disturbance should be in accordance with that for Attentive flight phases given in the second part of Table 2.

LEAFLET 601/2

LONGITUDINAL FLIGHT HANDLING QUALITIES

AIRSPEED CONTROL

1 INTRODUCTION

1.1 This leaflet relates to the requirements contained in Chapter 601 para 3.

2 LONGITUDINAL STATIC STABILITY

2.1 Under the forward flight conditions of Chapter 601 para 3.2.2., and where operation of the rotorcraft calls for close regulation of airspeed, especially over protracted periods of time, or ease of trimming to speeds close to flight envelope limits, the relationship between control displacement or force and the change in trimmed airspeed should have the following characteristics:

2.1.1 For LEVEL 1 handling qualities the slopes through a number of datum trim speeds throughout the relevant straight and level speed range should:-

(i) All be positive (i.e., forward with increasing speed).

(ii) Generally increase with increasing datum trim speed.

(iii) Not show a marked difference between successive datum points.

2.1.2 For LEVEL 2 handling qualities the criterion referred to in para 2.1.1(i) is applicable.

2.2 Compliance with the requirements of Chapter 601, para 3.2.2 should be demonstrated by establishing a steady datum trimmed condition and then, by means of longitudinal control adjustments only, re-establish steady flight conditions at a number of airspeeds in the range of 20 knots (unless otherwise stated) on either side of the original datum airspeed. The datum conditions should embrace the operating envelope, with any relevant rotorcraft configuration changes, including as a minimum:-

2.2.1 Cruise with the rotorcraft initially trimmed in steady straight and level flight at 0.9 maximum normal operating speed, and representative cruising altitude(s).

2.2.2 Level approach with the rotorcraft trimmed in steady straight and level flight at the minimum power speed or minimum speed for IFR, whichever is the least.

2.2.3 Descending approach at an airspeed and rate of descent or glide slope recommended by the Contractor or required by the Rotorcraft Specification.

2.2.4 Maximum continuous power climb at the recommended climbing speed.

2.2.5 Steady autorotational descent at indicated airspeeds to be agreed with the Rotorcraft Project Director in the range from 20 knots below the speed for minimum rate of descent, up to maximum normal operating speed. The ranges of retrimmed airspeed on either side of the datum speed should be determined so as to avoid any unacceptable effects on rotor speed or rate of descent at the fixed collective pitch settings.

2.3 Positive longitudinal static stability throughout the Low Speed Envelope would be highly desirable, but the relaxed requirement of Chapter 601, para 3.2.3 recognises that this is unlikely to be fully achieved in practice, due to various aerodynamic interference effects and non-linearities inherent in low speed flight conditions.

2.3.1 Pending alternative data, it is proposed that the slope of longitudinal control displacement or force versus fore and aft speed should be positive, and with little variation in magnitude, over the range 15 knots rearward to 15 knots forward, in the presence of lateral components up to ±10 knots.

2.3.2 At fore and aft speeds outside of the range given in para 2.3.1, but within the Low Speed Envelope, the longitudinal control slope should remain positive, but may reduce.

2.3.3 Specific reference to "indicated" airspeed has been avoided in para 2.3 since not all rotorcraft will necessarily be fitted with low airspeed indicating systems. The relevant parameter is intended to be the relative airspeed at the rotorcraft, this being the vector sum of rotorcraft groundspeed and any prevailing windspeed.

2.3.4 Since most operations in the Low Speed Envelope are likely to be carried out at low level, height variations associated with constant collective/power at varying translational speeds will not be acceptable. These are avoided by specifying constant height in the requirement of Chapter 601 para 3.2.3(ii).

3 AIRSPEED TRIMMING

3.1 BASIC ROTORCRAFT RESPONSE AND CONTROL

3.1.1 The airspeed response of a conventional rotorcraft to a simple longitudinal control input is typically governed by a long aperiodic time constant characteristic, of the form discussed in Leaflet 600/8 paras 2.1 and 3.2.

3.1.2 In the absence of an automatic flight control facility making use of appropriate feedback signals, the pilot is faced with the tasks of setting up, and making small adjustments to the rotorcraft airspeed. The basic technique for dealing with long time constant systems is outlined in Leaflet 600/8 para 4.1, the major problem being that of anticipating the appropriate net trim input to achieve the intended speed change, as required by Chapter 601 para 3.3.

3.2 TRIM SENSITIVITY

3.2.1 To avoid excessive hunting around in attempting to trim to a specific airspeed, the steady state speed change associated with the smallest discrete trim input which the pilot can apply consistently, should not exceed 2kt for LEVEL 1, and 3kt for LEVEL 2.

3.3 RATE OF CONVERGENCE

3.3.1 The longer the effective time constant of the airspeed response, the more difficult is the task of judging the correct control/trim input to achieve a specified final airspeed.

3.3.2 With the rotorcraft initially trimmed to a steady flight condition within the Forward Flight Envelope, following a small single input to the normal means of speed trimming, with no other control inputs, and allowing 2 secs for initial short term transients to decay, the ratio of increments in airspeed over 2 successive time intervals of 5 sec, as shown on Leaflet 600/8 Figs.1 and 2, should be not less than 1.8 for LEVEL 1, 1.4 for LEVEL 2, and 1.2 for LEVEL 3 handling qualities.

4 AIRSPEED IN CLIMB AND DESCENT

4.1 AERODYNAMIC TRIM BALANCE

4.1.1 In general, the effects of variation in rotorcraft angle of incidence over a range of rates of climb and descent at a given airspeed along the flight path, will dictate a variation in the longitudinal control actually applied to the rotorcraft control surfaces to maintain a pitching moment balance in the presence of combinations of:

(i) Variations in pitching moment generated by the airframe and any non-rotating aerodynamic surfaces.

(ii) Variations in direct rotor head moment and thrust vector rotation associated with flap bending, or effective disc tilt, due to changes in lifting rotor through flow conditions.

4.1.2 It should also be noted that, in general, as a consequence of the redistribution of pitching moments outlined in para 4.1.1, the trimmed pitch attitude of the rotorcraft will vary with rate of climb and descent at a given airspeed.

4.2 WORKLOAD REDUCTION

4.2.1 The requirements in Chapter 601 para 3.4 may be addressed by a number of approaches:

(i) Provision of an airspeed hold mode within the automatic flight control system.

(ii) Flying control mixing that applies an appropriate adjustment of effective longitudinal control when the lifting controller is operated to adjust the vertical velocity, while the pilot's longitudinal flying control remains at its original setting.

(iii) The corollary of para 4.1.2 should also be noted; namely that, if an automatic flight control system holds pitch attitude rigidly to some datum, appropriate to say level flight, then a variation in airspeed is likely to be induced in sustained climbing or descending conditions.

LEAFLET 601/3

LONGITUDINAL FLIGHT HANDLING QUALITIES

TRANSLATIONAL AND POSITIONAL CONTROL

1 INTRODUCTION

1.1 This Leaflet relates to requirements in Chapter 601 para 4, particularly those concerned directly with longitudinal positioning of the rotorcraft in the Low Speed Envelope.

1.2 The general background to the basic task of translational and positional control of rotorcraft is outlined in Leaflet 600/8.

2 VELOCITY CONTROL

2.1 The requirements in Chapter 601 para 4.3 are based on the assumption that the rotorcraft can be accurately positioned if the fore and aft velocity response to a pilot input is well conditioned. Ideally, following a step longitudinal control input, the fore and aft velocity should settle to a new value with the minimum delay or overshoot.

2.2 Achievement of these aims in a conventional configuration rotorcraft will almost certainly require a control augmentation facility, including translational parameter feedback. Insofar as this may be incompatible with operation in other flight regimes attention will need to be paid to means of engaging and disengaging such modes, including under transitory flight conditions as the Low Speed Envelope is entered and exited.

2.3 CONTROL SENSITIVITY

2.3.1 Under adverse operating conditions, precision of positional control will generally be more important than generating a relatively high final velocity. This is consistent with seeking a modest control sensitivity, combined with a short settling time.

2.3.2 Although this remains to be substantiated, it is proposed that for LEVEL 1 handling qualities a longitudinal control input equivalent to 10% of the full end to end range should produce a fore and aft velocity change in the range, 2-5 knots.

2.3.3 A greater control sensitivity is likely to lead to a lower LEVEL being achieved, due to increased pilot workload in anticipating control adjustments necessary to position the rotorcraft.

2.4 SETTLING TIMES

2.4.1 In the context illustrated by Leaflet 600/8 Figs. 3 and 4, 'settling time' is similar to the aperiodic time constant discussed in Leaflet 601/2 para 3 in connection with airspeed trimming. However, because of the shorter elapsed times allowable in the present context, and the likely presence of other transient modes, decaying during the early response time, it will not be possible to satisfactorily treat the time history of fore and aft velocity as a simple exponential curve. Hence the requirements in Chapter 600, para 11.5.3 being expressed in terms of a finite time interval which can be readily identified, after which the response should not vary significantly. This then allows the pilot to judge the adequacy of his control input for achieving his goal.

2.4.2 The settling time following a step input, as defined in Chapter 600 para 11.5.3, is provisionally set to 5 seconds for LEVEL 1. It remains to be established that this time interval allows adequate precision of control over the rotorcraft, and that it can in fact be realistically achieved within the other constraints on system design and rotorcraft response behaviour.

2.4.3 Where the performance required in para 2.4.2 could not be achieved, it is anticipated that the Level of handling quality would be degraded by:

(i) Increased pilot workload in having to compensate for control augmentation system shortcomings and apply a more adaptive input than the simple step associated with Level 1, to achieve a similar convergence towards a final velocity.

(ii) Increased time to get within the 10% final margin.

2.4.4 Meaningful values remain to be established for restrictions on the control input profile, and the associated settling times:

(i) It is anticipated that the control overshoot, as illustrated in Leaflet 600/8 Fig 4, could not realistically exceed 50% - 100% of the final control input.

(ii) Probably it would also be relevant to specify that no more than two reversals of control be allowed after the initial application, before reaching the final control setting.

(iii) The settling time should still be defined from the point of finalising the control input, as this determines the pilot's ability to judge the effectiveness of his control activity.

2.5 CROSS COUPLING EFFECTS

2.5.1 Under the conditions defined in Chapter 601, para 4.3.4 the rotorcraft lateral velocity component should not exceed ± 10% of the fore and aft velocity change within the following time intervals from the finalization of the discrete control input:

(i) LEVEL 1: Three times the fore and aft settling time.

(ii) LEVEL 2: Twice the fore and aft settling time appropriate to LEVEL 2.

LEAFLET 601/4

LONGITUDINAL FLIGHT HANDLING QUALITIES

LONGITUDINAL MANOEUVRES

1 INTRODUCTION

1.1 This Leaflet relates to the requirements in Chapter 601 para 5.

2 CRITICAL ROTOR OPERATING CONDITIONS

2.1 Encroachment of blade stall over the rotor disc area and compressibility effects on rotor aerofoil characteristics provide inherent limitations to lifting rotors under high speed and high rotor loading conditions.

2.1.1 Potentially damaging conditions may arise from combinations of increased vibratory loading and elevated mean levels of forces and bending moments in critical elements of rotor systems.

2.1.2 Penetration of adverse conditions, initiated by entry to an aggressive manoeuvre situation, may be exacerbated if the inherent tendency for significant blade stalling to generate nose-up rotor response is allowed to develop.

2.2 Onset of the effects referred to in para 2.1 may be delayed by suitable rotorcraft, blade, aerofoil or system design, but not entirely eliminated. Full exploitation of the potential of a high performance rotorcraft inevitably involves operation close to these critical boundaries. Any design refinements modifying the inherent rotorcraft behaviour in order to extend the operational envelope, should not achieve this at the expense of a more marked breakaway, should the deferred boundaries be subsequently reached.

3 CUES AND WARNINGS

3.1 In the highly Active flight phases, with his attention concentrated on the external situation, the pilot must be made aware of any approach to critical conditions by a recognisably progressive build up in suitable tactile or aural cues.

3.2 TACTILE CUES

3.2.1 Increase in the inherent rotor-generated vibration is a potentially powerful cue of penetration into the regimes of aerodynamic non linearity referred to in para 2.1. However, this may lack decisiveness if well founded engineering measures for vibration reduction or isolation are successful.

3.2.2 Control force cues may be augmented by introducing an additional force, related to the severity of the manoeuvre, which the pilot has to hold on in order to maintain a heavily loaded flight condition. The additional force should be based on monitored values, and rates of change of relevant manoeuvre or rotorcraft stress parameters, and as these approach critical levels the force should become a significant feature demanding conscious pilot reaction. Alternatively, some form of high frequency - low amplitude "stick shaker" triggered by the severity of the manoeuvre, might be applicable in certain control configurations.

3.2.3 If an automatic control sub-system is employed to provide structural protection by curtailing the severity of pilot induced manoeuvres to specified limits, the extent to which his control inputs are being backed off by the sub-system should be made unambiguously apparent to the pilot.

3.3 AURAL CUES AND WARNINGS

3.3.1 Variations in rotor and transmission noises associated with highly loaded, aggressive manoeuvring situations may provide the pilot with a valuable commentary on the increasing severity of a particular manoeuvre. However, this again may not have sufficiently fine resolution to cover full exploitation of the vehicle manoeuvre capability.

3.3.2 Audio warning signals, based on levels reached and rates of change of specifically monitored manoeuvre and rotorcraft parameters, may be fed to the pilot's communication system.

3.3.3 In the context of para 3.3.2 care must be taken to avoid unnecessary signals being passed to the pilot. It should also be established with the Rotorcraft Project Director for a particular rotorcraft that audio warnings can in fact provide sufficiently compulsive guidance to the pilot during critical mission phases, compared with perhaps more positive tactile cueing.

4 CONTROL POWER UNDER LOW-g CONDITIONS

4.1 A lifting rotor produces pitching moment control contributions from two sources:

4.1.1 A moment due to offset of the rotor thrust vector about the rotorcraft centre of gravity.

4.1.2 A direct head moment due to the mass characteristics of the blades and the effective flapping stiffness of their attachment to the shaft.

4.2 Under low rotor thrust conditions the control moment of para 4.1.1 will diminish, and will reverse under negative-g conditions, whereas the head moment of para 4.1.2 is totally independent of rotor thrust.

4.3 Pitch rate damping inherent in a lifting rotor, is also subject to the same considerations as in para 4.1 so that, unless supported by a reasonable direct head moment contribution the effective rotor response time constant will increase at low-g conditions.

4.4 The rotor head and blade attachment system should be designed to ensure adequate control power and an acceptably short response time constant down to the minimum thrust limits of the Service Flight Envelope.

CHAPTER 602

LATERAL FLIGHT HANDLING QUALITIES

1 INTRODUCTION

1.1 The requirements in Chapter 602 relate to those specific aspects of operation of the lateral flying controls, associated rotorcraft response, and lateral translation and rolling motion not already dealt with in Chapter 600.

2 ROLL ATTITUDE BEHAVIOUR (See Leaflet 602/1 para 2)

2.1 CHANGES OF FLIGHT CONDITION

2.1.1 The rotorcraft shall be designed to minimise, as far as is practicable, the lateral control adjustments necessary to maintain zero roll attitude when trimmed at a particular airspeed in forward flight as power is varied over the full range applicable.

2.1.2 During fore and aft translational flight throughout the Low Speed Envelope, with the aggressiveness necessary to meet the mission task requirements of the Rotorcraft Specification, the lateral control displacements or forces to maintain sufficiently balanced, near level roll attitude flight shall not be excessive.

2.1.3 In a steady, zero rate of yaw sideslip induced by a directional control input applied in a steady, straight and level flight condition anywhere in the Forward Flight Envelope, an increased roll attitude to the right shall be associated with increased sideslip to the right, and conversely for left sideslip.

2.2 LONG TERM LATERAL DYNAMIC STABILITY

2.2.1 The requirements in para 2. 2 refer primarily to the non-manoeuvring near-steady state conditions prevailing in the Attentive and Passive flight phases, with flying control settings retained as described in Chapter 600 para 9.2.2. Unless otherwise stated steady atmospheric conditions are assumed. (See Leaflet 602/1 para 3.1).

2.2.2 Attentive Flight Phases. The rotorcraft shall not deviate unacceptably in roll from its trimmed flight condition if the flying controls are left unattended for short periods of time. (See Leaflet 602/1 para 3.2.2).

2.2.3 Passive Flight Phases. The rotorcraft shall be well stabilised against the effects of external disturbances, allowing the pilot's attention to be diverted from flying the rotorcraft for protracted periods. (See Leaflet 602/1 para 3.2.3).

2.3 SHORT TERM LATERAL RESPONSE CHARACTERISTICS

2.3.1 The requirements of para 2.3 relate primarily to the up and away Active Flight Phases, but also have some relevance to the Attentive and Passive Phases. (See Leaflet 602/1 para 4).

2.3.2 Active Flight Phases

(i) Aggressive Manoeuvring Tasks. Roll rate response shall be in accordance with Leaflet 602/1 para 4.2.4.

(ii) Moderate Manoeuvring Tasks. Roll rate and attitude response characteristics shall be in accordance with Leaflet 602/1 para 4.2.5.

(iii) For those rotorcraft which have response characteristics enabling aggressive manoeuvring tasks to be carried out at LEVEL 1, the pilot shall also have no difficulty (e.g. due to any oversensitivity) in the precise execution of gentle manoeuvres or establishing steady flight conditions when required.

2.3.3 Attentive Flight Phases. Roll attitude response shall be in accordance with Leaflet 602/1 para 4. 3.

2.3.4 Passive Flight Phases. The operation of any autopilot modes available in the rotorcraft shall not result in any unacceptable short term roll characteristics. (See Leaflet 602/1 para 4.4).

3 SIDESLIP BEHAVIOUR

3.1 The requirements of this paragraph relate to the behaviour of the rotorcraft in sideslipping flight throughout the Service Flight envelope. (See Leaflet 602/2 para 2).

3.2 NORMAL OPERATING STATE

3.2.1 The rotorcraft design shall be such that during steady straight forward flight at nominally zero roll angle, sideslip angles giving rise to excessive parasitic drag forces or flow separation phenomena are avoided. (See Leaflet 602/2 para 3).

3.3 FAILURE STATES

3.3.1 In the case of rotorcraft configurations employing a thrust device to directly balance the torque reaction associated with driving rotating lifting surfaces, and where the Rotorcraft Specification calls for continued operation following failure of that torque compensating device the following requirements are applicable:

(i) Operational procedures and practicable envelopes of rotorcraft loading, altitude, forward speed and sideslip shall be defined within which straight and level flight can be sustained, and from which a safe landing could be made.

3.4 LATERAL STATIC STABILITY

3.4.1 A lateral control displacement or force to the right shall be required to maintain an increased steady, zero yaw rate, sideslip to the right initiated by a directional control input. Conversely a left lateral control input shall be required for a similar sideslip increase to the left.

3.4.2 The requirements in para 3.4.1 shall be met with the sideslip initiated from any trimmed flight condition within the Forward Flight Envelope:

(i) Above the minimum power speed or minimum speed IFR (if defined) whichever is the lower, up to a maximum normal operating speed.

(ii) With the collective pitch or power control whichever is appropriate, fixed at the trimmed setting.

(iii) See also Leaflet 602/2 para 4.1.

3.4.3 For LEVEL 1 and 2 qualities, with reference to para 3.4.1, the variations of lateral control input with sideslip angle shall be substantially linear. (See Leaflet 602/2 para 4.2).

3.4.4 The positive dihedral effect shall not be so great that, for any condition within the Sideslip Envelope, the lateral control inputs required by para 3.4.1 are greater than indicated in Leaflet 602/2 para 4.3.

3.4.5 From trimmed conditions in the Low Speed Envelope a lateral control displacement or force to the right shall be required to initiate and maintain an increased lateral velocity to the right. Conversely a left lateral control input shall be required for a lateral speed change to the left. These requirements shall be met:-

(i) With the rotorcraft height, heading and fore and aft speed maintained substantially constant.

(ii) See also Leaflet 602/2 para 4.4.

3.4.6 For LEVEL 1 and 2 qualities, with reference to para 3.4.5, the variations of lateral control input with lateral velocity shall be substantially linear (See Leaflet 602/2 para 4.5).

4 OPERATIONS IN THE LOW SPEED ENVELOPE

4.1 The requirements of this paragraph relate primarily to low level, operation and low speed manoeuvring during, Active Flight Phases. The major requirements centre around the precision of translation velocity and positioning of the rotorcraft relative to external references, fixed or moving. (See also Leaflet 600/8, para 4).

4.2 ATTITUDE CONTROL

4.2.1 Where control augmentation systems involving feedback of translation and/or positional data are introduced in order to achieve high levels of positional control, as referred to in Para 4.3, overall system characteristics shall be such that:

(i) Adequate dynamic attitude stabilisation is retained (Leaflet 600/8 para 4.1.3).

(ii) Attitude excursions are not sufficiently large or rapid to be disconcerting or impede critical outside visual referencing.

4.2.2 Where the form of control augmentation indicated in para 4.2.1 is not adopted the rolling response characteristics shall meet the requirements in para 2.3.2.

4.3 TRANSLATIONAL AND POSITIONAL CONTROL

4.3.1 For those Active flight phases which demand precise control over the lateral positioning of the rotorcraft relative to an external reference, fixed or moving, the requirements in paras 4.3.2 to 4.3.4 shall be met under the following conditions:

(i) In steady winds up to the limits of the Low Speed Envelope.

(ii) With rotorcraft height maintained substantially constant by means of appropriate adjustments to the power available, and torque compensation where relevant.

(iii) Assuming the pilot has adequate visual cues relevant to the particular external reference.

4.3.2 The change in lateral relative air velocity per unit lateral control input shall satisfy the general requirements in Chapter 600 para 11.5.2, and be in accordance with Leaflet 602/3, para 2.3.

4.3.3 The transient response in lateral relative air velocity to a discrete lateral control input shall be in accordance with the general requirements in Chapter 600 para 11.5.3 and Leaflet 602/3 para 2.4.

4.3.4 Following a discrete lateral control input (as in para 4.3.3), with the longitudinal control retained at its trimmed setting and heading maintained substantially constant by appropriate directional control input, the change in the rotorcraft longitudinal velocity component shall not be excessive (See Leaflet 602/3 para 2.5).

4.4 FLIGHT UNDER AUTOPILOT CONTROL

4.4.1 Where the Rotorcraft Specification calls for particular mission tasks to be carried out under automatic control either in order to achieve a repeatable degree of positional accuracy, or because the absence of adequate external reference cues precludes active pilot execution of the task:

(i) The required level of response and/or positional accuracy will be defined by the Rotorcraft Specification or the autopilot system specification.

(ii) The requirements of para 4.2 are still applicable.

5 MANOEUVRES IN THE FORWARD FLIGHT ENVELOPE

5.1 SYMMETRIC MANOEUVRES

5.1.1 The requirements of para 5.1 are mainly concerned with cross coupling effects during manoeuvres primarily initiated through the longitudinal and/or lifting control channels.

5.1.2 Lateral control displacements and forces required in order to maintain balanced flight during pull-up, push-over, accelerating and decelerating manoeuvres shall meet the following requirements:

(i) Be progressive in application during the course of the manoeuvre, not objectionably large and in harmony with other control inputs.

(ii) For a given symmetric manoeuvre, the lateral control inputs shall not be unduly sensitive to variations in rotorcraft loading or small variation in the degree of aggressiveness with which the manoeuvre is carried out.

5.1.3 In 'Aggressive' pull-up manoeuvres:

(i) There shall be no unexpectedly rapid increase in roll attitude or rate which is not preceded by unambiguous tactile cues to the pilot that the lifting system is approaching an operating state involving aerodynamic non-linearities and/or potentially critical loading conditions. (See also Leaflet 601/4 paras 2 and 3).

(ii) In the event of any indication of a breakaway in roll, as in (i), it shall be possible to correct this by an instinctive application of lateral control which is not excessive.

5.1.4 In 'Push-Over' Manoeuvres:

(i) With load factors approaching the low or negative design limits, the rotorcraft roll control power shall be such that a lateral control input to the right results in a rotorcraft response rolling to the right, and conversely for a left lateral input.

5.2 TURNING MANOEUVRES

5.2.1 The requirements of para 5.2 refer to banked turn manoeuvres, unless otherwise stated, initiated from steady, straight flight conditions. These requirements complement a number of those contained in para 2.3.

5.2.2 Active Flight Phases

(i) During entry to and recovery from banked turns, and also roll reversals, initiated by smooth continuous lateral control inputs, there shall be no objectionable hesitations in the changes of roll rate of attitude.

(ii) Having initiated a banked turn, the lateral control displacement or force required to maintain the steady bank angle and airspeed shall return close to the initial datum value.

(iii) For LEVEL 1 qualities any residual control displacement or force (prior to any retrimming) shall be to the right in right turning flight, conversely for left turning flight. It shall also not be sufficiently large to impair the precision of control in the longitudinal, or any other axis, effected by means of the same controller.

(iv) It shall be possible to establish a sustained banked turning flight path, in either direction, by application of a lateral control input only, without involving sideslipping in the opposite direction to the turn, or objectionable lateral out of balance forces at the pilot's location.

5.2.3 Autopilot controlled Turns. Where banked turn manoeuvres are executed under autopilot control:

(i) The pilot shall be aware of the impending manoeuvre, either by virtue of his having selected the mode, or some attention getting indicator.

(ii) The bank angle to be achieved shall be known to the pilot, either by virtue of his selection or some predetermined schedule.

(iii) The demanded bank angle shall be achieved smoothly ideally with a small single overshoot announcing arrival.

6 CONTROL MARGINS

6.1 There shall be adequate lateral control available in the rotorcraft to meet the general requirements of Chapter 600 para 12, with reference to Leaflet 600/9.

6.2 LOW SPEED ENVELOPE

6.2.1 There shall be adequate lateral control available in either direction to meet Chapter 600 para 12.3.1 when the rotorcraft is:-

(i) Loaded to the critical total weight with the extreme lateral, combined with any adverse longitudinal centre of gravity location.

(ii) In steady trimmed flight with the critical combination of relative wind speed and azimuth direction called for by Rotorcraft Specification.

6.2.2 In those cases where the critical flight condition in para 6.2.(ii) arises with the rotorcraft stationary relative to some external reference, there shall be adequate lateral control available to allow the pilot to fly the rotorcraft to that condition, from any other condition in the Service Flight Envelope, without exercising undue skill or anticipation in initiating the manoeuvre to avoid overshooting the external reference.

6.3 FORWARD FLIGHT ENVELOPE

6.3.1 There shall be adequate lateral control available in either direction to meet Chapter 600 para 12.3.1 when the rotorcraft is:-

(i) Loaded to the critical total weight with the extreme lateral, combined with any adverse longitudinal centre of gravity location.

(ii) In steady trimmed flight at any condition within the Forward Flight Envelope, including the range from autorotation to maximum power climb.

6.3.2 The requirements of para 6.3.1 shall also be met throughout the Sideslip Envelope.

6.4 AFCS TRIMMING

6.4.1 For those rotorcraft in which a series actuator trimming device is used to avoid the roll autostabiliser actuator system remaining offset from null under sustained flight conditions, that trimming device shall have sufficient authority to allow null offsets to be kept small to ensure:

(i) Adequate remaining actuator authority for stabilisation of the rotorcraft against external disturbances.

ii) Acceptable pilot intervention times in the event of an actuator runaway in the adverse sense from any such null offset.

6.4.2 The requirements of para 6.4.1 shall be met for the most adverse combinations of flight condition and loading, including centre of gravity shifts due to asymmetric release of stores, and system design tolerances.

LEAFLET 602/1

LATERAL FLIGHT HANDLING QUALITIES

ROLL RESPONSE CHARACTERISTICS

1 INTRODUCTION

1.1 This Leaflet provides quantitative guidance on the initial response to lateral control inputs and roll dynamic stability characteristics referred to in Chapter 602 para 2.

2 ROLL ATTITUDE

2.1 As in the case of rotorcraft pitch attitude, changes in roll attitude and its derivatives with respect to time are prime indicators of lateral dynamic stability and control response characteristics.

2.2 In the Low Speed Envelope, steady state roll attitude of the conventional single shaft driven main rotor configuration is uniquely determined by the lateral relative velocity component and lateral centre of gravity offset.

2.3 In Forward Flight a compromise can be drawn with aerodynamic sideslip (Leaflet 602/2 para 3) allowing a straight flight path to be maintained at zero roll attitude.

3 LONG TERM DYNAMIC STABILITY

3.1 RATIONALE

3.1.1 The requirements of Chapter 602 para 2.2 are expressed qualitatively, in line with the general rationale that rotorcraft handling characteristics should not prevent any specific mission accomplishment, whatever that implies.

3.1.2 Para 3.2 provides quantitative guidance applicable to general operation in Attentive and Passive flight phases. For those rotorcraft which are required to carry out specialised tasks demanding closer control over long term roll attitude behaviour, appropriately stricter limitations will be given in the Rotorcraft Specification.

3.1.3 No specific requirement is laid down for Active flight phases since the virtually continuous control applications inherent in these phases would render any demonstration of compliance impossible. Nevertheless for satisfactory operations in the active category long period stability characteristics should be reasonably damped, no worse than those described in para 3.2.2(iii).

3.1.4 The overall objective of the attitude limitations in para 3.2 is to contain long term drift, or the amplitude of any long period response modes. The roll rate criteria is intended to prevent noticeable changes in attitude occurring from time to time while the controls are unattended, as a result of automatic control system operation in any axis.

3.1.5 The quantitative motion criteria used in para 3.2.1 are virtually threshold levels and it is appropriate to use the same values for each flight phase, and also for LEVEL 1 and 2 handling qualities, where the differing time intervals over which the criteria can be met reflect the associated changes in pilot workload.

3.2 QUANTITATIVE GUIDANCE

3.2.1 Following release of the flying controls roll attitude should not vary by more than ± 1 deg, and roll rate should not exceed ± 3 deg/sec during the intervals given in paras 3.2.2(i), (ii), 3.2.3(i) and (ii).

3.2.2 Attentive Flight Phases

(i) LEVEL 1: 20 seconds

(ii) LEVEL 2: 10 seconds

(iii) LEVEL 3: Any oscillatory mode with a period greater than 5 sec but less than 10 sec should halve its amplitude in no more than 2 cycles. Oscillations with periods up to 20 sec should be damped, those with periods greater than 20 sec should not double amplitude in less than 20 sec. Any aperiodic mode should not double amplitude in less than 9 sec.

3.2.3 Passive Flight Phases

(i) LEVEL 1: 3 minutes

(ii) LEVEL 2: 1 minute

(iii) LEVEL 3: is not addressed here, as it is not compatible with passive phases of flight.

4 SHORT TERM RESPONSE CHARACTERISTICS

4.1 RATIONALE

4.1.1 For those operational phases away from ground restraints or obstacles, or not requiring precise positional control of the rotorcraft, short term lateral handling is dominated by roll response characteristics. This includes both the initial responsiveness to pilot control inputs, and dynamic stability characterised by the transient recovery from a disturbance, as dealt with in general terms by Chapter 600, paras 10.1 and 11.3.

4.1.2 Generalized criteria, identified in Leaflet 600/7 are here related to roll rate or roll attitude, depending on the particular flight phase or level of handling quality involved.

4.1.3 Quantitative data provided in para 4 is provisional, pending confirmation or the derivation of more meaningful data.

4.1.4 The time at which the initial response should be monitored for adequate responsiveness without oversensitivity (T_I in Fig. 1 of Leaflet 600/7) remains to be optimised in terms of practicality of timing and resolution between acceptable and unacceptable response characteristics. It is anticipated that 0.5 sec may be suitable and this is provisionally proposed for all cases.

4.2 ACTIVE FLIGHT PHASES

4.2.1 Continuous flying of the rotorcraft through the pilots' flying controls has been further categorised into Aggressive and Moderate manoeuvring tasks with the intention of specifying minimum requirements for each, where relevant.

4.2.2 The requirement in Chapter 602, para 2.3.2 (iii) is included to highlight the fact that, although an aggressive manoeuvre capability may be called for by the Rotorcraft Specification, this can only represent part of the total operational spectrum of the rotorcraft, and that rotorcraft designed to meet exacting requirements for aggressive manoeuvring must also be easy to fly in the gentle manoeuvre and near-steady conditions comprising the remainder of the service envelope.

4.2.3 Table 1 ascribes provisional numerical values to the initial response and stability criteria identified in Leaflet 600/7, paras 3 and Fig. 1.

4.2.4 Aggressive Manoeuvres

(i) For LEVEL 1 handling characteristics a pulse input through the lateral flying control should produce a roll rate type of rotorcraft response in accordance with the first column of Table 1.

(ii) Reduced handling qualities in terms of less responsiveness, greater sensitivity, larger overshoot and longer settling time are reflected in the wider parameter ranges quoted for LEVEL 2 compared with LEVEL 1.

(iii) LEVEL 3 is currently not addressed in Table 1. It would be inappropriate for the pilot to embark upon deliberately aggressive manoeuvres with the rotorcraft in a sufficiently degraded operating state that led to the workload in controlling the rotorcraft approaching the limits of the pilots' capability.

4.2.5 Moderate Manoeuvres

(i) For LEVEL 1 handling characteristics a lateral control input should also produce a roll rate type of rotorcraft response. As shown in Table 1 the peak response to the standard control input does not have to be as high as for aggressive tasks, but no distinction is made for the dynamic stability criteria.

(ii) If the lateral control input generates a roll attitude, rather than a roll rate, type of response the additional pilot anticipation required in accurately executing manoeuvres is likely to lead to LEVEL 2 or 3 qualities, depending on control sensitivity or dynamic stability characteristics expressed in terms of roll attitude as in Table 1.

(iii) For LEVEL 3 handling characteristics any short period oscillatory modes should be damped. Where flight under IFR is required, oscillations having a period of 5 sec or less should halve amplitude in less than 1 cycle, and those with a period greater than 5 sec in less than 2 cycles. For flight under VFR, oscillations with a period of 5 sec or less should halve amplitude in less than 2 cycles.

TABLE 1

SHORT TERM RESPONSE - ACTIVE FLIGHT PHASES

Man.Clas.	**Aggressive**			**Moderate**		
LEVEL	**1**	**2**	**3**	**1**	**2**	**3**
Response Parameter	Roll Rate	Roll Rate		Roll Rate	Roll Att.	Roll Att.
Peak Response	15-20°/sec	10-15°/sec		10-15°/sec	8-12°	6-15°
T_I(sec)	0.5	0.5	N/A	0.5	0.5	0.5
Y_1%	30-TBA	30-TBA		30-TBA	30-TBA	30-TBA
Y_2%	5	10		5	0	0
T_{30}(sec)	<1	<1		<1	<1.5	
T_{11}(sec)	-	1-2		-	1.5-3	
T_{01}(sec)	1.2	-		1.2	-	See Para 4.2.5 (iii)
X_1%	15	20	N/A	15	25	
T_{02}(sec)	>2	>2		>2	>2.5	
X_2%	10	15		10	15	
T_F(sec)	3	5		3	5	
X_F%	10	10		10	10	

4.3 ATTENTIVE FLIGHT PHASES

4.3.1 For the satisfactory execution of attentive flight phases the rotorcraft should have the stability characteristics associated with a roll attitude type of response. Following the standard pulse input through the lateral flying control, the roll attitude response should be in accordance with the criteria identified in Leaflet 600/7 and quantified in Table 2.

TABLE 2

SHORT TERM RESPONSE - ATTENTIVE FLIGHT PHASES

LEVEL	1	2	
Peak Response	6-10°	4-15°	
T_1(sec)	0.5	0.5	
Y_1%	30-TBA	30-TBA	
Y_2%	0	0	
T_{30}(sec)	<1	<1.5	
T_{11}(sec)	-	1.5-3	
T_{01}(sec)	1-2	-	
X_1%	10	15	
T_F(sec)	3	5	
X_F%	10	10	

4.4 PASSIVE FLIGHT PHASES

4.4.1 Pilot applied control inputs are not relevant to Passive flight phases, but short term dynamic stability characteristics are.

4.4.2 Rotorcraft roll attitude behaviour should be well stabilised against external disturbances (e.g,. atmospheric turbulence). Also the functioning of any autopilot modes (e.g., controlling rotorcraft heading or height) should not result in any untoward excursions in roll attitude or the setting up of any short period oscillatory coupling with roll attitude.

4.4.3 Pending the derivation of any alternative criteria, the transient recovery of roll attitude from a single disturbance should be in accordance with that for Attentive flight phases given in the second part of Table 2.

LEAFLET 602/2

LATERAL FLIGHT HANDLING QUALITIES

SIDESLIP BEHAVIOUR

1 INTRODUCTION

1.1 This Leaflet provides quantitative and descriptive guidance in support of Chapter 602 para 3.

2 AERODYNAMIC SIDESLIP

2.1 In general mission task elements do not call for the rotorcraft to be operated at specific sideslip angles, per se, in the same way that airspeed or bank angle for example can be defined. Nevertheless at any particular flight condition there will be an associated sideslip angle dictated by the flight mechanics of the situation, and the rotorcraft must have adequate handling qualities in this respect.

2.2 Aerodynamic sideslip angle indication is not generally part of the standard rotorcraft flight instrumentation fit. It may be necessary to fit specific test instrumentation to provide sideslip data for development work and demonstration of compliance.

3 SIDESLIP - ROLL ANGLE COMPROMISE

3.1 If the rotorcraft configuration is based on a shaft driven single main lifting rotor, with a tail rotor providing torque compensation, there is an element of asymmetry in the balance for forces and moments governing the equilibrium of the rotorcraft. Appropriate body-axis sideforce components can be derived from: gravity, when the rotorcraft is inclined in roll and inertia force due to linear lateral acceleration, together with inertial force due to rate of yaw, and airframe and rotor components due to aerodynamic sideslip, in the presence of forward speed.

3.1.1 In manoeuvring flight piloting technique determines the extent to which the various components described in para 3.1 are balanced against each other to achieve a particular net level of co-ordination in the manoeuvre. The handling quality rating will be affected by the ease with which the rotorcraft response characteristics allow the pilot to manipulate these various elements to achieve the required levels of coordination and aggressiveness.

3.1.2 In sustained forward flight conditions a zero roll option is generally preferred on the grounds of comfort and a readily identified flight condition reference datum, with the associated sideslip angle being accepted, often with little quantitative appreciation of its magnitude.

3.2 The rotorcraft design should minimise as far as is practicable any adverse effects of sideslip arising, as in para 3.2, by:-

3.2.1 Minimising the sideslip angles involved at critical flight conditions.

3.2.2 Minimising the susceptibility to sideslip of parasitic drag rises, buffeting, and any adverse effects on pressure sensitive devices providing flight conditions data to the pilot or automatic flight control system.

4 LATERAL STATIC STABILITY

4.1 Compliance with the requirements of Chapter 602 para 3.4.2 should be demonstrated over a range of flight conditions embracing as a minimum, those initial trimmed cruise, approach, climb and descent cases identified in Leaflet 601/2 paras 2.2.1 to 2.2.5.

4.2 The linearity called for by Chapter 602 para 3.4.3 should extend over a sideslip range on either side of the initial trimmed condition, of ± 15 deg or 75% of the Sideslip Envelope, whichever is the least. Over the remainder of the Sideslip Envelope, at any particular forward speed, any departure from the initial linearity should be progressive with no sharp discontinuities.

4.3 Steady, zero yaw rate sideslips as referred to in Chapter 602 para 3.4.4 should not require lateral control inputs greater than 50% of the margin available at the initial trimmed condition for LEVEL 1. For LEVEL 2 no more than 75% of the lateral control margin available at the initial trimmed condition should be used.

4.4 The objective of the requirement in Chapter 602 para 3.4.5 is to extend the concept of lateral static stability into the Low Speed Envelope. In this operational environment sideslip angle loses significance, and lateral velocity becomes the relevant parameter.

4.4.1 The lateral velocity referred to in para 4.4 is intended to be the relative air velocity component at the rotorcraft, and, in the absence of an installed low airspeed indicating system in the rotorcraft, specific test instrumentation or techniques may be required for the demonstration of compliance.

4.4.2 Since most operations in the Low Speed Envelope are likely to be carried out at low level, height variations associated with constant collective pitch or power at varying translational speeds are avoided by specifying constant height in Chapter 602 para 3.4.5(i).

4.5 The linearity called for by Chapter 602 para 3.4.6 should be achieved over a lateral speed range of at least ± 15 knots in the presence of fore and aft velocity components up to ± 10 knots. Over the remainder of the Low Speed Envelope any departure from this initial linearity should be progressive with no sharp discontinuities.

LEAFLET 602/3

LATERAL FLIGHT HANDLING QUALITIES

TRANSLATIONAL AND POSITIONAL CONTROL

1 INTRODUCTION

1.1 This Leaflet relates to the requirements in Chapter 602 para 4 particularly those concerned directly with lateral positioning of the rotorcraft in the Low Speed Envelope.

1.2 The general background to the basic task of translational and positioned control of rotorcraft is outlined in Leaflet 600/8.

2 VELOCITY CONTROL

2.1 The requirements in Chapter 602 para 4.2 are based on the assumption that the rotorcraft can be accurately positioned if the lateral velocity response to pilot input is well-conditioned. Ideally, following a step lateral control input, the lateral velocity should settle to a new value with the minimum delay or overshoot.

2.2 Achievement of these aims in a conventional configuration rotorcraft is likely to require a control augmentation facility, including translational parameter feedback. Insofar as this may be compatible with operation in other flight regimes, attention will need to be paid to means of engaging and disengaging such modes, including under transitory flight conditions as the Low Speed Envelope is entered and exited.

2.3 CONTROL SENSITIVITY

2.3.1 Under adverse operating conditions, precision of positional control will generally be more important than generating a relatively high final velocity. This is consistent with seeking a modest control sensitivity, combined with a short settling time.

2.3.2 Although this remains to be substantiated, it is proposed that for LEVEL 1 handling qualities a lateral control input equivalent to 10% of the full end to end range should produce a lateral velocity change in the range 2-5 knots.

2.3.3 A greater control sensitivity is likely to lead to a lower LEVEL being achieved, due to increased pilot workload in anticipating control adjustments necessary to position the rotorcraft.

2.4 SETTLING TIMES

2.4.1 In the context illustrated by Leaflet 600/8 Figs. 3 and 4 'settling time' is similar to the aperiodic time constant in Leaflet 601/2 para 3 in connection with airspeed trimming. However, because of the shorter elapsed time allowable in the present context, and the likely presence of other transient modes, decaying during the early response time, it will not be possible to satisfactorily treat the time history of lateral velocity as a simple exponential curve. Hence the requirements in Chapter 600 para 11.5.3 being expressed in terms of a finite time interval, which can be readily identified, after which the response should not vary significantly. This then allows the pilot to judge the adequacy of his control input for achieving his goal.

2.4.2 The settling time following a step input, as defined in Chapter 600 para 11.5.3 is provisionally set to 5 secs for LEVEL 1. It remains to be established that this time interval allows adequate precision of control over the rotorcraft, and that it can in fact be realistically achieved within the other restraints on system design and rotorcraft response behaviour.

2.4.3 Where the performance required in para 2.4.2 above cannot be achieved, it is anticipated that the level of handling quality will be degraded by:

(i) Increased pilot workload in having to compensate for control augmentation system shortcomings and apply a more adaptive control input than the simple step associated with LEVEL 1, to achieve a similar convergence towards a final velocity.

(ii) Increased time to get within the 10% final margin.

2.4.4 Meaningful values remain to be established for restrictions on the control input profile, and associated settling times.

(i) It is anticipated that the control overshoot as illustrated in Leaflet 600/8 Fig.4 could not realistically exceed 50% - 100% of the final control input.

(ii) It will also probably be relevant to specify that no more than two reversals of control be allowed after the initial application, before reaching the final control setting.

(iii) The settling time should still be defined from the point of finalising the control input, as this determines the pilot's ability to judge the effectiveness of his control activity.

2.5 CROSS COUPLING EFFECTS

2.5.1 Under the conditions defined in Chapter 602 para 4.3.4 the rotorcraft longitudinal velocity component should not exceed ± 10% of the lateral velocity change within the following time intervals from the finalisation of the discrete control input:

(i) LEVEL 1: Three times the LEVEL 1 lateral velocity settling time.

(ii) LEVEL 2: Twice the lateral velocity settling time appropriate to LEVEL 2.

CHAPTER 603

DIRECTIONAL FLIGHT HANDLING QUALITIES

1 INTRODUCTION

1.1 The requirements in Chapter 603 relate to those specific aspects of operation of the directional flying controls and associated rotorcraft response, about the yaw axis and along the lateral axis, not already dealt with in Chapter 600.

2 YAWING MOTION BEHAVIOUR (See also Leaflet 603/1, para 2)

2.1 CHANGES OF FLIGHT CONDITIONS

2.1.1 The rotorcraft shall be designed to minimise, as far as is practicable, the directional control adjustments necessary to maintain zero yaw rate when trimmed at any flight condition in the Service Flight Envelope as the power is varied over the full range applicable.

2.1.2 Over the range from minimum power speed to maximum normal operating speed, the directional control adjustments necessary to maintain steady straight and level flight shall also be minimised.

2.2 LONG TERM DIRECTIONAL DYNAMIC STABILITY

2.2.1 The requirements of para 2.2 refer to the near-steady state conditions prevailing in the Attentive and Passive flight phases with flying control settings retained in position as described in Chapter 600 para 9.2.2. Unless otherwise stated steady atmospheric conditions are assumed. (See Leaflet 603/1 para 3.1).

2.2.2 Attentive Flight Phases. The rotorcraft shall not deviate unacceptably in heading or sideslip from a trimmed flight condition if the flying controls are left unattended for short periods of time. (See Leaflet 603/1 para 3.2.2).

2.2.3 Passive Flight Phases. The rotorcraft shall not deviate unacceptably in heading or sideslip from a trimmed flight condition if the flying controls are left unattended for protracted periods of time. (See Leaflet 603/1 para 3.2.3).

2.3 SHORT TERM DIRECTIONAL RESPONSE CHARACTERISTICS

2.3.1 The requirements of para 2.3 relate primarily to the Active Flight Phases, but also have some relevance to the Attentive and Passive Phases. (See Leaflet 603/1 para 4.1).

2.3.2 Active Flight Phases

(i) Aggressive Manoeuvring Tasks. Yaw rate response to a directional control input shall be in accordance with Leaflet 603/1 para 4.2.4.

(ii) Moderate Manoeuvring Tasks. Yaw rate response to a directional control input shall be in accordance with Leaflet 603/1 para 4.2.5.

(iii) For those rotorcraft which have response characteristics enabling aggressive manoeuvring tasks to be carried out at LEVEL 1, the pilot shall also have no difficulty (e.g., due to any oversensitivity) in the precise execution of gentle manoeuvres or establishing steady flight conditions when required.

(iv) The initial yaw rate responses to discrete control inputs in the longitudinal and lifting or power available channels shall not be unduly large. Yaw rate responses following upon lateral control inputs shall not be excessive, or in the opposite sense to the lateral input. (See Leaflet 603/1 para 4.2.2 and 4.2.6).

2.3.3 Attentive Flight Phases. Yaw response characteristics shall be in accordance with Leaflet 603/1 para 4.3.

2.3.4 Passive Flight Phases. The operation of any autopilot modes available in the rotorcraft shall not result in any unacceptable short term directional response characteristics. (See Leaflet 603/1 para 4.4) .

2.4 HEADING TRIM

2.4.1 When an autopilot heading hold mode is engaged a means shall be provided for the pilot to readily make small, precise adjustments to the datum heading. (See Leaflet 603/1 para 4.4.4).

3 SIDESLIP BEHAVIOUR

3.1 The requirements of this paragraph relate to the part played by rotorcraft directional characteristics in sideslipping flight throughout the Forward Flight Envelope.

3.2 CHANGE OF OPERATING CONDITIONS

3.2.1 Failure State. In the case of rotorcraft configurations employing a rotor to balance the torque reaction associated with driving rotating lifting surfaces, and where a single failure can result in the loss of a controlled input to the pitch change mechanism of that rotor while it is still being driven, the following requirements are applicable:

(i) The contractor shall define the pitch angle adopted by the blades and subsequent rotor behaviour during any associated recommended flight procedures within the Service Flight Envelope.

3.2.2 Under the conditions of continued operation following failure of the torque reaction compensating device (Chapter 602 para 3.3.1) the following requirements are applicable:

(i) Any aerodynamic surface configuration introduced in order to achieve this continued operation shall not result in unduly adverse effects during normal operation due to blockage of the torque reaction compensating device, or the degrading of crosswind performance and handling in critical flight conditions within the Low Speed Envelope.

(ii) In reaching any design compromise associated with para 3.2.2 (i), note shall be taken of Chapter 600 para 2.3.

3.3 DIRECTIONAL STATIC STABILITY

3.3.1 A directional control input to the right shall be required to maintain an increased steady, zero yaw rate, sideslip to the left. Conversely a left directional control input shall be required for increased sideslip to the right.

3.3.2 The requirements in para 3.3.1 shall be met with the sideslip initiated from any trimmed flight condition within the Forward Flight Envelope:

(i) Above the minimum power speed or maximum speed for IFR (if defined) whichever is the lower, up to maximum normal operating speed.

(ii) With the collective pitch or power control, whichever is appropriate fixed at the trimmed setting.

(iii) See also Leaflet 603/2 para 2.1.

3.3.3 For LEVEL 1 and 2 qualities, with reference to para 3.3.1, the variations of directional control input with sideslip angle shall be substantially linear (See Leaflet 603/2 para 2.2).

3.3.4 From trimmed conditions in the Low Speed Envelope a directional control input to the left shall be required to initiate and maintain an increased lateral velocity to the right. Conversely for an increased lateral speed to the left. These requirements shall be met:

(i) With the rotorcraft height, heading and fore and aft speed maintained substantially constant.

(ii) See also Leaflet 603/2 para 2.3.

3.3.5 For LEVEL 1 and 2 qualities, with reference to para 3.3.4 the variations of directional control input with lateral velocity shall be substantially linear (See Leaflet 603/2 para 2.4).

4 OPERATIONS IN THE LOW SPEED ENVELOPE

4.1 The requirements of paragraph 4 relate primarily to low level, low speed manoeuvring during Active Flight Phases. The major requirements centre around the precision of heading control, through the directional flying controls, while manoeuvring and positioning relative to external constraints. These requirements shall be met under all rotorcraft loading conditions and the most adverse combinations of relative airspeed and direction covered by the Low Speed Envelope.

4.2 HEADING RETENTION

4.2.1 Near Vertical Manoeuvring. In those rotorcraft which are required to execute near vertical manoeuvres involving abrupt power changes and excursions in normal load factor, about an initial trimmed flight condition, it shall be possible to maintain the required heading within limits given by the Rotorcraft Specification, without undue skill or directional activity on the part of the pilot (Leaflet 603/3, para 2.1).

4.2.2 Translational Manoeuvring. While executing longitudinal or lateral translational manoeuvres, including acceleration or deceleration, it shall be possible to maintain the flight path along the required track, and the rotorcraft on the required heading, without undue skill or directional control activity on the part of the pilot. This requirement shall be met while power is adjusted to maintain substantially constant height, or increased to the appropriate maximum rating, whichever is more critical (Leaflet 603/3 para 2.2).

4.3 HEADING CHANGES

4.3.1 Spot Turns

(i) For those roles or mission phases in which the rotorcraft has to be positioned accurately within a confined space, it shall be possible to make heading changes in either direction while maintaining the axis about which the rotorcraft rotates fixed within limits relative to an external reference point, without undue skill or activity on the part of the pilot in any flying control channel.

(ii) In executing the spot turns of para 4.3.1(i) it shall be possible to contain the variations in yaw rate within the limits given in Leaflet 603/3 para 3.2(i).

(iii) It shall be possible to stop the rotation at a predetermined rotorcraft heading, without the overshoot exceeding the criteria given in Leaflet 603/3 para 3.2 (ii).

(iv) The requirement of para 4.3.1 shall be met under the most adverse wind conditions contained by the Low Speed Envelope.

4.3.2 Turning Translational Flight

(i) The directional control sensitivity shall be sufficiently well harmonised with the lifting/power and lateral control channels that, where mission tasks require rapid roll reversals to be flown at substantially constant height the anticipated curvature of the flight path can be achieved within acceptable margins of track error, and without unacceptably high lateral out of balance forces at the pilots position.

5 MANOEUVRES IN FORWARD FLIGHT ENVELOPE

5.1 SYMMETRIC MANOEUVRES

5.1.1 The requirements of para 5.1 are mainly concerned with cross coupling effects during manoeuvres initiated through the longitudinal and/or lifting control channels.

5.1.2 Directional control displacements and forces required in order to maintain balanced flight during pull-up, push-over, accelerating and decelerating manoeuvres shall meet the following requirements:

(i) Be progressive in application during the course of the manoeuvre, not objectionably large and in harmony with other control inputs.

(ii) For a given symmetric manoeuvre, the directional control inputs shall not he unduly sensitive to variations in rotorcraft loading or small variation in the degree of aggressiveness with which the manoeuvre is carried out.

5.1.3 In 'Aggressive' pull-up manoeuvres:

(i) There shall be no unexpectedly rapid increase in yaw rate, in either direction, due to the airframe or the lifting system approaching operation states involving aerodynamic non-linearities, which is not preceded by unambiguous tactile cues to the pilot.

(ii) In the event of any indication of a breakaway in yaw as in (i) above, it shall be possible to correct this by instinctive application of directional control.

5.1.4 In "push-over" manoeuvres, with load factors approaching the low or negative design limits, the pilot shall have no difficulty in applying appropriate directional control inputs, or avoiding making inadvertent ones. This requirement is particularly relevant to foot contact with pedal type directional controls.

5.2 TURNING MANOEUVRES

5.2.1 The requirements of para 5.2 refer to banked turn manoeuvres, unless otherwise stated, initiated from steady straight flight conditions. These requirements complement a number of those contained in para 2.3.

5.2.2 Active Flight Phases

(i) The pilot shall have no difficulty in coordinating banked turn manoeuvres initiated by smooth continuous lateral control inputs in either direction, by means of instinctive directional control inputs with appropriate adjustments to the other flying controls to maintain the required rotorcraft height and speed.

(ii) The requirement (i) above shall also be met throughout the power range, from autorotation to maximum power climb.

(iii) Care shall be taken in AFCS design to avoid any possible adverse cross coupling effects in steeply banked turns (such as motion about the rotorcraft yaw body axis being detected by a vertical gyro and used inappropriately in the pitch channel).

5.2.3 Autopilot Controlled Turns

(i) When automatic turn coordination is employed to balance turns initiated through pilot applied lateral control inputs, there shall be no unacceptable irregularities in rotorcraft response due to passage through small roll angle thresholds etc, and transient lateral out of balance forces shall not be unduly sensitive to the rate of roll initiation.

(ii) When navigational turn and heading acquire modes are employed in addition to the requirements in Chapter 602 para 5.2.3, the turn shall be reasonably well coordinated and the rotorcraft shall roll out on the demanded new heading with minimal overshoot.

6 CONTROL MARGINS

6.1 There shall be adequate directional control available in the rotorcraft to meet the general requirements of Chapter 600 para 12 with reference to Leaflet 600/9.

6.2 LOW SPEED ENVELOPE

6.2.1 There shall be adequate directional control available to meet Chapter 600 para 12.13.1 when the rotorcraft is:

(i) Operated in the most adverse combination of rotorcraft configuration, weight and centre of gravity location.

(ii) In steady trimmed flight with the critical combination of relative wind speed and azimuth direction called for by the Rotorcraft Specification.

6.2.2 In those cases where the critical flight condition in para 6.2.1(ii) arises with the rotorcraft stationary relative to some external reference, there shall be adequate directional control available to allow the pilot to fly the rotorcraft to that condition, from any other condition in the Service Flight Envelope, without exercising undue skill or anticipation in initiating the manoeuvre to avoid overshooting the external reference.

6.3 FORWARD FLIGHT ENVELOPE

6.3.1 There shall be adequate directional control available in either direction to meet Chapter 600 para 12.3.1 when the rotorcraft is:

(i) Operated in the most adverse combination of rotorcraft configuration, weight and centre of gravity location.

(ii) In steady trimmed flight at any condition within the Forward Flight Envelope, including the range from autorotation to maximum power climb, and throughout the Sideslip Envelope.

6.4 OPERATIONS WITH AFCS

6.4.1 For those rotorcraft in which the yaw channel autostabiliser series actuation system may be offset during flight conditions of sustained rate of yaw, there shall be sufficient basic directional control available to accommodate this offset and still meet the requirements of Chapter 600 para 12.3.1 throughout the power range from maximum continuous to autorotation, in the presence of rates of yaw up to 10 deg/sec in the adverse sense.

LEAFLET 603/1

DIRECTIONAL FLIGHT HANDLING QUALITIES

DIRECTIONAL RESPONSE CHARACTERISTICS

1 INTRODUCTION

1.1 This Leaflet provides quantitative guidance on the initial response to directional control inputs and yaw dynamic stability characteristics referred to in Chapter 603, para 2.

2 YAWING MOMENT BALANCE

2.1 The requirements of Chapter 603, para 2.1 mainly relate to quasi-steady cross coupling and interference effects, rather than direct manoeuvring issues, with the objectives of:

2.1.1 Reducing pilot workload in the directional channel in compensating for variations in lifting rotor torque reaction and the effects of the localized flow at rotors and aerodynamic surfaces contributing directly to the rotorcraft yawing moment balance, associated with changes in rates of climb/descent and airspeed.

2.1.2 Particularly in the case of conventional pedal type directional controls, reducing pilot discomfort in forward flight conditions due to prolonged uneven positioning of the pedals.

2.2 Even when automatic flight control systems are involved in order to relieve pilot workload, it is highly desirable that the basic rotorcraft design should minimise the need for these compensatory directional control inputs. This should then alleviate the design conflict between the needs for authority, speed of operation and safety considerations in the event of failures, for the automatic systems' actuation subsystem(s).

3 LONG TERM DYNAMIC STABILITY

3.1 RATIONALE

3.1.1 The requirements of Chapter 603, para 2.2 are expressed qualitatively, in line with the general rationale that rotorcraft handling characteristics should not prevent any specific mission accomplishment, whatever that implies.

3.1.2 Para 3.2 provides quantitative guidance applicable to general operation in Attentive and Passive flight phases. For those rotorcraft which are required to carry out specialised tracking or aiming tasks demanding closer control over long term heading behaviour, appropriately stricter limitations will be given in the Rotorcraft Specification.

3.1.3 No specific requirement is laid down for Active flight phases since the virtually continuous control applications inherent in these phases will render any demonstration of compliance impossible. Nevertheless for satisfactory operations in the active category long period stability characteristics should be reasonably damped, no worse than those described in para 3.2.2(iii).

3.1.4 The overall objective of the heading limitations in para 3.2 is to contain long term drift, or the amplitude of any long period response modes. The yaw rate criteria is intended to prevent noticeable directional changes occurring from time to time while the controls are unattended, as a result of automatic control system operation in any axis.

3.1.5 The quantitative motion criteria used in para 3.2.1 are virtually threshold levels and it is appropriate to use the same values for each flight phase, and also for LEVEL 1 and 2 handling qualities, where the differing time intervals over which the criteria can be met reflect the associated changes in pilot workload.

3.2 QUANTITATIVE GUIDANCE

3.2.1 Following release of the flying controls, heading should not vary by more than ± 2 deg, and yaw rate should not exceed ± 3 deg/sec during the intervals given in paras 3.2.2(i), (ii) and 3.2.3(i), (ii).

3.2.2 Attentive Flight Phases

(i) LEVEL 1: 20 seconds
(ii) LEVEL 2: 10 seconds
(iii) LEVEL 3: Any oscillatory mode with a period greater than 5 sec should halve its amplitude in no more than 2 cycles. Oscillations with periods up to 20 secs should be damped, those with periods greater than 20 sec should not double amplitude in less than 20 sec. Any aperiodic mode should not double amplitude in less than 9 sec.

3.2.3 Passive Flight Phases

(i) LEVEL 1: 3 minutes
(ii) LEVEL 2: 1 minute
(iii) LEVEL 3: is not addressed here, as it is compatible with passive phases of flight.

4 SHORT TERM RESPONSE CHARACTERISTICS

4.1 RATIONALE

4.1.1 Satisfactory short term directional handling characteristics depend upon good yaw rate behaviour in virtually all operational flight phases. This includes not only the initial responsiveness to deliberate directional control inputs, and dynamic stability characterised by the transient recovery from a disturbance, (Chapter 600 paras 10.1 and 11.3), but also the suppression of unintentional yaw motion due to external disturbances or manoeuvring in other rotorcraft axes.

4.1.2 Generalized criteria identified in Leaflet 600/7 are here related to yaw rate and heading behaviour, depending on the degree of pilot attentiveness dictated by the mission phase.

4.1.3 Quantitative data provided in para 4 is provisional, pending confirmation or the derivation of more meaningful data.

4.1.4 The time at which the initial response should be monitored for adequate responsiveness without oversensitivity (T_I in Fig 1 of Leaflet 600/7) remains to be optimised in terms of practicality of timing and resolution between acceptable and unacceptable response characteristics. It is anticipated that 0.5 sec may be suitable, and this is provisionally proposed for all cases.

4.2 ACTIVE FLIGHT PHASES

4.2.1 The requirements in Chapter 603 para 2.3.2(i), (ii) and (iii) relate to the rotorcraft directional response characteristics necessary for the execution of manoeuvres fundamentally aimed at changing the heading of the rotorcraft.

4.2.2 Manoeuvres initiated through the other control channels, with various degrees of aggressiveness, are also likely to induce yawing disturbances to the rotorcraft. The requirement in Chapter 603 para 2.3.2 (iv) seeks to contain these to minimise the directional control inputs necessary to either maintain the original rotorcraft heading, or coordinate a turning manoeuvre, whichever is appropriate (see also para 4.2.6).

4.2.3 Table 1 ascribes provisional numerical values to the initial response and stability criteria identified in Leaflet 600/7 paras 3 and 6, and Fig 1.

4.2.4 Aggressive Manoeuvres

(i) For LEVEL 1 handling characteristics a pulse input through the directional flying control should produce a yaw rate type of rotorcraft response in accordance with the first column of Table 1.

(ii) Reduced handling qualities in terms of less responsiveness, greater sensitivity, larger overshoot and longer setting time are reflected in the wider parameter ranges quoted for LEVEL 2 compared with LEVEL 1.

(iii) LEVEL 3 is currently not addressed in Table 1. It would be inappropriate for the pilot to embark upon deliberately aggressive manoeuvres with the rotorcraft in a sufficiently degraded operating state that led to the workload in controlling the rotorcraft approaching the limits of the pilots' capability.

4.2.5 Moderate Manoeuvres

(i) For LEVEL 1 and 2 handling characteristics a directional control input should also produce a yaw rate type of rotorcraft response. As shown in Table 1, the peak responses to the standard control input do not have to be as high as for aggressive tasks, but no distinction is made for the dynamic stability criteria.

(ii) For LEVEL 3 handling characteristics any short period oscillatory modes should be damped. Where flight under IFR is required, oscillations having a period of 5 sec or less should halve amplitude in less than 1 cycle, and those with a period greater than 5 sec, in less than 2 cycles. For flight under VFR, oscillations with a period of 5 sec or less should halve amplitude in less than 2 cycles.

4.2.6 Secondary Yaw Response

(i) The rotorcraft should be designed to minimise the basic secondary yaw response referred to in Chapter 603 para 2.3.2 (iv). As a handling criterion this consideration should be quantified in terms of a maximum corrective directional control input allowable, (irrespective of the control power of the directional control device), rather than a limit on the absolute level of the yaw rate disturbance.

(ii) Suitable directional control input limits remain to be determined.

TABLE 1

SHORT TERM RESPONSE - ACTIVE FLIGHT PHASES

Man.Clas.	**Aggressive**			**Moderate**		
LEVEL	**1**	**2**	**3**	**1**	**2**	**3**
Response Parameter	Yaw Rate	Yaw Rate		Yaw Rate	Yaw Att.	Yaw Att.
Peak Response	10-15°/sec	5-20°/sec		5-10°/sec	TBA	TBA
T_I(sec)	0.5	0.5		0.5	0.5	0.5
Y_1	30-TBA	30-TBA	N/A	30-TBA	30-TBA	30-TBA
Y_2	5	10		5	0	0
T_{30}(sec)	<1	<1		<1	<1	
T_{11}(sec)	-	1-2		-	1-2	See
T_{01}(sec)	1-2	-		1-2	-	para
X_1%	15	20	N/A	15	20	4.2.5 (ii)
T_{02}(sec)	>2	>2		>2	>2	
X_2%	10	15		10	15	
T_F(sec)	3	5		3	5	
X_F%	10	10		10	10	

4.3 Attentive Flight Phases

4.3.1 For any adjustments in yaw to flight conditions following a period with flying controls unattended, the initial response of the rotorcraft to a directional control input should be similar to that described for Moderate Manoeuvring flight in para 4.2.5.

4.3.2 Any flight control system augmentation introduced to meet para 4.3.1 must also allow the longer term dynamic stability characteristics described in para 3.2.2 to be achieved.

4.4 PASSIVE FLIGHT PHASES

4.4.1 Pilot-applied flying control inputs are not relevant to Passive flight phases, but short term dynamic stability characteristics are.

4.4.2 The functioning of any autopilot modes (e.g., controlling rotorcraft speed or height) under disturbed atmospheric conditions, or during automatically controlled manoeuvres, should not result in any unexpectedly large or rapid excursions in yaw or promote any short period oscillatory yawing motion.

4.4.3 The transient recovery in yaw rate from an initial peak offset resulting from a single discrete disturbance, should be in accordance with Moderate Manoeuvring characteristics shown in the second part of Table 1 in para 4.2.5.

4.4.4 The heading hold trim facility should allow the pilot to select a specific heading change, resolvable down to 1 deg for LEVEL 1. The new heading should be established with no more than one overshoot. The allowable overshoot and time taken to reach the new heading will be defined in the Rotorcraft or Automatic Flight Control System Specifications.

LEAFLET 603/2

DIRECTIONAL FLIGHT HANDLING QUALITIES

SIDESLIP BEHAVIOUR

1 INTRODUCTION

1.1 This Leaflet provides quantitative and descriptive guidance in support of Chapter 603 para 3.

2 DIRECTIONAL STATIC STABILITY

2.1 Compliance with the requirements of Chapter 603 para 3.3.2 should be demonstrated in conjunction with the lateral static stability referenced in Leaflet 602/2 para 4.1. This should embrace a range of flight conditions including, as a minimum, the initial trimmed cruise, approach, climb and descent cases identified in Leaflet 601/2 paras 2.2.2 to 2.2.5.

2.2 The linearity called for by Chapter 603 para 3.3.3 should extend over a sideslip range on either side of the initial trimmed condition of ± 15 deg. or 75% of the Sideslip Envelope, whichever is the least. Over the remainder of the Sideslip Envelope, at any particular forward speed, any departure from the initial linearity should be progressive with no sharp discontinuities.

2.3 The objective of the requirement in Chapter 603 para 3.3.4 is to extend the concept of directional static stability into the Low Speed Envelope. In this operational environment sideslip angle loses significance, and lateral velocity becomes the relevant parameter.

2.3.1 The lateral velocity referred to in para 2.3 is intended to be the relative air velocity component at the rotorcraft, and, in the absence of an installed low airspeed indicating system in the rotorcraft, specific test instrumentation or techniques may be required for the demonstration of compliance.

2.3.2 Since most operations in the Low Speed Envelope are likely to be carried out at low level, height variations associated with constant collective pitch or power at varying translational speeds are avoided by specifying constant height in Chapter 603 para 3.3.4(i).

2.4 The linearity called for by Chapter 603 para 3.3.5 should be achieved over a lateral speed range of at least ± 15 knots in the presence of fore and aft velocity components up to ± 10 knots. Over the remainder of the Low Speed Envelope any departure from this initial linearity should be progressive with no sharp discontinuities.

LEAFLET 603/3

DIRECTIONAL FLIGHT HANDLING QUALITIES

DIRECTIONAL MANOEUVRES

1 INTRODUCTION

1.1 This Leaflet provides quantitative guidance and support to the requirements of Chapter 603 para 4.

2 HEADING RETENTION

2.1 VERTICAL MANOEUVRING

2.1.1 For tactical manoeuvring involving precise height changes, and a high mission task work load, the pilot should have to pay little attention to maintaining rotorcraft heading.

2.1.2 For LEVEL 1 handling qualities the heading holding requirements of the Rotorcraft Specification should be met without any directional control input by the pilot, but by means of appropriately designed control mixing between the lifting/power and directional channels, and automatic stability and control augmentation.

2.1.3 Lower Levels of handling quality will be associated with the additional pilot workload in maintaining the final heading with sufficient accuracy and acceptable transient overswings.

2.1.4 Meaningful quantitative data remains to be established, but it is anticipated that for:

(i) Aggressive Manoeuvres. Abrupt power changes of the order ± 25% maximum continuous, and load factors ± 0.25 g may need to be considered. Under these conditions a prompt return to within ± ½ deg of the initial heading with a transient excursion less than ± 5 deg would probably represent a LEVEL 1 target. LEVEL 2 will accept, say, a prompt return within ± 1 deg with ± 10 deg transient.

(ii) Moderate Manoeuvres. A similar degree of heading retention will be required for power and normal workload factor excursions half of those referred to in para (i) above.

2.2 TRANSLATIONAL MANOEUVRING

2.2.1 In the context of translational manoeuvring in a critical low level operating environment, maintenance of the appropriate rotorcraft track, relative to potential external obstructions, is the relevant consideration.

2.2.2 Insofar as track depends on the vector sum of longitudinal and lateral velocity components, combined with rotorcraft heading, monitoring and controlling it calls for complex processing of groundspeed component and heading sensor information in the absence of adequate outside visual reference cues.

2.3.3 Suitable criteria for limiting track errors remain to be established.

3 SPOT TURNS

3.1 The maximum rate of heading change to be achieved and, where relevant, limits on the positional accuracy necessary for particular mission tasks, will be defined by the Rotorcraft Specification.

3.2 Unless stated otherwise in the Rotorcraft Specification the following, criteria are given for general guidance:

(i) Yaw rate excursions about the mean rate should not exceed ± 33% of the mean rate, or ± 15 deg/sec, whichever is the least.

(ii) Rotorcraft behaviour in decelerating from a steady rate of yaw should be sufficiently predictable to allow the pilot to anticipate the required directional control application in order to stop at a predetermined heading without the overshoot exceeding 6 deg from yaw rates up to 30 deg/sec, 12 deg from yaw rates up to 45 deg/sec and 24 deg from yaw rates up to 60 deg/sec.

CHAPTER 604

AUTOMATIC FLIGHT CONTROL SYSTEMS

1 INTRODUCTION

1.1 This Chapter contains requirements governing airworthiness and essential flying qualities of all rotorcraft when operating under the control of, or assisted by, an automatic flight control system (AFCS) which is designed to:

(i) enhance the stability of the rotorcraft (autostabilisation modes) and/or,

(ii) maintain a particular flight condition or execute a manoeuvre (autopilot modes).

1.2 The Chapter is divided for convenience into four sections:

Section 1 Design for flight under automatic or partially automatic control.

Section 2 Definitions

Section 3 Safety Assessment, Analyses, and Tests.

Section 4 Flight tests with failure simulation.

1.3 The strength requirements relevant to these installations are given in Chapter 204.

SECTION 1 - DESIGN FOR FLIGHT UNDER AUTOMATIC OR PARTIALLY AUTOMATIC CONTROL

2 GENERAL REQUIREMENTS

2.1 The Rotorcraft Specification will state whether provision shall be made for the fitting of an approved type of AFCS and, if so, it will define the extent to which the AFCS is required to control the rotorcraft throughout the flight envelope, including configuration changes.

2.2 The Rotorcraft Specification may indicate the type of Safety Assessment of the AFCS and Rotorcraft Systems required (see para 6 below and Section 3).

2.3 The Rotorcraft Specification may also define certain critical modes of flight segments in the particular mode, for which a particular Intervention Time is specified. In general the Design Aim should be to achieve the largest practical Intervention Time in each mode or flight segment.

2.4 Section 4 gives values for Rotorcraft Response Times and Pilot Response Times which are dependent on the degree of Pilot Involvement in each flight segment. These requirements shall be met unless agreed otherwise with the Rotorcraft Project Director.

2.5 The effects of the AFCS shall be taken into account when assessing fatigue lives.

3 PERFORMANCE AND DESIGN

3.1 AFCS performance requirements will be stated in the Rotorcraft Specification.

3.2 The flying qualities of the rotorcraft with the AFCS engaged correctly in the appropriate flight condition shall be acceptable to the aircrew and suitable for the mission. They should have a pilot rating of level 1, (see Volume 1, Leaflet 600/1). In particular any residual oscillation linear or angular, following a displacement or correction of a displacement should also have a level 1 rating.

3.3 If component tolerances and their variation with ambient temperature could have a significant effect on the performance of the AFCS (including those requirements stated in the Rotorcraft Specification) or on the severity of the consequences of a failure, then these effects shall be taken into account in ensuring compliance with the requirements of this Chapter and those of the Rotorcraft Specification.

3.4 Consideration shall be given to the provision of:

(i) a means of testing the AFCS system in the Rotorcraft in Service before take-off, and

(ii) indication of actuator positions or of signals controlling the actuators.

4 ENGAGEMENT AND DISENGAGEMENT

4.1 It shall be possible to engage and disengage the AFCS in flight and on the ground.

4.2 Means for engagement and disengagement shall be provided in accordance with the requirements of Chapter 107.

4.3 On engagement the AFCS shall come into operation smoothly and shall promote no Dangerous Effect (see Section 2 para 11.3).

4.4 Complete disengagement of the AFCS shall be possible, quickly, safely and positively at any time independently of all other primary services unless it can be shown that the consequences of remaining engaged in all circumstances are more desirable.

4.5 It shall be possible at all times for the pilot to override the AFCS by the use of his flying controls without the prior need to disengage all or part of the AFCS. The force/displacement characteristics of the controls when overridden shall not give rise to any Dangerous Effect.

4.6 Where it is possible to engage, in addition to the autostabiliser, one or more modes or channels which are dependent on signals from other systems, provision shall be made for the disengagement of these modes and channels without affecting the autostabiliser. The means for so doing shall be in accordance with the requirements of Chapter 107. Consideration shall also be given to the need for disengagement of any lane which has failed.

4.7 Changeover from any one mode to any other mode shall be smooth and shall not cause a Dangerous Effect (see Section 2 para 11.3).

5 CONTROLLABILITY

5.1 The normally functioning AFCS shall not cause a Dangerous Effect when the most adverse conditions occur after:

(i) failure to release properly any external stores,

(ii) failure of any one engine in a multi-engined rotorcraft.

5.2 If the engine of a single-engined Rotorcraft fails in flight, the AFCS shall not cause a Dangerous Effect.

5.3 The time for which the rotorcraft remains controllable following the recognition of a failure (whether or not the pilot takes action to disengage the AFCS or the mode engaged shall be greater than the Pilot Response Time (see Section 4 para 31). Consideration shall be given to increasing the Intervention Time whenever this would be beneficial. The design aim should be to achieve 10 seconds wherever possible.

5.4 Particular attention shall be paid to the means by which the pilot recognises an AFCS failure resulting in oscillatory rotorcraft behaviour.

5.5 The pilot must have the means to ascertain the required recovery action and the necessary authority remaining in the flying controls to achieve a recovery whether all or part of the AFCS is still engaged or not. However, it must be assumed that the pilot will not normally disengage all or part of the AFCS until he has stabilised the rotorcraft within any limitations specified for flight with the appropriate parts of the AFCS inoperative. The recovery must be possible within values of speed, normal acceleration, sideslip angle, and rotor speed, specified by the designer. The flight envelope thus specified, may, if the designer wishes, be larger than that cleared for normal use. Nevertheless, it must be an envelope that can be used safely for an adequate number of flight test demonstrations.

5.6 Following recovery from a failure and consequent disengagement of the AFCS system or part of it, the pilot shall have sufficient control, within limitations, appropriate to the new condition of the AFCS:

(i) to provide control of heading, attitude, speed and height, and

(ii) to fly on instruments and carry out instrument let-down procedures including an overshoot, and

(iii) make a successful landing by day or by night.

5.7 Any requirement more demanding, than that of para 5.6 (such as completion of all or part of the mission) will be stated in the Rotorcraft Specification. Inability to meet any of the requirements of para 5.6 shall be assessed as a catastrophe (see Section 3 para 20).

6 SAFETY ASSESSMENT (see also Section 3)

6.1 At an early stage in the design a Failure and Effects Analysis of the AFCS and its related systems shall be submitted. A study shall also be submitted of the need for, and cost of, a full Safety Assessment of all Systems, including the AFCS, in accordance with current civil requirements. The study should state the applicable date and status of the civil requirements proposed together with any derogations or alternatives considered necessary.

6.2 At a later stage the designer shall either submit the full Safety Assessment or issue an updated Failure and Effects Analysis as required by the Rotorcraft Project Director.

6.3 At the final stage the designer may be required to re-submit the Safety Assessment or the Failure and Effects Analysis based on the results of flight trials.

7 FAILURE INDICATION

7.1 The Failure and Effects Analysis shall show the manner in which the designer intends the pilot to be cued to the existence of each failure. The cue may take the form of an adequate tactile, audio, or visual warning, the latter being in the central warning facility. Alternatively the designer may elect to regard the Rotorcraft's response as the cue that stimulates the pilot to recognise the failure. Whichever cue is selected the designer shall ensure that after the recovery has been made the pilot then has the means to identify the axis and lane in which the failure has occurred.

7.2 If the effect of a failure is such that the crew needs to be aware of it at any time during the mission and it will not be normally apparent then appropriate indication of the failure shall be provided. Artificial indication shall be provided where normal indication would be inadequate.

7.3 Where the operation of an automatic safety device prevents a failure being apparent to the crew, indication of the operation of the safety device shall be provided unless the requirements of Section 3 para 20.2 are still met after both the failure and the operation of the device.

7.4 Consideration shall also be given to the provision of a test facility to check whether the device is operating correctly.

8 TESTING

8.1 GROUND TESTS

8.1.1 The aim of ground testing is to develop the AFCS to a satisfactory state of performance, safety, and reliability, to a suitable time scale, and at the minimum cost compatible with effective results. The extent of such testing will vary, depending on the complexity of the system and on the evidence available from subsidiary rigs and from experience on other rotorcraft, and shall be discussed and agreed with the Rotorcraft Project Director.

8.1.2 Any new AFCS shall be submitted to ground tests for flight clearance on the general lines given in Volume 1 Leaflet 729/3 para 2 before the first flight of the installation.

8.1.3 A fully representative specimen of any new AFCS shall be submitted to ground tests for design clearance on the general lines given in Volume 1 Leaflet 729/3 para 3.

8.1.4 The tests required by paras 8.1.2 and 8.1.3 shall be done partly in a ground test rig and partly in the rotorcraft on the ground.

8.2 FLIGHT TESTS

8.2.1 Flight Tests shall be done to assess the performance of the AFCS in relation to the requirements of para 3 above (see Section 4).

8.2.2 Flight Tests shall be done, in which failures are simulated throughout the range of speed, altitude, and manoeuvre capability required to be cleared for Service use in the appropriate roles (see Section 4). The objectives of the tests will be:

(i) to afford practical evidence of the flight regimes where a safe recovery can be achieved following a delay equal to the appropriate intervention time,

(ii) to show that the requirements of paras 4 and 5 above have been met,

(iii) to assess the consequences of a failure in accordance with the requirements of Section 3.

SECTION 2 - DEFINITIONS

9 This section gives essential definitions, which are applicable to the requirements of all sections of this Chapter.

10 GENERAL DEFINITIONS

10.1 AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) The function of an AFCS is described in para 1.1. The AFCS includes all those components or elements required primarily for the performance of its functions but would not normally include rotorcraft power supplies or the primary flying controls whether powered or not. However there may be circumstances arising from consideration of overall integrity or of the extent of functional integration where such items should be included within the definition of the AFCS for a particular rotorcraft. In general an AFCS is under the control of the pilot or a member of the crew in certain circumstances, in that the system can be engaged or disengaged by them. However certain modes may, in special circumstances, be applied independently of the crew while others may be modified by the crew through the normal manual flying controls or by the operation of switches.

10.2 MODE

10.2.1 Rotorcraft control, whether powered or not, can be either wholly under direct pilot control (pilot modes) or, in varying degrees, under AFCS control (AFCS modes).

10.2.2 In this Chapter, where the word 'mode' is used alone it refers to an AFCS mode. That is any separately identifiable objective of rotorcraft control performed by an AFCS.

10.2.3 AFCS modes range from pilot control with limited autostabilisation to full Autopilot Control Hands-Off. They can be classified in two main groups:

(i) Autostabilisation Modes (pilot flight, with or without servo assistance, with varying degrees of autostabilisation).

(ii) Autopilot Modes (which include Operational Modes).

10.2.4 Operational Modes may be divided into two groups:

(i) 'Locks' and 'Holds' involving control of a datum (e.g., Heading Hold, Height Lock).

(ii) 'Manoeuvres' (e.g., Automatic Approach) according to whether the mode controlled is held static or is dynamically programmed. Most Navigational Modes (but not all) are 'Locks' or 'Holds' and most Automatic Flight Path Control Systems (AFPCS) Modes (but not all) are 'Manoeuvres'.

10.3 CHANNEL

10.3.1 All those elements of a complete AFCS (including any that may be considered to be redundant) which execute all or part of a mode in one degree of freedom of the rotorcraft, e.g.

(i) Airspeed Hold channel.

(ii) Height Hold channel.

(iii) Lateral Speed Hold channel.

10.3.2 All those elements of a complete AFCS (including any that may be considered to be redundant) that operate in a single rotorcraft control axis, e.g.:

(i) Pitch channel.

(ii) Collective channel.

(iii) Roll channel.

10.3.3 These definitions include any redundant elements that may be provided by, for example, multiplexing or multiplicating.

10.4 LANE

10.4.1 A part of a channel, with no substantial redundancy, which is capable of achieving the basic objectives of the channel, e.g., the provision of a mode. In a simplex system a channel and lane will be identical.

10.5 MULTIPLEXING

10.5.1 A channel is said to be multiplexed when more than one lane is provided and more than one is used simultaneously.

10.6 MULTIPLICATING

10.6.1 A channel is said to be multiplicated when more than one lane is provided but only one is used at a time; the additional lanes being brought into use only as a consequence of a failure.

10.7 FLIGHT SEGMENT

10.7.1 Any period of flight in which the effect of a failure is or may be assumed to be constant throughout the period.

11 OCCURRENCES, EFFECTS, AND PROBABILITY TERMS

11.1 Except as modified by the definitions given below, the definitions of Occurrences (Failures, Events, and Errors) and Effects, as given in BCAR Chapter G1-2 para 7, are applicable.

11.2 FAILURES

11.2.1 The definitions given in BCAR Section G Chapter G1-2 para 7.1.1 will apply to the general Safety Assessment of Systems. For the limited assessment of the AFCS the failures considered should be confined to those systems which directly affect or modulate the AFCS (see Section 1 para 6.1).

11.2.2 Most AFCS failures are described by their effect on the AFCS actuators, e.g.:

(i) rapid runaway,

(ii) slow runaway,

(iii) freeze,

(iv) oscillatory runaway.

11.3 DANGEROUS EFFECT

Any effect more severe than a Minor Effect.

12 PILOT INVOLVEMENT

12.1 In specifying the intervention times that must be demonstrated during flight testing, various levels of Pilot Involvement in the flying task have been defined as follows:

12.1.1 ACTIVE FLIGHT Any flight segment during which the characteristics of the rotorcraft and its autostabiliser necessitate continuous flying of the rotorcraft by the pilot via the flying controls; for example take-off, and tactical low flying.

12.1.2 ATTENTIVE FLIGHT Any flight segment requiring particular attention from the pilot for short periods; for example automatic approach, automatic hovering, and short periods of instrument flight.

12.1.3 PASSIVE FLIGHT Any flight segment of long duration requiring the minimum of attention from the pilot; for example cruise or long periods of instrument flight using autopilot holds.

12.2 The 'Attentive' and 'Passive' phases of flight can be further sub-divided into 'Hands-on' and 'Hands-off' the latter being applicable if the role of the rotorcraft demands that the pilot shall be able to release the flying controls for substantial periods of time.

13 TIMES AND PERIODS

Note: The times and periods defined below are further illustrated and values given in Section 4.

13.1 Rotorcraft Response Time: This is the period between the failure occurring and the pilot being alerted to it by a suitable cue. The cue may take the form of an adequate tactile, audio or visual warning, the latter being in the central warning facility (the eye cannot be relied upon to distinguish abnormal instrument indications sufficiently early for these to be regarded as an adequate cue). In the absence of the cues listed above it can be assumed that a pilot will be alerted when either acceleration along any axis or a change in the rate of rotation in any plane exceeds a specified level (see Section 4 para 30).

13.2 Pilot Response Time: This period commences at the time the pilot is cued to the fact that something abnormal is happening and terminates when the controls are moved to commence the recovery manoeuvre. The period consists of the decision time plus the reaction time. The decision time is assumed to increase as the pilot relaxes his involvement level. The reaction time is longer for 'hands off, than 'hands on' as the pilot has to locate the controls before he can move them (see Section 4 para 31).

13.3 Intervention Time: The total time (i.e. rotorcraft response time plus pilot response time) between failure and commencement of control movement to effect recovery.

SECTION 3 - SAFETY ASSESSMENT, ANALYSES, AND TESTS

14 This section gives requirements for a Failure and Effects Analysis and for the application of a general Safety Assessment of Systems to the AFCS and the rotorcraft as a whole in amplification of the basic requirements given in Section 1 para 6. It also gives minimum requirements for the interpretation of the results of flight tests with failure simulation.

15 The Rotorcraft Specification or the Rotorcraft Project Director will indicate which of the following are required. A minimum analysis will consist of paras 16 (i) and 18 below.

16 At an early stage in the design, using estimated parameters where necessary, the designer shall submit:

(i) a Failure and Effects analysis of the AFCS in accordance with the requirements of para 20 below to show compliance with the requirements of paras 21 and 22 below, and

(ii) a study of the need for, and cost of, a complete Safety Assessment of all Systems in accordance with current civil requirements (see Section 1 para 6.1).

17 At a later design stage if required by the Rotorcraft Project Director, using estimated parameters where necessary, the designer shall submit:

(i) a re-issue of the Failure and Effects analysis to validate the final design, and

(ii) a complete Safety Assessment of all Systems.

18 At the flight test stage, using flight measured parameters wherever possible, the designer shall:

(i) revise and re-issue the earlier analysis whichever was done,

(ii) discuss the results of para 18(i) with the Rotorcraft Project Director,

(iii) do further flight tests as required.

19 The study of para 16 (ii) above shall include a list of the Civil and Military Occurrences (Failures, Effects, and Errors) together with their technical, natural or operational causes where relevant, which it is proposed to consider, and these shall be agreed with the Rotorcraft Project Director (see also Section 2 para 11.2).

20 AFCS FAILURE AND EFFECTS ANALYSIS

20.1 GENERAL

20.1.1 The objective of the analysis shall be to consider every possible failure (see Section 2 para 11.2 and para 19 above) within the AFCS and related applicable failures in the systems which support it or modulate any of its modes or channels.

20.1.2 The extent to which the objective of this section is pursued by the Designer, and the results of the analysis, shall be interpreted in accordance with the appropriate paras of this section as required by, and to the satisfaction of, the Rotorcraft Project Director.

20.1.3 A list of those parts of the AFCS and related systems which support it or modulate any of its modes or channels, which are considered to be sufficiently reliable to be excluded from the analysis shall also be prepared. The list shall be agreed with the Rotorcraft Project Director.

20.1.4 For each flight segment the Rotorcraft Designer shall assign an appropriate Level of Pilot Involvement and Intervention Time (see Section 2 para 12 and Section 4 paras 30 and 31).

20.1.5 The analysis shall indicate the probability and effect of each failure or combination of failures and the total probability of each effect which may arise from one or a number of failures.

20.2 PROBABILITY OF EFFECTS AND REQUIREMENTS FOR FAILURE INDICATION

20.2.1 The probability of a single failure having a Catastrophic Effect at any time shall be less than once in 10^7 flying hours. The design aim should be to eliminate such effects completely. Where this is not possible, indication shall be provided (artificial if necessary) if the failure is not otherwise immediately apparent and if the indication might avert the catastrophe.

20.2.2 The probability of a single failure having a Hazardous Effect within the Intervention Time, shall be less than once in 10^6 flying hours.

20.2.3 The probability of a single failure having a Major Effect within the Intervention Tune shall be less than once in 10^4 flying hours.

20.2.4 The probability of a single failure having a Minor Effect within the Intervention Time shall be less than once in 10^2 flying hours.

21 ANALYSIS OF THE CONSEQUENCES OF FAILURE

21.1 IMMEDIATE CONSEQUENCES OF A FIRST FAILURE

21.1.1 Whatever the type of failure, including irregular control inputs, a safe recovery shall be possible if the pilot delays the commencement of his recovery action for the appropriate Intervention Time.

21.1.2 There shall be no dangerous system malfunctions resulting from the recovery manoeuvre.

21.2 LONG TERM CONSEQUENCES OF A FIRST FAILURE

21.2.1 Following a failure and recovery, it shall be possible to continue the flight to the next landing point unless this capability is specifically excluded in the rotorcraft specification. Where disengagement of the AFCS or any part of it is necessary the requirements of Section 1 para 5.5 shall be met.

21.2.2 Any reduction of the capability to continue the flight within the normal flight envelope shall be agreed with the Rotorcraft Project Director.

21.3 SECOND AND SUBSEQUENT FAILURES

21.3.1 Consideration shall be given to the possibility of a second failure and of subsequent independent failures in the AFCS occurring, in the period between the first failure and the termination of the flight.

21.3.2 The requirements of paras 21.1 and 21.2 above will apply to second and subsequent failures, except that the level of pilot involvement and the flight envelope may change is a consequence of the earlier failures.

21.4 RECOVERY - ALL FAILURES

21.4.1 Compliance with paras 21.1, 21.2 and 21.3 above shall be shown for first and subsequent failures in the most adverse conditions of loading and configuration throughout the appropriate range of altitude and throughout the flight envelope required for the various roles for which the rotorcraft is to be cleared for Service use.

22 FINAL REVIEW (AFCS ANALYSIS)

22.1 If compliance in certain areas of the flight envelope cannot be shown, then the designer shall demonstrate to the satisfaction of the Rotorcraft Project Director that, taking account of the role of the rotorcraft, the probability of a failure occurring in circumstances that would not allow a recovery to be made within the terms of para 21.1 above is less than once in 10^6 flying hours and that the probability of a catastrophe occurring is less than once in 10^7 flying hours.

22.2 The most important effect will be the number of catastrophes which might occur in service. In practice this figure cannot be estimated without making assumptions which are difficult to justify numerically. Among the factors which may be considered are:

(i) the possibility of recovery outside the envelope within which recovery has been shown to be safe.

(ii) the effect of training, during which recoveries from simulated failures are made on the likely pilot intervention times in an emergency,

(iii) the probability of pilot intervention before the nominal intervention point, either on a statistical basis or in response to some special warning device.

22.3 In view of the uncertainties associated with these factors it is necessary that not too much emphasis should be placed on them, and the 1 in 10^6 requirement, which would be based on test evidence must be used to limit the extent to which they may contribute to the estimate of the catastrophe rate.

23 FINAL REVIEW (GENERAL SAFETY ASSESSMENT)

23.1 The requirements of paras 20, 21 and 22 above apply to the AFCS whether or not it is considered in isolation or included in a general Safety Assessment of Systems. If the latter, then the risks arising from consideration of other systems and of other occurrences will need to be considered. The overall level of risk which will be accepted is approximately one order worse than those given in para 22.1 above (that is 10^5 in lieu of 10^6 and 10^6 in lieu of 10^7 approximately).

SECTION 4 - FLIGHT TESTS WITH FAILURE SIMULATION

24 This Section gives requirements which will be used as the basis of the assessment of flight test results (see Section 1 para 8.2).

25 For definitions of terms used see Section 2.

26 Requirements for tests and analyses are given in Section 3.

27 The failures to be simulated in flight shall be agreed with the Rotorcraft Project Director following the completion of the Failure and Effects analysis of Section 3.

28 SPECIAL TEST EQUIPMENT

28.1 Equipment shall be provided for development tests capable of reproducing all the required failures. Provision for its installation in development rotorcraft shall not affect the performance of rotorcraft delivered to Service.

28.2 Unless otherwise agreed the equipment shall be capable of driving each flying control actuator in the AFCS, individually, to the limits of its travel or part way, from any intermediate position, and holding it at the selected position. The device should allow the time taken for the actuator to travel to be infinitely variable from the shortest possible to at least 10 seconds.

28.3 Similar devices shall be provided as necessary to create a freeze or oscillation.

28.4 The design of the test equipment shall be such that the aerodynamic response of the rotorcraft will not interfere with the selected rate of actuator travel or the ability of the actuator to remain in the selected position.

29 METHOD OF TEST

29.1 For the Active flight segments, runaways will be injected without warning the pilot. His ability to recover without a dangerous situation developing will be used to assess system acceptability.

29.2 For the Attentive and Passive flight segments, runaways will be injected but the pilot will be warned.

30 ROTORCRAFT RESPONSE TIMES

30.1 For Active and Attentive flight segments the Rotorcraft Response Time shall be the least of:

(i) the time it takes for the rotorcraft to achieve an angular rate of change about any axis of 3 deg per sec,

(ii) the time it takes for the rotorcraft to increase or decrease acceleration along any axis by 0.2g.

(iii) the time it takes for the relevant attention-getter to function.

30.2 For Passive flight segments the Rotorcraft Response Time shall be the least of (i) (ii) and (iii) above except that the angular rate-of-change shall be 5 deg per sec and the acceleration shall be 0.25g.

31 PILOT RESPONSE TIMES

Flight Segment	Decision Time (Sec)	Reaction Time (Sec)	Pilot Response Time (Sec) (See Note 1)
Active	-	0.5 (See Note 2)	0.5
Attentive Hands on	1.0	0.5	1.5
Attentive Hands off	1.5	1.0	2.5
Passive Hands on	2.0	0.5	2.5
Passive Hands off	3.0	1.0	4.0

Notes (1) Pilot Response Time = Decision Time + Reaction Time
Intervention Time = Pilot Response Time + Rotorcraft Response Time.

(2) But see para 29.1 above.

CHAPTER 605

ROTOR SPEED CONTROL

1 INTRODUCTION

1.1 The requirements in Chapter 605 relate to control over rotor rotational speed in power-on and power-off flight conditions throughout the Permissible Flight Envelope.

2 ROTOR SYNCHRONISATION

2.1 In those rotorcraft configurations involving more than one rotor for lift or control:

2.1.1 Individual rotors shall be positively linked so that the ratio of rotational speed of each relative to the other(s), remains constant.

2.1.2 For those configurations in which rotors intermesh or overlap the probability of failure resulting in loss of synchronisation and possible inter-rotor blade contact shall be extremely remote as defined in Chapter 117 (in preparation - JAC Paper 1169).

3 POWER-ON FLIGHT

3.1 NORMAL OPERATING STATE

3.1.1 Once a datum rotational speed has been set by the pilot, in accordance with the operational procedures laid down for the rotorcraft, and the power demand remains in the governed range, rotor speed shall be controlled by the engine governing system with no specific action required on the part of the pilot to directly control rotor speed, unless otherwise stated in the Rotorcraft Specification. (see Leaflet 605/1, para 2.1).

3.1.2 The transient response characteristic of the engine and governing system shall be such that following a power demand change, the appropriate datum rotor speed shall be re-established without undue delay or overshoots.

3.1.3 In aggressive manoeuvres, especially those demanding large power increases from a low initial level, including autorotation, the transient rotorspeed droop shall not result in an unacceptable delay in the development of rotorcraft response to flying control inputs.

3.1.4 Natural frequency and damping characteristics of the engine and governing system shall not cause any adverse interaction with structural or vibratory modes of the rotor, or transmission sub-systems, at all power levels.

3.1.5 Engine governing systems may take advantage of input signals derived from flying control inceptors or other relevant rotorcraft sensors in order to meet the power plant response requirements of para 3.1, provided that they have satisfactory reliability and failure modes. (see Leaflet 605/1, para 3.1).

3.2 FAILURE STATES

3.2.1 Unless an engine governing system possesses a sufficient degree of redundancy, to the satisfaction of the Rotorcraft Project Director, in the event of a failure in that governing system, the pilot shall be able to exercise direct control over the power output of the affected engine.

3.2.2 It shall be possible to shut-down and restart an engine in flight, under appropriate conditions in accordance with established procedures.

3.2.3 In the event of an engine failure in a multi-engined rotorcraft operating in a critical flight condition, it shall be possible to exploit a permitted transient over-torque limit where this exists in order to fly away to a safe flight condition by trading rotational energy, allowing the rotor speed to drop to a value to be specified in a recommended emergency procedure. (see Leaflet 605/1, para 3.2).

4 POWER-OFF FLIGHT (see Leaflet 605/2, para 1)

4.1 ENTRY TO AUTOROTATION

4.1.1 With the rotorcraft in a Normal Operating State, the engine and governing system response characteristics shall be such that, during those manoeuvres in which the net rotor power required is off-loaded as a result of control inputs and/or rotorcraft nose-up incidence changes, the rate of decay of engine power is sufficiently high and the flight idle power level sufficiently low to avoid rotor overspeeding.

4.1.2 The transmission system design, and the engine governing system stability in the low power and flight-idle operating regimes, shall be such that as the rotor power requirement falls through zero the rotor system smoothly disengages from the drive system, with no hunting by the latter.

4.1.3 In the event of a total loss of drive to the rotor system while under a torque loading, the free-wheel process referred to in para 4.1.2 shall operate smoothly. In addition, the aerodynamic, dynamic and rotational inertial characteristics of the rotor system shall be such that a delay in pilot corrective action acceptable for the rotorcraft at the flight condition in which the failure occurred, shall not result in an irreparable loss of rotor speed that would prevent a subsequent safe recovery.

4.2 ROTOR SPEED IN AUTOROTATION

4.2.1 In any transient or steady autorotational flight condition the overall rotorcraft response characteristics shall be such that the pilot can readily co-ordinate control over rotor speed and the flight path through the flying controls, without the use of undue skill or physical strength. (see Leaflet 605/2, para 2.1).

4.2.2 At a given steady autorotative flight condition, with the other flying controls retained at their initial settings, following a small step input to the collective pitch channel, the rotor speed shall settle to a new value in accordance with Leaflet 605/2, para 2.2.2.

4.2.3 From a given steady autorotative flight condition, following a small discrete longitudinal control input aimed at adjusting airspeed by up to 5 kts in either direction, with the other flying controls retained at their initial settings, the rotor speed shall settle to a new value without excessive transient overshoots. (See Leaflet 605/2, para 2.2.3).

4.2.4 Variations in rotor speed response to given magnitudes of collective or longitudinal control inputs (as in para 4.2.2 and 4.2.3), over ranges of all-up weight, airspeed and initial rate of descent allowable within the Service Flight Envelopes, shall be progressive and predictable with no unexpectedly abrupt discontinuities (see also Leaflet 605/2, para 2.2.4).

4.2.5 Following an emergency entry to autorotation, as in para 4.1.3, in which a large initial collective pitch reduction within the range available may lead to excessive over-speeding if uncorrected, the pilot shall have no difficulty in reselecting a collective setting in order to establish an acceptable rotor speed within the design limits.

4.2.6 Variation in transient rotor speed response following small longitudinal control inputs with variation in initial flight conditions, and/or rotorcraft loading shall be progressive and predictable. It shall also be possible to counter any marked increase in sensitivity in rotor speed response to longitudinal control inputs with increasing initial airspeed, by means of an appropriate collective pitch adjustment, without setting up an oscillatory interchange of airspeed rotor speed or rate of descent.

4.3 POWER-OFF LANDING

4.3.1 The Contractor shall establish recommended procedures for safe emergency power-off landings covering the range of applicable rotorcraft loadings, at the critical relative atmospheric density called for by the Rotorcraft Specification (see Leaflet 605/2, para 3.2.1). The procedures will also include definitions of the limiting horizontal and vertical touch-down velocities. (See also Leaflet 605/2, para 3.2.3).

4.3.2 From a steady autorotational descent at the recommended rotor speed and approach speed, there shall be adequate longitudinal control available for the rotorcraft to be flared to an appropriate nose up attitude for deceleration to the limiting run-on landing speed or less. Maximum power-off rotor speed limit shall not be exceeded during this phase. (See leaflet 605/2, para 3.2.2).

4.3.3 It shall be possible to maintain the deceleration as required by para 4.3.2, while raising collective pitch in order to cushion the final descent to less than the vertical touch down velocity limits.

4.3.4 During the approach, flare and final touch down, the lateral and directional control activity necessary to maintain an acceptable track and attitude orientation shall not be unduly high.

4.3.5 At no stage during the power-off landing manoeuvre shall any vital rotorcraft service, facility or instrumentation cease to be available to the pilot, as a result of the effect of rotor speed variations on the behaviour of any rotor-driven pumps, generators or accessories.

4.3.6 If the Rotorcraft Specification calls for repeated power-off landings to be made, for example in a training role, then full account shall be taken of any cumulative fatigue and ultimate damage likely to be sustained under those conditions, and the Rotorcraft Project Director advised accordingly.

4.4 ROTOR SPEED MARGINS

4.4.1 The maximum transient power-off rotor speed limit shall be realistically set, having regard to the structural integrity of the rotor system, as far as practicable above the normal needles-split rotor speed in order to increase the time allowable for pilot reaction to audio or tactile cues, or an automatic overspeed warning if fitted, in critical or highly active flight phases.

4.4.2 The minimum transient rotor speed limits, power-on and power-off, shall be realistically set as low as practicable, having regard to structural integrity of the rotor system and system requirements on rotor-driven pump, generator or accessory characteristics in order to maximise the opportunity for safely carrying out power off landings or emergency fly-away manoeuvres, such as referred to in para 3.2.3.

LEAFLET 605/1

ROTOR SPEED CONTROL

POWER-ON FLIGHT

1 INTRODUCTION

1.1 The overall objective of the requirements in Chapter 605, para 3 is to reduce pilot workload by removing the necessity for him to take any control action specifically to regulate rotor speed, other than the setting of datum speeds.

2 NORMAL OPERATING STATE

2.1 Responsibility for control over rotor rotational speed under power-on conditions is generally to be vested in the engine governing system, although Chapter 605, para 3.1.1 does allow this to be waived in the Rotorcraft Specification where this would be appropriate, for example, in extremely small or unsophisticated rotorcraft, or alternative configurations not employing conventional shaft drive.

3 FAILURE STATES

3.1 Chapter 605, para 3.1.5 recognises the application of anticipatory flying control input signals and rotorcraft response feedback signals in the engine/rotor speed control loop.

3.1.1 Such cross feed may lead to enhanced levels of engine transient response and protection of design limits under manoeuvring conditions. However, the more factors taken into account, the more complex the numerous interactions are likely to be.

3.1.2 Care must be taken to avoid unwarranted assumptions on the reliability of the sources, and the validity of the scheduling, of the various signals referred to in para 3.1.

3.1.3 No single failure of any equipment introduced to achieve the enhancements referred to in para 3.1.1 should render the engine response characteristics inferior to the basic, unenhanced ones.

3.2 The requirement in Chapter 605, para 3.2.3 is intended to widen the options available, for example, in the case of a rotorcraft hovering under adverse conditions of all-up-weight, air temperatures etc., at a low height dictated by operational considerations, with a less than adequate power margin for continued hovering following an engine failure.

3.2.1 If some of the rotational kinetic energy inherent in the rotor system can be absorbed in arresting the rate of descent and accelerating to a forward flight condition which could be sustained on the remaining power available, this fly-away manoeuvre would provide a viable alternative to the otherwise inevitable forced landing or ditching.

3.2.2 The fly-away manoeuvre referred to in para 3.2.1 would be subject to an emergency procedure, with the piloting techniques necessary to achieve specific performance levels established by the Contractor for the particular rotorcraft, recognising the minimum rotor speed and maximum transient torque design limits.

3.2.3 Recommendations for logging extreme torque excursions and for subsequent inspection of relevant engine and transmission components would also form part of that emergency procedure.

LEAFLET 605/2

ROTOR SPEED CONTROL

POWER-OFF FLIGHT

1 INTRODUCTION

1.1 Autorotative conditions can be entered inadvertently or deliberately during manoeuvring flight, and of necessity in certain failure situations. The requirements in Chapter 605, para 4 seek to ensure that any transition into autorotation can always be effected smoothly and subsequent recovery made safely, and that the additional rotor speed degree of freedom can be adequately controlled.

1.2 The requirements of Chapter 605, para 4 are intended to embrace "autogyro" configurations in which lift is derived from an autorotating rotor system, with propulsion provided by additional devices.

2 TRANSIENT ROTOR SPEED CHARACTERISTICS

2.1 GENERAL

2.1.1 Two major factors affecting the control of rotor speed in autorotational flight are:

(i) Rotor collective pitch application. An increase initially increases the aerodynamic torque loading thereby decelerating the rotor until equilibrium is established at a lower rotational speed.

(ii) Rotor disc incidence. A nose up change, resulting from an increased rate of descent or rotorcraft pitching response to an aft longitudinal control input, offloads aerodynamic torque, leading to a higher rotational speed.

2.1.2 Both the factors outlined in para 2.1.1 (i) and (ii) individually result in transient increases in rotor thrust (load factor). The effects of (ii) increase progressively with forward speed for a given incidence change.

2.1.3 The dominant handling issue, in the context of rotor speed control, will depend on the particular flight phase. For example, during the manoeuvres addressed in para 3.1.1 the main concern will be to avoid overspeeding, whereas in the power failure situation of para 3.2 the pilot's major preoccupation is to preserve an adequate margin of kinetic energy in the rotor for the landing phase.

2.2 RESPONSE TO CONTROL INPUTS

2.2.1 In the complex interactive autorotational flight situation, it is difficult to define in detail any comprehensive quantitative criteria. The present approach is to identify characteristics of responses to individual control inputs which can form a sound basis for acceptable operational usage. It is anticipated that quantitative data may subsequently be refined or extended in the light of experience and test results.

2.2.2 With reference to Chapter 605, para 4.2.2:

(i) An increased collective pitch setting should result in reduced rotor speed. A collective pitch reduction of a similar magnitude should lead to a correspondingly similar increase in rotor speed. These speed variations should be substantially "first order" in character, subject to criteria of the type outlined in Leaflet 600/8, para 4.4 to 4.6 and figs 3 and 4, and similarly applied in Leaflet 607/1, para 4.2.

(ii) The new reference rotor speed should be reached in a short time interval, in a progressive manner with no irregularities in the variation of rate of change. In the absence of any other control inputs or disturbances to the rotorcraft, the new rotor speed should then remain, within specified limits. The time to reach the new reference rotor speed, and subsequent holding tolerance criteria remain to be established.

(iii) The reference value referred to in para 2.2.2 (ii) should be substantially proportional to the magnitude of the control input for small inputs, and the rotor speed response per unit control input should not increase above this rate for moderate or larger inputs.

(iv) It is important to avoid oversensitivity. The rate of change of rotor speed with collective control input should be in harmony with the margins between the normal needles-split rotor speed and the maximum and minimum transient rotor speed limits.

2.2.3 Rotor speed transient response to longitudinal control inputs are more difficult to define in quantitative terms because of the dependence on the build up of flow through the rotor, itself depending on rotorcraft pitching response, airspeed, etc. The 5 kt speed adjustment in Chapter 605, para 4.2.3 was arbitrarily chosen as an identifiable flight condition parameter.

(i) The rotor speed should increase following an aft longitudinal control input, and reduce following a nose down input. The rotor speed increment should be substantially proportional to the magnitude of the input, in either direction.

(ii) Avoidance of oversensitivity to longitudinal flying control inputs is also very important. Rotor speed excursion limits (TBD) should be related to the margins available within the rotor speed design envelope, as outlined in para 2.2.2 (iv).

2.2.4 Progressiveness and predictability throughout the range of operating conditions likely to be met are important issues, especially in the high pilot workload situations typical of autorotative flight. The Contractor should ensure that, as far as is practicable, no cross-couplings or abrupt discontinuities in rotor loadings are likely to occur, which would significantly affect the general degree of linearity or reverse any of response characteristics addressed in para 2.2 above, due to:

(i) Rotor blade aerodynamic or aeroelastic characteristics under the combined effects of incidence, compressibility etc., encountered in autorotational flight.

(ii) Inherent rotorcraft response cross-couplings, including the effects of control mixing designed to improve handling qualities in other flight operating regimes.

(iii) Functioning of any automatic flight control system facilities.

3 MANOEUVRES

3.1 NORMAL OPERATING STATE

3.1.1 In aggressive quick stop manoeuvres, where tactical considerations dictate that rotorcraft height excursions be severely restricted, containment of rotor over-speeding may constitute a potential handling problem.

(i) Conventional helicopter configurations rely on high nose up attitude to derive deceleration from the inclined thrust vector. The resulting throughflow then offloads the lifting rotor into an autorotative state. Rotorcraft kinetic energy released by the deceleration can not be converted into potential energy by allowing the rotorcraft to climb, and so goes into accelerating the rotor.

(ii) Piloting techniques, including turning to present a larger drag surface, may be applicable, depending on the airspace and external visibility available. This approach however is likely to impinge on the Sideslip Envelope, and may prompt a design review of those restrictions.

(iii) Configuration refinements open to the designer include the provision of air brakes, some form of rotational braking on the rotor system, and some direct longitudinal thrust control device providing rotorcraft retardation independently of main lifting rotor disc tilt. These various options would need to be considered carefully to balance overall mission effectiveness against cost and weight penalties and additional complexity associated with the control over their deployment and operation, compared with say, the implications of opening out the traditional Rotor Speed Limitations.

3.2 POWER-OFF LANDINGS

3.2.1 The requirement in Chapter 605, para 4.3.1 calls for a safe emergency landing capability.

(i) This means that the occupants of the rotorcraft shall survive the landing, but does not necessarily preclude minor damage to, or the need for subsequent inspection of critical components of, the rotorcraft in the most adverse combination of circumstances.

(ii) It is also tacitly assumed that a suitable landing site is available. This would consist of substantially flat, reasonably firm ground, sufficiently unrestricted to allow the rotorcraft to be brought to rest from the recommended touch down speed. Over water the success of the final ditching would be highly dependent on the current Sea State, which cannot be predicted. The objective of the emergency procedure would be to arrive at that final point with the best possible margins of control and energy stored in the rotating systems.

3.2.2 Chapter 605, para 4.3.2 assumes that it will have been possible to establish a controlled steady autorotational descent prior to initiating the power-off landing. Without prejudging the technique to be recommended for any particular rotorcraft, the following guidelines may be noted:

(i) Approach speed should be selected to allow a near-minimum rate of descent, together with a rotor speed affording a margin for acceleration as in para 3.2.2(iii).

(ii) Approach speed should also be such that the limiting horizontal touch down speed, or less, can be achieved by a comfortable deceleration as in para 3.2.2 (iii), compatible with the vertical element in Chapter 605, para 4.3.3.

(iii) The nose up attitude change required to decelerate to the horizontal touch down speed shall not be sufficiently high to unacceptably impair critical outside visual cues, cause disorientation of the crew or require collective pitch adjustments incompatible with the vertical element in Chapter 605, para 4.3.3 in order to prevent rotor overspeeding.

3.2.3 The landing capability of the undercarriage and the minimum horizontal and vertical touch down velocities which can be achieved from the recommended power-off landing technique must be compatible with each other.

CHAPTER 606

GROUND AND WATER HANDLING QUALITIES

1 INTRODUCTION

1.1 The requirements in Chapter 606 relate to the pre-flight and post-flight operations of the rotorcraft by the flight crew. Wherever relevant these operations will take place on:

(i) Fixed ground surfaces, including elevated and/or restricted clearance platforms;

(ii) Approved ship decks;

(iii) Water surfaces;

2 GROUND OPERATIONS

2.1 OPERATING CONDITIONS

2.1.1 The requirements in para 2 shall be met in wind conditions up to the most critical combinations of speed and direction as defined for the Low Speed Flight Envelope. (See Leaflet 600/3, para 2.1).

2.1.2 Unless otherwise stated the requirements in para 2 apply to rotorcraft operations on substantially flat and level, dry, firm surfaces.

2.1.3 The operations shall be possible over the ranges of altitude and temperature defined by the Rotorcraft Specification.

2.2 START UP AND SHUT DOWN

2.2.1 Under the conditions of para 2.1 it shall be possible for the aircrew to carry out the following operations in accordance with procedures laid down for the particular rotorcraft, without the assistance of personnel or equipment external to the rotorcraft:

(i) Start and stop each engine as required, and set up governed running conditions.

(ii) Engage accessory drive to rotorcraft sub-system generators, pumps etc.

(iii) Spread and fold as necessary rotor blades and any other rotorcraft component where these operations are automated.

(iv) Engage the rotor system, run up to any rotor speed within the operating limitations, and stop the rotor.

(v) Carry out all necessary preflight cheeks on rotorcraft controls and systems, etc.

2.2.2 It shall be possible to ensure that, prior to running the rotor system up to normal operating speed the flying control settings are such that, when that operating speed is reached, no forces or moments will be generated which are sufficiently large to:

(i) Apply undue stress or load to any rotorcraft component.

(ii) Significantly lighten the load on the undercarriage, displace or topple the rotorcraft about any axis.

2.3 POSITION RETENTION

2.3.1 During a vertical lift off from the surface it shall be possible for the pilot to readily maintain the last point of contact between the undercarriage and the surface effectively fixed, as lift is increased to the point of take-off. Similarly it shall be possible to maintain the first point of contact fixed as lift is reduced to zero during a vertical touch down.

2.3.2 Para 2.3.1 is not intended to preclude any minor "scrubbing" of the undercarriage necessary to accommodate rotorcraft attitude changes inherent in the manoeuvres.

2.3.3 At no time during the operations described in para 2.3.1 shall any oscillatory motion of the rotorcraft develop which can not be immediately damped out by a positive action on the part of the pilot to complete the lift off or touch down phase.

2.3.4 The requirements in para 2.3.1, 2.3.2 and 2.3.3 shall be applicable to surface slopes, and rotorcraft orientation to the slope, as defined by the Rotorcraft Specification.

2.3.5 It shall be possible to park the rotorcraft at any orientation on the surface slope required by the Rotorcraft Specification, and for that position to be retained after rotor and engine shut down.

2.4 TAXIING

2.4.1 The requirement in para 2.4.1 shall be met by rotorcraft with wheeled undercarriage under all operating conditions covered by para 2.1.

(i) It shall be possible, without the use of wheel brakes, to maintain a designated straight path.

(ii) It shall be possible to make complete 360 deg. turns in either direction, by effectively pivoting on either main landing gear.

(iii) It shall be possible to execute an emergency quick stop, without losing control of the rotorcraft.

(iv) Operation of wheel brakes shall be well balanced with no unacceptable jerkiness in the rotorcraft response.

2.4.2 It shall be possible to carry out all necessary taxi manoeuvres under the most adverse conditions of para 2.1, and also for protracted periods over surfaces such as jointed concrete, without:

(i) Causing damage to rotor stops.

(ii) Causing any rotor blade to contact any part of the rotorcraft structure.

(iii) Applying undue stress or load to any rotorcraft component.

(iv) Any rotorcraft component, other than the undercarriage or a protective device designed for that purpose, coming into contact with the ground.

2.5 RUNNING TAKE-OFF

2.5.1 When operational necessity requires a running take-off to be made, it shall be possible to accelerate the rotorcraft to the speed for lift off, under the operating conditions covered by para 2.1, while meeting the following requirements:

(i) Maintain a designated straight path, within the lateral drift design limits of the undercarriage, without the use of wheel brakes or unduly large or rapid control inputs, out of keeping with the normal in-flight characteristics of the rotorcraft.

(ii) No poorly damped oscillatory motion about any rotorcraft axis shall be promoted.

(iii) Potentially damaging occurrences such as those in para 2.4.2 (i) to (iv) shall be avoided.

2.5.2 At the point of lift off, no unduly large or rapid control inputs shall the required in order to prevent any unexpectedly marked deviation in the path followed by the rotorcraft.

2.5.3 Approved procedures for running take-offs shall be established for the particular rotorcraft. These shall include emergency procedures following engine control failures, and the definition of stopping distances for aborted take-offs.

2.5.4 When a running take-off is to be carried out in a wet or dusty environment, the rotorcraft design shall prevent unacceptable impairment of critical external visibility due to spray or raised dust. The risk of damage or loss of engine performance due to FOD shall also be minimised.

2.6 RUN-ON LANDING

2.6.1 The requirements of para 2.6 shall be met under the operating conditions covered by para 2.1.

2.6.2 At the first, (and any subsequent), contact between the main undercarriage and the ground, the vertical, forward and lateral ground velocity components of the rotorcraft shall have been brought within the design limits for the undercarriage.

2.6.3 Para 2.6.2 is applicable to all types of undercarriage, wheeled or otherwise, and whether the landing is being carried out as usual practice with the rotorcraft in its Normal Operating State, or as an emergency following some failure.

2.6.4 With the rotorcraft in its Normal Operating State, it shall be possible to meet the following requirements:

(i) Bring the rotorcraft smoothly to a stop in a designated straight line without unduly large or rapid control inputs out of keeping with the normal in-flight characteristics of the rotorcraft.

(ii) No poorly damped oscillatory motion about any rotorcraft axis shall be promoted.

(iii) Potentially damaging occurrences such as those in para 2.4.2 (i) to (iv) shall be avoided.

2.6.5 The requirements in para 2.6.4 shall also be met in the case of the repeated power-off landings for training, referred to in Chapter 605, para 4.3.6.

2.6.6 Where a run-on landing is necessitated by an emergency following a power or control system failure, it shall be possible to bring the rotorcraft safely to a stop.

2.7 RECIRCULATION

2.7.1 At no time during the Ground Operations addressed in para 2 shall the functioning of any rotor, power source, control system or associated rotorcraft sub-system or the operational efficiency of the aircrew be unacceptably impaired by the recirculation or ingress of exhaust fumes, moisture, snow, ice, dust or debris etc.

3 SHIPBORNE OPERATIONS

3.1 OPERATING CONDITIONS

3.1.1 The requirements in para 3 are applicable to operations on approved ship type decks referenced in the Rotorcraft Specification.

3.1.2 The requirements in para 3 shall be met in relative wind conditions up to the most critical combinations of speed and direction defined for the Low Speed Envelope, (Leaflet 600/3, para 2.1), having regard to any localised plume or superstructure - generated turbulence characteristics of the particular ship type.

3.1.3 Limiting components of acceleration, angular rate and periodicity of deck motion under which the rotorcraft is required to operate will be defined by the Rotorcraft Specification.

3.1.4 Atmospheric temperature, and other relevant environmental conditions will be defined by the Rotorcraft Specification.

3.2 START UP AND SHUT DOWN

3.2.1 Under the conditions of para 3.1 the requirements given in para 2.2.1 (i) to (v) and 2.2.2 are also applicable to shipborne operation.

3.3 DECK RESTRAINT

3.3.1 The rotorcraft design shall be such that, under the most adverse conditions of relative wind and deck motion referred to in para 3.1, the rotorcraft can be maintained at a fixed point on the deck without sliding or toppling.

3.3.2 In those cases where the rotorcraft-ship interface includes a device which physically attaches the rotorcraft to the ship, in order to meet the requirement in para 3.3.1, it shall be possible for the pilot to sever this connection instantly in the case of an emergency.

3.3.3 In those cases where the requirement in para 3.3.1 is met, or partially met, by exerting a rotor-generated down load to increase the adherence of the rotorcraft to the deck, the following requirements shall also be met:

(i) The normal in-flight minimum collective pitch setting which is crucial to the entry into autorotational flight (see Chapter 605, para 4.2.5), shall be positively identified to the pilot by means of an unambiguous tactile cue at the collective controller.

(ii) It shall be necessary for the pilot to exert a distinct force to break through from the normal in-flight range into the "sub-minimal" range to generate the reversed thrust referred to in para 3.3.3.

(iii) The requirements in para 2.2.2 are also applicable.

(iv) Any control mixing or interlinking to the collective pitch channel shall not result in any unacceptable effects when the "sub-minimal" regime is entered.

3.4 MOVEMENT ON DECK

3.4.1 It shall be possible to rotate the rotorcraft in yaw about a fixed point on the deck to any heading. This applies whether or not any deck restraint system as in para 3.3.2 is engaged.

3.4.2 It shall be possible to taxi the rotorcraft in any direction on the deck, when this is in accordance with the operating procedures drawn up for the particular rotorcraft and ship. Under these conditions the requirements in para 2.4.2 (i) to (iv) are also applicable.

4 WATERBORNE OPERATIONS

4.1 OPERATING CONDITIONS

4.1.1 Unless stated otherwise the requirements in para 4 are applicable to smooth sea conditions.

4.1.2 If operation of the rotorcraft in a Normal Operating State involves start up, take-off and landing on water surfaces, the Sea State conditions under which this is to be achieved will be defined by the Rotorcraft Specification.

4.1.3 In the event of an emergency necessitating ditching, the Rotorcraft Specification will define the highest Sea State in which this is to be safely carried out.

4.2 START UP AND SHUT DOWN

4.2.1 Under the conditions of para 4.1.2 the usual requirements of para 2.2.1 (i) to (v) and 2.2.2 (i) are applicable. The forces and moments on the rotorcraft during run-ups and run-downs shall not initiate any poorly damped oscillatory motion or topple the rotorcraft about an axis.

4.3 TAXIING

4.3.1 It shall be possible to taxi the rotorcraft at surface speeds up to the limit of the Rotorcraft Specification, in a straight line, and to make gentle turns, without:

(i) Causing damage to rotor stops

(ii) Causing any rotor blade to contact any part of the rotorcraft structure, or the water surface.

(iii) Applying undue stress or load to any rotorcraft element.

(iv) Causing critical water ingestion or buffeting due to the generation of surface waves or spray.

4.4 DITCHING

4.4.1 Procedures shall be established for reducing the forward speed, rate of descent and lateral drift of the rotorcraft at the point of contact to acceptable design levels. These procedures shall also have regard to the deployment of auxiliary flotation gear, and the achievement of satisfactory pitch and roll attitudes in order to maximise the probability of ditching safely.

4.4.2 Para 4.4.1 shall be applicable to both power-on and power-off cases.

4.5 EMERGENCY TAKE-OFF

4.5.1 Procedures shall be established for the management of engine power and rotor rotational energy, should this remain available, for a viable take-off at reduced all-up-weight under favourable circumstances.

CHAPTER 607

LIFTING FLIGHT HANDLING QUALITIES

1 INTRODUCTION

1.1 The requirements in Chapter 607 relate to those specific aspects of operation of the lifting flying controls and the associated rotorcraft responses in vertical motion and incremental normal acceleration, not already dealt with in Chapters 600, 601 and 602. (See also Leaflet 607/1, para 1.1 and 1.2).

2 VERTICAL MOTION BEHAVIOUR

2.1 CHANGES OF FLIGHT CONDITION (See also Leaflet 607/1, para 2.1)

2.1.1 The rotorcraft design shall be such that, while being operated close to a substantially level, horizontal surface at relative airspeeds up to the limits of the Low Speed Envelope, the following requirements are met:

(i) No large or abrupt vertical velocity transient, resulting in critical height variations, shall occur when the rotorcraft heading relative to the net airspeed vector is varied over the full range and at rates up to the limits of the Low Speed Envelope, without the pilot making any adjustments to the lifting flying control setting.

(ii) During continuous accelerations in any direction, any adjustments to the lifting flying control setting necessary to maintain substantially constant height shall be progressive, with no obtrusive discontinuities.

(iii) Any adjustments to the lifting flying control setting by the pilot necessary to maintain a zero ground speed hover, or a particular translational flight condition, as the height above the surface is varied within ground effect, shall be progressive, with no unexpected discontinuities.

2.1.2 The rotorcraft design shall be such that during deliberate transition from one flight phase to another anywhere in the Forward Flight Envelope, especially where this involves significant variations in rotorcraft angle of incidence, there are no unexpectedly abrupt changes in normal acceleration or the lifting flying control setting necessary to maintain a controlled flight condition.

2.1.3 When entry to an autorotative flight condition is necessitated by engine or transmission system failure, this shall be readily achievable by the appropriate adjustment of the lifting rotor collective pitch setting, without a need for unacceptably large inputs in other control channels in order to maintain a balanced flight condition (see Leaflet 607/1, para 2.2.2).

2.1.4 In the event of the failure referred to in para 2.1.3, while operating under autopilot control, the requirements of either para 2.1.4(i) or (ii) shall be met:

(i) The autopilot system shall automatically avoid taking inappropriate action to maintain height, vertical velocity or airspeed at the expense of a critical loss in rotor speed.

(ii) Any autopilot mode functioning through the collective channel shall readily be overridden by the action of the pilot intervening to correct the collective pitch setting, in accordance with the relevant emergency procedures laid down for the particular rotorcraft.

2.2 LONG TERM VERTICAL SPEED AND HEIGHT HOLDING

2.2.1 The requirements of para 2.2 refer primarily to the non-manoeuvring, near-steady state conditions prevailing in the Attentive and Passive flight phases, with flying control settings retained as described in Chapter 600, para 9.2.2.

2.2.2 In those mission flight phases which call for a steady flight condition to be maintained over a protracted period of time, albeit with the pilot Attentive in other axes, for LEVEL 1 handling qualities it shall not be necessary for the pilot to make any flying control adjustments in order to maintain the rotorcraft height (or rate of change of height where relevant) within acceptable limits about the established datum, (see Leaflet 607/1, para 3.2).

2.2.3 The acceptable limits for deviation from datum are dependent upon the particular mission flight phase, and will be defined by the Rotorcraft Specification or the Automatic Flight Control System Specification.

2.2.4 In those rotorcraft in which over-water automatic height holding facilities employ sensors referenced to the sea surface, adequate attenuation of surface motion shall be provided in accordance with the Rotorcraft or Automatic Flight Control System Specifications. The pilot shall be provided with unambiguous indications of either the actual or a smoothed mean height of the rotorcraft above the surface, or both, as relevant to the particular mission flight phase.

2.2.5 Performance requirements for height holding systems described in para 2.2.4 shall be expressed in terms of realistically observable parameters.

2.2.6 The lifting flying control system characteristics and the associated rotorcraft behaviour shall be such that in the event of a minor modification to an otherwise steady flight condition, or correction for the effects of an external disturbance, the appropriate lifting control adjustment can be completed without undue delay and any overshoot or hunting in the rotorcraft flight path or airspeed, (see also Leaflet 607/1, para 3.3).

2.3 SHORT TERM RESPONSE CHARACTERISTICS

2.3.1 The requirements in para 2.3 refer primarily to vertical velocity and incremental normal acceleration responses to discrete inputs, up to the moderate category, applied via a single controller. See Leaflet 607/1, para 4.1.

2.3.2 VERTICAL VELOCITY

(i) When initiated from a near-hovering condition, the vertical velocity response to a step input in collective pitch control shall satisfy the general requirements for consistency and settling time given in Chapter 600, paras 11.5.2 and 11.5.3. The response shall also be largely "first order" in character, and be in accordance with the recommendations of Leaflet 607/1, para 4.2.

2.3.3 INCREMENTAL NORMAL ACCELERATION

(i) Following a step input in the lifting flying control or the longitudinal control channels applied at any steady flight condition in the Forward Flight Envelope, the normal acceleration response shall rise progressively to a peak value which can be unambiguously associated with the specific control input.

(ii) During the rise time there shall be no objectionable hesitation or reversal of slope of the time history. (see Leaflet 607/1, paras 4.3.1 to 4.3.3).

(iii) Subsequent to the peak response of para (i), and in the absence of any other control input or external disturbance, the normal acceleration excursion shall decay smoothly towards zero with no objectionable hesitations or reversals of slope. (see Leaflet 607/1, para 4.3.4).

3 CROSS-COUPLING

3.1 Longitudinal, lateral and directional rotorcraft responses, and also rotor speed variations induced by lifting flying control inputs shall not be unacceptably obtrusive upon the execution of mission flight phases called for by the Rotorcraft Specification. (see Leaflet 607/2, para 2).

3.2 Containment of the cross couplings referred to in para 3.1 shall not require undue control activity, skill or physical strength on the part of the pilot to complete the mission flight phases.

3.3 Any variations in the responses referred to in para 3.1 with progressive changes in flight condition or the aggressiveness of the lifting control input, shall also be progressive, with no unexpectedly abrupt discontinuities.

3.4 Where stability and control augmentation devices are employed to meet the requirements of para 3, or to exploit any beneficial application of cross coupling effects, systems shall have adequate reliability and failure mode characteristics. Attention is drawn to Leaflet 607/2, para 3.2.3.

4 OPERATIONS IN THE LOW SPEED ENVELOPE

4.1 The requirements of para 4 relate primarily to low level operation and low speed manoeuvring during Active Flight Phases, and to Attentive Flight Phases under autopilot control. The major requirements centre around the precision of vertical velocity and height control of the rotorcraft, relative to external references, fixed or moving.

4.1.1 The requirements of para 4 shall be met under the following conditions:

(i) Most adverse combination of rotorcraft loading and ambient conditions of atmospheric relative density, wind speed and direction.

(ii) Unless stated otherwise assuming the pilot has adequate visual cues, relevant to the particular external reference.

4.1.2 The response of the rotorcraft to a discrete input to the collective pitch control, including any effects due to variation in ground effect shall be such that the pilot has no undue difficulty in making predetermined height adjustments. These height adjustments shall be made with the precision and aggressiveness necessary to execute the mission tasks called for by the Rotorcraft Specification, in either direction, with the minimal overshoot and settling time.

4.1.3 Engine response to power demands shall not prevent the requirements of para 4.1.2 being met.

4.2 FLIGHT UNDER AUTOMATIC CONTROL

4.2.1 Where the Rotorcraft Specification calls for rotorcraft height to be controlled automatically under critical low-level, low speed flight conditions, either to achieve a repeatable degree of accuracy or because of the absence of adequate external reference cues precludes active pilot execution of the tasks:

(i) The required accuracy of height hold for the particular mission will be defined by the Rotorcraft Specification or the Autopilot System Specification.

(ii) The rotorcraft design shall be such that in the event of a critical autopilot failure the pilot can readily override the collective channel to prevent a dangerous loss of height, or over-torque of the engine/transmission system, having regard to appropriate pilot intervention delay times.

4.2.2 In the event of a critical power plant or transmission system failure while operating in the Low Speed Envelope, the pilot shall be able to readily apply the appropriate collective pitch control inputs, in conjunction with other flying controls, to carry out either a fly-away manoeuvre to a safe flight condition, or a safe touch down, in accordance with recommended emergency procedures and having regard to appropriate pilot intervention delay times. The pilot's collective control action shall override any automatic flight-control function which is not compatible with the recommended emergency procedure.

4.2.3 In those rotorcraft where subsystem redundancy and/or available power margins in the Normal Operating State provide a failure survival capability, such that a single system or power plant failure is not critical in the sense of significantly disturbing the rotorcraft equilibrium or flight path, the pilot shall be given unambiguous warning of that failure if any subsequent failure would then become critical.

5 MANOEUVRES IN THE FORWARD FLIGHT ENVELOPE

5.1 Requirements in para 5 are primarily concerned with increments in nominal acceleration (load factor) and variations in height and vertical velocity, away from constraints of ground surfaces and associated obstacles.

5.2 INCREMENTAL NORMAL ACCELERATION

5.2.1 The requirements in para 5.2.1 complement those in para 5 of Chapters 601, 602 and 603. See also Leaflet 607/1, para 4.1.5.

5.2.2 Following a step input to the lifting flying control that results in peak variation in normal acceleration in the range ±0.5g, and still remaining within the Service Flight Envelope, the following requirements shall be met:

(i) The transient response of normal deceleration shall show the smooth rapid build up to the peak, followed by the progressive decay to zero described in Leaflet 607/1, para 4.3.

(ii) At a given initial airspeed, the peak increment in normal acceleration shall be substantially proportional to the magnitude of the control input.

(iii) For a given magnitude of control input, any variation of peak increment in normal acceleration with initial trimmed airspeed shall be progressively and sufficiently predictable not to cause any handling difficulty.

5.2.3 During aggressive manoeuvres involving significant lifting rotor collective pitch increase, the pilot shall be given unambiguous tactile cues or warnings of an approach to maximum allowable rotor thrust or transmission torque design limits.

5.2.4 The pilot cockpit interface design shall be such that during sustained executions into reduced or negative load factor flight regimes, the pilot has no difficulty in locating and operating critical rotorcraft sub-system controls, selectors and switches etc.

5.2.5 The rotorcraft design shall ensure that all fluid sub-systems such as fuel, hydraulics, and engine/transmission lubrication etc., are sufficiently tolerant of normal acceleration excursions to the maximum, and especially minimum limits of the Permissible Flight Envelope. In particular those sub-systems shall continue to function satisfactorily throughout any manoeuvre required for execution of the mission tasks called for by the Rotorcraft Specification or for recovery from any failure occurring within the Service Flight Envelope.

(i) Any transient interruption to fluid flow shall not be sufficiently sustained to prevent the device being supplied from continuing to function satisfactorily, or to sustain damage.

(ii) Following any brief interruption due to a transient load factor excursion, normal fluid flow conditions shall be restored automatically upon removal of that excursion.

5.2.6 The rotorcraft design shall be such that a single failure of position feedback round an actuator of an automatic control system does no result in a limit cycle developing in which:

(i) Control over the net flight path of the rotorcraft is lost.

(ii) Design load cases (fatigue or ultimate) are exceeded.

(iii) The pilot has difficulty in locating and operating the necessary switch or selector etc., in order to disengage the failed lane of actuation.

5.2.7 The Contractor shall provide adequate guidance on the diagnosis of the failure described in para 5.2.6, and recommend suitable recovery procedures.

5.3 FLIGHT PATH CONTROL

5.3.1 Following a discrete input in the lifting flying control channel, a new rate of climb or descent shall be established smoothly without undue delay and with no objectionable overshoot.

5.3.2 The rotorcraft response to lifting flying control inputs and the flight status data presentation to the pilot shall be such that they cause no difficulty to the pilot in making small adjustments to the rate of climb or descent in order to maintain a required flight path accurately under ground control or instrument flight conditions.

5.3.3 The flying control adjustments associated with the requirements of para 5.3.2 shall not call for undue pilot activity or anticipation in the longitudinal, lateral or directional channels in order to maintain the required rotorcraft airspeed and heading.

5.3.4 When automatic transition manoeuvres to and from the hover are carried out under autopilot control:

(i) Their progress shall be sufficiently predictable that normal operation can be readily monitored by the pilot, and any deviations due to system failures recognised.

(ii) In approaches to the hover, reductions in forward speed shall not be scheduled to be completed before reductions in height.

6 CONTROL MARGINS

6.1 There shall be an adequate range of lifting control available in the rotorcraft to meet the general requirements of Chapter 600, para 12, with reference to Leaflet 600/9.

6.2 The installed power margin shall be compatible with the control margins available in each channel. This requirement applies to both maximum and minimum power levels.

6.3 LOW SPEED ENVELOPE

6.3.1 There shall be adequate main rotor collective pitch available to generate the rotor thrust margins called for by the Rotorcraft Specification, or Leaflet 600/9, para 2.2 when the rotorcraft is:

(i) Loaded to the maximum scheduled total weight in the critical configuration.

(ii) Operated in still air at the critical relative atmospheric density.

6.4 FORWARD FLIGHT ENVELOPE

6.4.1 There shall be adequate lifting control available to meet the steady maximum rate of climb, and maximum level flight speed requirements of the Rotorcraft Specification when the rotorcraft is:

(i) Loaded to the maximum total weight, at the most adverse centre of gravity location, in the critical configuration.

(ii) Operated at the critical relative atmospheric density or under the most adverse environmental conditions.

6.4.2 The minimum in-flight main rotor collective pitch setting shall be sufficiently low to allow a steady autorotational flight condition to be established when the rotorcraft is:

(i) Loaded to a minimum practicable operational total weight in the critical configuration.

(ii) Operated at the optimum airspeed/rotor speed combination to allow a safe power-off landing to be made (see also Chapter 605).

6.4.3 The residual engine power delivered at the flight idle setting shall not be sufficiently high to significantly contribute to rotor overspeeding during rapid descent or quick stop manoeuvres executed with the aggressiveness necessary for mission flight phases called for by the Rotorcraft Specification.

LEAFLET 607/1

LIFTING FLIGHT HANDLING QUALITIES

PRIMARY RESPONSE CHARACTERISTICS

1 INTRODUCTION

1.1 The flying control primarily addressed in Chapter 607 is that which controls the force balance in the nominal plane of symmetry of the rotorcraft, producing variations substantially along the direction of the rotorcraft normal body axis, independently of changes in rotorcraft angle of incidence.

1.1.1 This control is readily identifiable as main rotor collective pitch in the case of conventional rotorcraft configurations.

1.1.2 However, since the requirements of Part 6 apply to all types of rotorcraft, and some may exercise direct lift control by means other than rotor collective pitch in certain flight conditions and configurations, the relevant control is referred to by the more general term, lifting control, rather than collective pitch control.

1.1.3 For some rotorcraft configurations and operating conditions the function referred to in para 1.1 may be more conveniently related to a "power" concept, rather than force on an aerodynamic surface. In this present context, wherever appropriate in Part 6, interpretation of the term "lifting control" can be extended to embrace the control of a power source.

1.2 Depending on the particular context, the rotorcraft motion of major interest in Chapter 607 can be described as either:

1.2.1 Vertical, where substantially symmetric flight path or rotorcraft positioning issues are predominant.

1.2.2 Referred to the rotorcraft normal body axis, (mutually perpendicular to the longitudinal and lateral body axes), where "incremental normal acceleration", in manoeuvres for example, is the significant response parameter.

2 CHANGES OF FLIGHT CONDITION

2.1 NORMAL OPERATING STATE

2.1.1 During operation in the Low Speed Envelope, changing flow conditions and interference effects between rotating and non-rotating aerodynamic surfaces, and ground effects on either, may give rise to significant variation in lifting forces acting on the rotorcraft, together with associated moments about each axis. Similarly in forward flight, significant trim changes or manoeuvring may induce flow separation effects, affecting the aerodynamic forces on the rotorcraft.

2.1.2 Chapter 607, para 2.1.1 and 2.1.2 recognise that those disturbances can not be eliminated entirely, but require that they shall not be excessive. More importantly, it is also required that any such disturbances, and the control inputs necessary to contain them, be orderly and predictable having regard to the change in flight condition being carried out. This is particularly relevant in high-workload operating conditions in order to reduce the probability of the pilot being taken by surprise, leading to the development of a delayed over-controlling or PIO situation.

2.2 FAILURE STATES

2.2.1 In the event of total loss of power driving the rotating lifting system, there is no alternative to entering an autorotative flight condition in accordance with emergency procedures laid down for the particular rotorcraft, in order to preserve rotational energy for sustained flight, culminating in a controlled descent.

2.2.2 The appropriate adjustment to rotor collective pitch setting, referred to in Chapter 607, para 2.1.3 will, in general involve a significant reduction, applied with minimal delay. The requirement recognises that the necessary magnitude and rate of application of this collective pitch adjustment may be large as an integral part of the recommended emergency procedure. For good rotorcraft handling qualities, such action in the collective channel should not call for undue activity on the part of the pilot in the other control channels in order to contain unwanted cross-coupled responses. Achievement of this aim may be assisted by carefully designed control mixing or crossfeed. (see Leaflet 607/2, para 3.2).

2.2.3 Under certain forward flight conditions, especially where height loss would be critical, the appropriate initial reaction to the failure described in para 2.2.1 may be to offload the rotor torque and reduce forward speed by applying an aft longitudinal control input to flare the rotorcraft to a nose up angle of incidence. It may also be appropriate to combine some collective pitch reduction with the nose-up flare to avoid inducing local blade stall, but in any case it would still subsequently be necessary to reduce rotor collective pitch setting in order to preserve rotational speed during any sustained autorotative flight condition.

3 ROTORCRAFT HEIGHT HOLDING

3.1 Insofar as rotorcraft aerodynamic characteristics are generally neutral with respect to small changes in height, except when operating within ground effect, specific measures are necessary in order to maintain the rotorcraft at a particular height under realistic operational conditions.

3.2 AUTOMATIC HEIGHT CONTROL

3.2.1 The objective of the requirements in Chapter 607, para 2.2.2 is to reduce the pilot workload in Attentive flight phases by removing the need for him to attend to the lifting flying control channel. The absence of height holding autopilot facilities is likely to result in LEVEL 2 or 3 ratings.

3.2.2 The requirements for holding performance and the operational environment such as referred to in Chapter 607, para 2.2.3 and 2.2.4, together with the criticality of single system failures, will dictate the type of height sensor, degree of redundancy and system philosophy used in order to meet the requirements of Chapter 607, para 2.

3.3 The objective of the requirement in Chapter 607, para 2.2.6 is to ensure that during changes of flight condition, the sensitivity of the lifting control channel and the degree of interaction with other control channels do not lead to any oscillatory rotorcraft behaviour, or difficulty in determining the appropriateness of the lifting control adjustment applied. The requirement is applicable when the pilot is attentive to the lifting flying control channel, and also when the rotorcraft is under automatic flight path control.

4 TRANSIENT RESPONSE CHARACTERISTICS

4.1 GENERAL

4.1.1 In the Low Speed Envelope the fundamental rotorcraft vertical response to a collective pitch control input into a lifting rotor is characterised by a "first order" type of velocity response largely decoupled from the other body degrees of freedom. In practice this basic form is then likely to be degraded by additional lags, delays and non-linearities associated with flight control system actuation and engine power response characteristics. The intention of the requirements is to lay down a framework by means of which deviations from this idealised first-order behaviour can be limited.

4.1.2 In the Forward Flight Envelope vertical motion per se is a complex issue, coupling virtually all of the rotorcraft body degrees of freedom. Under these conditions incremental normal acceleration and the balance of forces along the rotorcraft normal axis become the relevant parameters.

4.1.3 The requirements of Chapter 607, para 2.3 refer to the response to moderate control inputs, the intention here being to cater for the majority of operational cases without being constrained by significant non-linearities likely to be associated with gross manoeuvres.

4.1.4 In the Low Speed Envelope attention can be limited to direct lift control. For conventional rotorcraft configurations this clearly relates to rotor collective pitch. If an alternative configuration includes lift augmentation in addition to main rotor thrust, control over this additional lift should be integrated with rotor collective control in a single inceptor to be operated by the pilot. Where the additional lift is only a secondary contribution this inceptor may still be referred to as a collective pitch controller.

4.1.5 In the Forward Flight normal acceleration arises from the balance between inertial forces and increments in the net lift on the rotorcraft arising from any combination of:

(i) Rotor blade pitch variations or other devices that directly control lift on the rotorcraft independently of the angle of incidence.

(ii) Variations in rotor thrust or lift on airframe and aerodynamic surfaces due to variations in incidence, primarily induced through longitudinal control applications, but also possible as a result of cross coupled response to rotor collective pitch variations.

(iii) Variations in rotor forces due to rotational speed in (transiently) autorotational flight conditions.

4.2 VERTICAL VELOCITY RESPONSE

4.2.1 The vertical velocity response in the Low Speed Envelope can be reviewed against the framework of criteria presented in Leaflet 600/8, para 4.4 to 4.6 inc and figs 3 and 4.

4.2.2 Following a step input in the collective pitch control channel:

(i) The vertical velocity should rise progressively, with no objectionable hesitations or accelerations, to an acceptable level (X_1), within a time (T_1).

(ii) During the rise time in (i) the rate of change of vertical velocity should experience only one local maximum.

(iii) The maximum rate of change in (ii) should occur within a band (TBD) about the mid point of the rise time.

(iv) Subsequent to the completion of the rise time, the response should not vary by more than ± 10% of the reference value (X_1), in the absence of any other control inputs or external disturbances.

4.2.3 For control inputs up to ± 15% of the in-flight collective control range the reference velocity (CX_1) achieved should be substantially proportional to the magnitude of the input.

4.3 INCREMENTAL NORMAL ACCELERATION

4.3.1 The net response to a step control input has two types of constituent:

(i) The immediate effect of a direct change in lift, as referred to in para 4.1.5 (i), which is only subject to the lags inherent in the control actuation system and aerodynamic processes.

(ii) The effects referred to in para 4.1.5 (ii) and (iii) which take a finite time to develop, depending on the rotorcraft angle of incidence transient response characteristics.

4.3.2 The relative magnitudes of the two constituents in para 4.3.1 will vary, depending on which control is applied and the airspeed, rotorcraft loading, etc., at the initial flight condition. The direct contribution of para 4.3.1 (i) will reach an initial peak and then begin to decay while that of para 4.3.1 (ii) is still building up to its own peak. The summation of two such contributions could possibly result in an initial local peak, followed by a delay before the full response develops.

4.3.3 The objective of the requirements in Chapter 607, para 2.3.3 (i) and (ii) is to avoid any such hesitation in the build up of the response described in para 4.3.2, becoming obtrusive and leading to uncertainty on the part of the pilot as to the appropriateness of his specific control input to achieve the intended manoeuvre. Interpretation of Chapter 607, para 2.3.3 (ii) should disregard any high frequency vibratory content superimposed on the basic rigid body response.

4.3.4 The transient normal acceleration response to a step control input should return and stay within 0.1g of datum within a time (TBD) from initiation of that input.

LEAFLET 607/2

LIFTING FLIGHT HANDLING QUALITIES

CROSS-COUPLING

1 INTRODUCTION

1.1 Throughout the Service Flight Envelope, collective pitch inputs to main lifting rotors, and variations in the rotorcraft angle of incidence have effects in virtually all other rotorcraft axes. The significance of these effects varies with flight condition and rotorcraft operating state.

2 MAJOR EFFECTS

2.1 ROTOR TORQUE

2.1.1 In rotorcraft with shaft driven lifting rotors, engine torque reaction on the airframe is predominantly in the yawing sense for conventional rotorcraft configurations, but will also introduce a rolling moment dependent on the fore and aft shaft tilt relative to the rotorcraft body axis.

2.1.2 Rotor rotational speed may vary with torque loading in power-on flight, depending on engine governing characteristics, but becomes a significant degree of freedom in power-off flight conditions.

2.1.3 The torque required to drive the rotating lifting system increases with increased blade collective pitch setting, and is reduced by increasing nose-up incidence of the rotor disk in forward flight, and conversely for the opposite sense.

2.2 BLADE FLAPPING

2.2.1 Rotor coning angle increases with rotor thrust and induces lateral flapping, up on the retreating side, dependent on forward speed.

2.2.2 Nose up flapping is induced by the increased blade lift effects on the advancing side of the disk.

2.2.3 Both the trends referred to in para 2.2.1 and 2.2.2 increase with increased blade collective pitch and nose-up incidence, and conversely for the opposite sense.

2.3 VIBRATORY LOADING

2.3.1 The localised effects of aerofoil section aerodynamic non-linearities , blade dynamics, and wake interaction, etc., encountered at critical combinations of local blade incidence and Mach number give rise to vibratory forces and moments on both rotating and non-rotating elements of the rotor and control systems.

2.3.2 The vibratory loads of para 2.3.1 in combination with elevated mean levels, during manoeuvres involving high load factor and/or high speed are potentially damaging. Consequently rotorcraft design to minimise the adverse effects together with the provision of unambiguous indications to the pilot of their approach, is very relevant to handling quality issues.

3 COMPENSATION

3.1 The effects described in para 2 are fundamental to lifting rotors and, while the onset of the vibratory characteristics of para 2.3 may be delayed by suitable rotor aerodynamic and dynamic design, little can be done to remove at source the couplings described in para 2.1 and 2.2.

3.2 However, the adverse results of these couplings can in general be reduced by means of control mixing or effective autostabilisation in the affected channels.

3.2.1 Direct control mixing, in which compensating increments dependent on the collective pitch inputs are applied to the other control channels, in series with the pilot's own inputs on those channels, has traditionally been widely used. Mechanical simplicity has generally dictated that this approach addresses the more predictable variations between steady state conditions throughout the Service Flight Envelope, some being more "optimised" than others depending on the mission flight phases required by the Rotorcraft Specification.

3.2.2 In general the compensation for transient cross-coupling effects calls for different degrees of control mixing compared with the trim changes in para 3.2.1. In view of the variability of this over the Service Flight Envelope, and the dependence on the aggressiveness of manoeuvres etc. , transient compensation is probably best dealt with as a "disturbance suppression" task for the autostabilisation system in the relevant channels.

3.2.3 Alternative modern control signalling techniques and processes offer potential for widening the scope, and marrying together, the approaches of para 3.2.1 and 3.2.2. But caution must be exercised to avoid over-design, with its attendant potential for increased costs and development time due to system complexity and reliability issues, and pending clear definition of the absolute need while some of the criteria required for detail design remain to be adequately quantified.

3.3 In some instances advantage may be gained in applying certain cross-couplings to achieve pitch stability augmentation under critical operating conditions, with a measure of "dissimilar redundancy".

PART 6 APPENDIX No. 2
FLIGHT AND GROUND HANDLING QUALITIES
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 600: GENERAL REQUIREMENTS AND DEFINITIONS

600	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-1763	AIRCRAFT/STORES CERTIFICATION PROCEDURES
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION

600 1.	MIL-H-8501A	PARA: 1.1, 3.1
	MIL-F-8490D	PARA: 1.1, 3.3
	MIL-C-18244A	PARA: 3.4.2, 3.5
	MIL-D-23222A	PARA: 3.3.5.2.1
	MIL-F-83300	PARA: 3.5

2. SCOPE AND APPLICABILITY

600 2.	MIL-H-8501A	PARA: 3.1.1, 3.1.2

3. ROTORCRAFT STATES

600 3.	MIL-H-8501A	PARA: 3.1.2
	MIL-F-9490D	PARA: 1.2.2, 1.2.3
	MIL-D-23222A	PARA: 3.8.2.2
	MIL-F-83300	PARA: 3.1.5, 3.1.6, 3.1.6.1, 3.1.6.2, 3.1.6.3

4. OPERATIONAL PHASES

600 4.	MIL-H-8501A	PARA: 3.1

5. FLIGHT ENVELOPES

600 5.	MIL-H-8501A	PARA: 3.1.2, 3.5.5
	MIL-C-18244A	PARA: 3.1.1.5
	MIL-D-23222A	PARA: 3.3.7.2.2, 3.8.2.3
	MIL-F-83300	PARA: 3.1.7, 3.1.8, 3.1.9, 3.1.10.3.3

6. AMBIENT ENVIRONMENTAL CONDITIONS

600 6.	MIL-H-8501A	PARA: 3.2.11.2
	MIL-C-18244A	PARA: 3.1.1.5
	MIL-D-23222A	PARA: 3.10.7.5, 3.15
	MIL-F-83300	PARA: 3.1.10.3

7. LEVELS OF HANDLING QUALITIES

600 7.	MIL-C-18244A	PARA: 3.1.1.5, 3.1.1.5.1
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 1.5, 3.1.10, 3.1.10.2

8. BASIC OPERATIONAL REQUIREMENTS

600 8.	MIL-STD-250D	PARA: 3.2
	MIL-H-8501A	PARA: 3.1, 3.2.1, 3.3.19, 3.5, 3.5.8, 3.5.9
	MIL-C-18244A	PARA: 3.1.1.6.4, 3.2.1, 3.5.1, 4.4.3
	MIL-D-23222A	PARA: 3.1.10, 3.3.7.2.1, 3.17
	MIL-F-83300	PARA: 3.5.5.1, 3.7, 3.8.1, 3.8.2, 3.8.10

9. CONTROL CHARACTERISTICS

600 9.	MIL-STD-250D	PARA: 4.1, 4.4
	MIL-H-8501A	PARA: 3.2, 3.3, 3.4, 3.7
	MIL-S-8698	PARA: 3.2.1.3
	MIL-F-9490D	PARA: 3.3, 3.3.3.1.2
	MIL-C-18244A	PARA: 3.1.1.3, 3.1.1.3.7, 3.1.1.3.12, 3.1.1.5.4.1
	MIL-F-18372	PARA: 3.3
	MIL-D-23222A	PARA: 3.3.7.2.1, 3.10.4.1
	MIL-F-83300	PARA: 3.1.11, 3.2.1, 3.2.3, 3.5

10. RESPONSE TO CONTROL INPUTS

600 10.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.2, 3.3, 3.4, 3.5.4.1, 3.5.5, 3.5.9, 3.5.10, 3.5.11
	MIL-F-9490D	PARA: 3.3, 3.3.2
	MIL-C-18244A	PARA: 3.1.1.3
	MIL-D-23222A	PARA: 3.3.7.2.1, 3.7.3.3.1, 3.10.2.3, 3.10.3.3, 3.10.4.1, 3.10.6, 3.10.7.1
	MIL-F-83300	PARA: 3.2.2, 3.2.3.2, 3.2.4, 3.3.2, 3.5.1.3, 3.5.1.4, 3.5.4.1, 3.5.5

11.	STABILITY	
600 11.	MIL-H-8501A	PARA: 3.2, 3.3, 3.6.3
	MIL-D-23222A	PARA: 3.3.5.1.(2), 3.3.5.2.1
	MIL-F-83300	PARA: 3.2, 3.3.3, 3.3.7, 3.5.2, 3.7, 3.8.3
12.	CONTROL MARGINS	
600 12.	MIL-H-8501A	PARA: 3.2.13.3.4, 3.4.1, 3.5.5, 3.5.8, 3.5.9.(e), 3.5.11
	MIL-F-9490D	PARA: 3.2.5.1
	MIL-F-83300	PARA: 3.1.10.3.3, 3.2.3.1, 3.2.3.3, 3.2.5, 3.4.4, 3.8.8
13.	CARRIAGE AND RELEASE OF EXTERNAL STORES AND LOADS	
600 13.	MIL-STD-1763	PARA: 5.1.4.5.4
	MIL-C-18244A	PARA: 3.1.1.5
	MIL-D-23222A	PARA: 3.11.5, 3.11.6
	MIL-F-83300	PARA: 3.1.4, 3.8.4, 3.8.5, 3.8.6

CHAPTER 601: LONGITUDINAL FLIGHT HANDLING QUALITIES

601	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

601 1.	MIL-H-8501A	PARA: 3.1, 3.2
	MIL-F-9490D	PARA: 1.1, 3.3
	MIL-D-23222A	PARA: 3.3.5.1.(2), 3.8.2
	MIL-F-83300	PARA: 3.3.1, 3.5

2. PITCH ATTITUDE BEHAVIOUR

601 2.	MIL-H-8501A	PARA: 3.2.3, 3.2.5, 3.2.10, 3.2.11, 3.2.14
	MIL-F-9490D	PARA: 3.3.2.1
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.1

3. AIRSPEED CONTROL

601 3.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.2.8, 3.2.9, 3.2.10, 3.2.12, 3.5.9
	MIL-F-9490D	PARA: 3.1.2.7, 3.3.2.6
	MIL-F-18372	PARA: 3.3.1.1
	MIL-D-23222A	PARA: 3.8.2

4. OPERATIONS IN THE LOW SPEED ENVELOPE

601 4.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.2.2, 3.2.5, 3.2.9, 3.5.9
	MIL-F-9490D	PARA: 3.1.2, 3.3
	MIL-C-18244A	PARA: 3.1.1.3.1, 3.1.1.5.4.1
	MIL-D-23222A	PARA: 3.8.2

5. MANOEUVRES IN THE FORWARD FLIGHT ENVELOPE

601 5.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.2.4, 3.2.5, 3.2.8, 3.5.9
	MIL-F-9490D	PARA: 3.3.2.1
	MIL-C-18244A	PARA: 3.1.1.5.4.2
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.2

6. CONTROL MARGINS

601 6.	MIL-H-8501A	PARA: 3.3.4, 3.3.5, 3.3.6, 3.3.18, 3.5.9.(e)
	MIL-F-9490D	PARA: 3.2.5.1
	MIL-C-18244A	PARA: 3.1.1.3.5
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.1, 3.2.3.3

CHAPTER 603: DIRECTIONAL FLIGHT HANDLING QUALITIES

603	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F -18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
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1. INTRODUCTION

603 1.	MIL-H-8501A	PARA: 3.1, 3.3
	MIL-F-9490D	PAR.A: 1.1, 3.1
	MIL-D-23222A	PARA: 3.3.5.1.(3), 3.8.2
	MIL-F-83300	PARA: 3.3.7, 3.5

2. YAWING MOTION BEHAVIOUR

603 2.	MIL-H-8501A	PARA: 3.3.7, 3.3.9, 3.3.10, 3.3.19, 3.5.9
	MIL-F-9490D	PARA: 3.1.2, 3.3
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.1

3. SIDESLIP BEHAVIOUR

603 3.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.3.9, 3.3.16
	MIL-F-18372	PARA: 3.3.1.1, 3.3.1.3
	MIL-D-23222A	PARA: 3.8.2

4. OPERATIONS IN THE LOW SPEED ENVELOPE

603 4.	MIL-H-8501A	PARA: 3.3.5, 3.3.6, 3.3.7, 3.3.9
	MIL-D-23222A	PARA: 3.8.2

5. MANOEUVRES IN FORWARD FLIGHT ENVELOPE

603 5.	MIL-STD-250D	PARA: 4.4
	MIL-H-8501A	PARA: 3.3.8, 3.3.9, 3.3.17, 3.5.9
	MIL-F-9490D	PARA: 3.1.2, 3.3
	MIL-C-18244A	PARA: 3.1.1.5.4.2
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.2

6. CONTROL MARGINS

603 6.	MIL-H-8501A	PARA: 3.3.18, 3.5.9.(e)
	MIL-F-9490D	PARA: 3.2.5.1
	MIL-D-23222A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.1, 3.2.3.3

CHAPTER 604: AUTOMATIC FLIGHT CONTROL SYSTEMS

604	MIL-STD-882	SYSTEM SAFETY PROGRAM REQUIREMENTS
	MIL-STD-1629	PROCEDURES FOR PERFORMING A FAILURE MODE, EFFECTS AND CRITICALITY ANALYSIS
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F -83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. INTRODUCTION

604.1.	MIL-F-9490D	PARA: 1.1
	MIL-C-18244A	PARA: 3.1.1.1, 3.1.1.5.4
	MIL-F-18372	PARA: 1.1, 3.3

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604 2.	MIL-H-8501A	PARA: 3.5.9
	MIL-S-8698	PARA: 3.2.1.3
	MIL-C-18244A	PARA: 1.1
	MIL-F-83300	PARA: 3.2.3

3. PERFORMANCE AND DESIGN

604 3.	MIL-H-8501A	PARA: 3.5.9
	MIL-F-9490D	PARA: 3.3 4.2
	MIL-C-18244A	PARA: 3.1.1.2.1, 3.1.1.3, 3.1.1.5, 3.1.1.6, 3.5.1
	MIL-D-23222A	PARA: 3.3.5.1, 3.8.2

4. ENGAGEMENT AND DISENGAGEMENT

604 4.	MIL-H-8501A	PARA: 3.5.8, 3.5.9
	MIL-C-18244A	PARA: 3.1.1.3.1, 3.1.1.5.4.1.1
	MIL-D-23222A	PARA: 3.7.3.3.1
	MIL-F-83300	PARA: 3.5.4

5.	CONTROLLABILITY	
604 5.	MIL-H-8501A	PARA: 3.5.9.(d)
	MIL-F-9490D	PARA: 3.1.3.2, 3.1.4.(d), 3.1.9, 3.2, 3.2.1.4.2, 3.3.51
	MIL-F-18372	PARA: 3.1.1.1.1
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6.	SAFETY ASSESSMENT	
604 6.	MIL-F-9490D	PARA: 4.1, 4.2
7.	FAILURE INDICATION	
604 7.	MIL-F-9490D	PARA: 3.1.10, 3.2.1.4.2
	MIL-C-18244A	PARA: 3.1.1.3.12.1, 3.2.2
8.	TESTING	
604 8.	MIL-T-8679	PARA: 3.1.92
	MIL-F-9490D	PARA: 4.1, 4.3, 4.3.3, 4.3.4
	MIL-C-18244A	PARA: 3.5.4, 3.5.5, 3.5.6, 3.5.8, 4
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604 12.		
13.	TIMES AND PERIODS	
604 13.	MIL-F-83300	PARA: 3.8.8.1

14. SAFETY ASSESSMENT, ANALYSES, AND TESTS
604 14. MIL-STD-882B PARA: 1.3, 4

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20. AFCS FAILURE AND EFFECTS ANALYSIS
604 20. MIL-STD-1629A PARA: 1, 4, 5

21. ANALYSIS OF THE CONSEQUENCES OF FAILURE
604 21. MIL-STD-1629A PARA: 5
MIL-F-83300 PARA: 3.1.10.2.1

22. FINAL REVIEW (AFCS ANALYSIS)
604 22.

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604 24. MIL-T-8679 PARA: 3.1.9.2
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MIL-C-18244A PARA: 3.5.4, 3.5.5, 3.5.6, 3.5.8, 4.2
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28. SPECIAL TEST EQUIPMENT
604 28. MIL-C-18244A PARA: 3.5.4

29. METHOD OF TEST
604 29.

30. ROTORCRAFT RESPONSE TIMES
604 30. MIL-F-83300 PARA: 3.2.3.2

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CHAPTER 605: ROTOR SPEED CONTROL

605	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. INTRODUCTION

605 1.	MIL-H-8501A	PARA: 3.1
	MIL-F-9490D	PARA: 3.3
	MIL-D-23222D	PARA: 3.3.5.2.1.(1), 3.8.2
	MIL-F-83300	PARA: 3.5

2. ROTOR SYNCHRONISATION

605 2.	MIL-F-83300	PARA: 3.1.6.2.1, 3.1.10.2.1

3. POWER-ON FLIGHT

605 3.	MIL-H-8501A	PARA: 3.7.3
	MIL-S-8698	PARA: 3.6.4, 3.10, 3.10.8
	MIL-F-9490D	PARA: 3.2.1.2.2
	MIL-F-18372	PARA: 3.3.1.2.1
	MIL-F-83300	PARA: 3.1.10.2.1, 3.8.8

4. POWER-OFF FLIGHT

605 4.	MIL-H-8501A	PARA: 3.5.4.3, 3.5.5.1, 3.5.6, 3.5.7, 3.5.8.(e)
	MIL-S-8698	PARA: 3.2.2.4, 3.2.4.1
	MIL-F-9490D	PARA: 3.2.5
	MIL-F-83300	PARA: 3.8.8, 3.8.9, 3.8.10.2

CHAPTER 606: GROUND AND WATER HANDLING QUALITIES

606	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-T-81259	TIE-DOWNS, AIRFRAME DESIGN, REQUIREMENTS FOR
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. INTRODUCTION

606 1.	MIL-D-23222A	PARA: 3.3.5.1, 3.7.2, 3.8.2

2. GROUND OPERATIONS

606 2.	MIL-H-8501A	PARA: 3.1, 3.3.1, 3.5
	MIL-D-23222A	PARA: 3.3.5, 3.3.5.1, 3.7.2.1, 3.12.7.2, 3.12.14
	MIL-F-83300	PARA: 3.1.1, 3.6.5.1, 3.6.5.3, 3.6.6, 3.6.7, 3.7

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606 3.	MIL-H-8501A	PARA: 3.3.1
	MIL-T-81259	PARA: 3.1, 3.2
	MIL-D-23222A	PARA: 3.3.5.1, 3.7.2.1, 3.12.8.3

4. WATERBORNE OPERATIONS

606 4.	MIL-H-8501A	PARA: 3.5.5
	MIL-D-23222A	PARA: 3.3.5.1, 3.9

CHAPTER 607: LIFTING FLIGHT HANDLING QUALITIES

607	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT, AND FACILITIES
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

607 1.	MIL-H-8501A	PARA: 3.1, 3.4
	MIL-F-9490D	PARA: 1.1, 3.3
	MIL-D-23222A	PARA: 3.3.5.1.(1),3.8.2
	MIL-F-83300	PARA: 3.2.5, 3.5

2. VERTICAL MOTION BEHAVIOUR

607 2.	MIL-H-8501A	PARA: 3.4.1, 3.4.2, 3.4.3, 3.5.5, 3.5.5.1
	MIL-F-9490D	PARA: 3.1.2, 3.3, 3.3.2.3
	MIL-F-18372	PARA: 3.3.1.2
	MIL-C-18244A	PARA: 3.1.1.5.4.1.1
	MIL-D-2322A	PARA: 3.8.2
	MIL-F-83300	PARA: 3.2.3.2

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607 3.	MIL-H-8501A	PARA: 3.4.3
	MIL-F-9490D	PARA: 3.1.11
	MIL-F-83300	PARA: 3.8.7

4. OPERATIONS IN THE LOW SPEED ENVELOPE

607 4.	MIL-H-8501A	PARA: 3.1.2
	MIL-F-9490D	PARA: 3.1.3.9.2, 3.1.4.(d), 3.1.9.6(d),3 3.2.3
	MIL-C-18244A	PARA: 3.1.1.3.1
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607 5.	MIL-STD-250	PARA: 4.9
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	MIL-H-8501A	PARA: 3.4.1, 3.5.10, 3.5.11, 3.6.3
	MIL-F-9490D	PARA: 3.2, 3.2.1.2
	MIL-D-23222A	PARA: 3.8.2, 3.12.12
	MIL-F-83300	PARA: 3.1.10.2.1, 3.8.1

6. CONTROL MARGINS

607 6.	MIL-H-8501A	PARA: 3.5.9(e)
	MIL-F-9490D	PARA: 3.2.5.1.
	MIL-D-23222A	PARA: 3.8.2.
	MIL-F-83300	PARA: 3.2.3.1, 3.2.3.3, 3.2.5.1, 3.2.5.2

PART 7

INSTALLATIONS

CONTENTS

*To be written
** In Preparation

CHAPTER 700

PROPULSION SYSTEM INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this Chapter are applicable to the design and construction of all turbine engine propulsion system installations including all the equipment and sub-systems necessary for the satisfactory operation of the engine.

1.2 The rotorcraft designer shall be responsible for the propulsion system installation and shall be responsible for carrying out liaison with the engine and equipment manufacturers to ensure compatibility of the operating characteristics and limitations of all equipment included in the propulsion system.

1.3 For each propulsion system a failure mode and effects analysis (FMEA) shall be completed to ensure that no single failure or probable combination of failures will jeopardise the safe operation of the rotorcraft. (Failure of structural elements need not be considered if the probability of such failures is extremely remote as defined in Chapter 117 (in preparation - JAC Paper 1169)).

1.4 Components in the propulsion system not approved as part of the engine shall be designed in accordance with Chapter 200 and graded as defined in Chapter 400, para 2.

1.5 Each Auxiliary Power Unit (APU) must meet the applicable provisions of this Chapter.

1.6 The composition of an Engine Change Unit (ECU) shall be as detailed in the engine specification.

1.7 Where the requirements of this Chapter impose severe and unavoidable penalties or constraints on the rotorcraft design, their applicability shall be decided by the Rotorcraft Project Director who will carry out the necessary consultations.

2 GENERAL

2.1 The propulsion system installation shall be designed to ensure that no adverse engine characteristics such as stall, surge or flame out are induced to a hazardous degree during normal and emergency operation within the range of operating limitations of the rotorcraft and the engine.

2.2 Precautions shall be taken in the power plant installation to ensure that there are no significant adverse effects resulting from vibration generated by the engine, gearboxes and other relevant rotating machinery. If flexible mountings are used to isolate such vibrations, the maximum deflections of such mountings shall be taken into account in the design of the relevant propulsion unit components.

2.3 The engine air inlet system shall not cause vibration harmful to the engine as a result of airflow distortion during normal operation within the operating limits of the rotorcraft and the engine.

2.4 CONTROLS (See also Chapter 107, Table 5)

2.4.1 Each engine power or thrust control must provide positive and immediately responsive means of controlling its engine.

2.4.2 Each control must have sufficient strength and rigidity to withstand operating loads without failure and without excessive deflection, backlash or creep.

2.4.3 The range of operation of each control device which is operable by the crew shall have the range of movement limited by positive stops coinciding with the 'on' and 'off' positions of the control device and forming an integral part of the control device itself.

2.4.4 Where flexible controls are used their characteristics shall be suitable for the environmental conditions applicable.

2.4.5 Cocks and fluid control valves and devices shall be positive in operation, easy to operate and shall not alter their settings or creep as a result of vibration or changes in environmental conditions. The device and any operating levers and linkages shall be designed so that they cannot be incorrectly fitted in relation to the direction of flow or correct mode of operation.

2.5 INSTRUMENTATION

2.5.1 The design of the propulsion system installation shall enable the applicable instrumentation requirements of Chapter 105 and 107 to be met.

2.5.2 Additional instrumentations or indicators which are necessary for use by the crew because of unusual features of the propulsion system shall be provided.

2.5.3 The use of instrumentation lines carrying flammable fluids under pressure shall be minimised and such lines shall be installed so that escape of fluid will not create a hazard. Such lines shall also have restricting orifices or similar devices to prevent excessive fluid loss if the line fails.

2.5.4 Special consideration shall be given in the FMEA to ensure that no single failure or likely double failures adversely affects the instrumentation necessary for safe control of the engine and propulsion unit system. This may involve special routing or duplication of electrical wiring if electronic surveillance systems are used.

2.6 FLEXIBILITY

2.6.1 Where relative movement between components within the propulsion system and between such components and the rotorcraft can occur, there shall be adequate provision for flexibility.

2.6.2 Fluid lines under pressure which are subjected to relative movement shall employ flexible hose assemblies or equivalent means. Such flexible hoses and their couplings shall be of an approved type suitable for the particular application.

2.7 INTERCHANGEABILITY

2.7.1 All engine change units of a particular Mark Number shall be interchangeable on the type(s) of rotorcraft for which they have been approved and shall comply with the requirements of Chapter 805.

2.7.2 All parts and components of the propulsion system and APU shall comply with the requirements of Chapter 805 and strict attention shall be paid to the accurate location of all accessory attachment points which could affect interchangeability.

2.7.3 To ensure satisfactory interchangeability on multi-engined rotorcraft, the propulsion system installation shall be such that an engine change unit can be installed in any engine position without any major alteration or change of equipment. If different Mark numbers of the same engine change unit are approved for installation in a given rotorcraft, the additional work in changing from one mark number to the other must not result in the total engine change time exceeding the specified time limit (see para 7.1.4).

2.8 CORROSION (see also Chapter 407)

2.8.1 The propulsion system and its associated equipment shall be adequately resistant to corrosion throughout the range of environmental conditions in which the rotorcraft is specified to operate. This corrosion protection shall take into account likely foreign contaminants (e.g. rain, salt water spray, cleaning and de-icing fluids, hydraulic fluid and engine oil leakage and industrial pollution).

2.9 BONDING AND SCREENING

2.9.1 The propulsion system shall comply with the bonding and screening requirements of Chapter 708. Particular care shall be taken to ensure that the essential electrical/electronic control systems are not hazarded as a result of lightning strike.

3 ENVIRONMENTAL

3.1 TEMPERATURE

3.1.1 The propulsion system shall be designed for satisfactory operation over the temperature range defined in Chapter 101, fig 1 up to the maximum height specified for operation of the rotorcraft.

3.1.2 Account shall be taken of the effect of solar radiation with the rotorcraft being parked in direct sunlight.

3.2 ICING

3.2.1 The propulsion system shall operate throughout the flight power range established for the rotorcraft without accumulation of ice that would adversely affect rotor performance, operation of the engine or any system essential for continued safe flight.

3.2.2 The propulsion system design shall take account of ice and slush accumulations building on the airframe and shedding.

3.3 HAIL

3.3.1 The impact of hail over the range of airspeeds and altitudes envisaged for operation of the rotorcraft shall not cause adverse effects so as to affect safe operation (see Table 2).

3.4 RAIN, SNOW AND SLUSH

3.4.1 Ingestion of rain, snow and water/slush from airfield surfaces shall not cause significant adverse effects. Particular attention shall be paid to the effect of such ingestion on engine power or thrust levels during take-off and hover near the ground.

3.5 BIRDS

3.5.1 The engine air intake and other propulsion systems essential for safe flight shall not be adversely affected by bird strike and shall have resistance to bird strikes as listed in Chapter 206, para 3.

4 PROPULSION SYSTEM FEATURES

4.1 AIR INTAKE

4.1.1 The air intake assembly will normally be designed by the rotorcraft constructor but may be provided by the engine constructor.

4.1.2 The air intake shall be designed to ensure suitable airflow distribution into the engine over the range of engine operating conditions, altitude and rotorcraft attitude permitted for operation. Ingestion of gases or pressure waves from guns, rockets and missiles shall be taken into account (see Leaflet 710/3).

4.1.3 The air intake and any equipment fitted in the intake shall be designed to minimise the formation of ice or build up of snow in or near the intake which could adversely affect the engine or APU operation if ingested.

4.1.4 The air intake assembly shall have sufficient strength to withstand any engine surges likely to occur over the operational range of the rotorcraft including those due to inadvertent crew mishandling.

4.1.5 Particular attention shall be paid in the design of the complete air intake assembly including all fasteners to minimise the possibility of generating foreign objects (including ice and slush accretion) which could adversely affect engine or APU operation if ingested.

4.1.6 Any equipment fitted in the intake assembly (e.g., guards and filters) shall have the minimum number of components which could enter the engine if they become detached.

4.1.7 Features in the complete intake assembly into which foreign objects can be trapped and subsequently released into the engine shall be avoided or easily inspected.

4.1.8 Hazardous quantities of fuel or flammable fluid leakage or discharge shall be prevented from entering the intake. Failures of pipes or components shall be considered.

4.1.9 Damage to the air intake assembly caused by bird strikes as defined in Chapter 206 shall not cause structural damage which would have hazardous effects on engine operation by causing detached debris being ingested.

4.1.10 Provision shall be made for fitting removable guards for use during ground running on intakes where a hazard exists to Maintenance Personnel. Attachment features of such guards shall not enter the engine in case of failure or malfitting

4.1.11 Provision shall be made for fitting intake blanks during prolonged ground parking or storage. It must be obvious to Maintenance Personnel when these blanks are fitted.

4.2 EXHAUST SYSTEM

4.2.1 The responsibility for the design of the exhaust system and its mounting provisions in the rotorcraft shall be defined and agreed between the airframe and engine constructors.

4.2.2 The exhaust system shall be designed such that exhaust gases do not impinge on any part of the rotorcraft or on stores or weapons carried on the rotorcraft unless adequate protection is provided. Deflection of exhaust gases by crosswinds etc., during ground manoeuvring shall be taken into account. The design shall prevent significant hot gas re-ingestion into the engine for all flight and ground conditions.

4.2.3 Thermal protection and protection from sonic fatigue shall be provided on the rotorcraft as necessary.

4.2.4 The specification will state if noise suppression devices are required.

4.3 ACCESSORY GEARBOXES

4.3.1 Where accessory gearboxes are provided for engine driven accessories and are not included in the engine approval testing and documentation, then a test programme must be agreed with the Rotorcraft Project Director.

4.3.2 For equipment not approved as part of the engine, failures in the gearbox and gearbox mounted accessories shall not require the engine to be shut down (e.g., by providing suitable disconnects or shear necks).

4.4 ROTORS

4.4.1 Engine power absorption at any rotor rotational speed shall not exceed the limits for which the rotor is approved under all normal and emergency operating conditions approved for the rotorcraft.

4.5 ENGINE REFRIGERANT INJECTION SYSTEMS

4.5.1 If an engine refrigerant injection system is required, this must provide the flow rates at the pressures defined by the engine manufacturer, due allowance being made for likely deterioration in system components due to service use.

4.6 SUDDEN ENGINE SEIZURE

4.6.1 Specific design precautions shall be taken to ensure that no hazard to continued safe flight is caused by sudden engine seizure. In the absence of more accurate data applicable to the particular engine type, a seizure time of 0.5 second from max RPM to rest for any engine spool shall be assumed.

4.7 ENGINE DISC BURST

4.7.1 The propulsion system design shall take into account the possibility of burst of engine compressor and turbine discs. An analysis of the effects of disc burst shall be made. For guidance in making the analysis refer to ACJ No. 2 to JAR 25.903 d(1).

4.7.2 The design aim shall be to ensure that continued safe flight shall not be prejudiced and that no injury be caused to the air crew.

5 RELATED SYSTEM FEATURES

5.1 DRAINS

5.1.1 Flammable fluids which are discharged during normal operation of the engine and APU including shut down on the ground and in flight, shall be suitably disposed of so that no contamination of the rotorcraft or external surfaces is caused.

5.1.2 Readily accessible service drain points shall be provided to enable flammable fluid systems to be safely drained in the normal ground attitude until the residue is of insignificant magnitude. During drainage there shall be no leakage that would be a fire or corrosion risk.

5.1.3 Drainage points shall have positive manual or automatic locking in the closed position which shall be positively identified.

5.2 COOLING AND VENTILATION SYSTEM

5.2.1 The cooling and ventilation system shall ensure that the relevant temperature limits for all propulsion system components and fluids are not exceeded during all normal and emergency rotorcraft operating conditions including ground operations and after engine shut down.

5.3 FUEL HEATING

5.3.1 The fuel supply system must ensure that the fuel flow required by the engine is not impaired by freezing, waxing or solidification of water in the fuel. The level of water contamination catered for shall include such water normally in the fuel plus the maximum condensation that can occur, but credit may be taken for mandatory water drain maintenance checks and for fuel treatment by FSII, if applicable.

5.3.2 If the engine fuel system is not inherently self protecting, the fuel heating means shall be automatic but the crew shall have means to override the automatic system.

6 OPERATIONAL

6.1 STARTING[1]

6.1.1 Reliable and, when required, rapid starting of the engine(s) over the range of climatic conditions defined in the rotorcraft specification must be provided using simple crew procedures.

6.1.2 The rotorcraft designer shall carry out such liaison with the engine constructor to ensure inputs provided to the engine starter are adequate to meet the operational requirements and do not exceed limits established for the engine. (These limits may include pressure, temperature, maximum torque, shock torque, electrical current and voltage as appropriate to the type of starting system used).

6.1.3 Starter type, starting times will be stated in the rotorcraft specification.

6.1.4 For single, engined rotorcraft the system must be capable of providing 3 "starts" (two attempts plus one successful) under the most adverse conditions and permit the rotorcraft to be fit for immediate take-off.

6.1.5 For multi-engined rotorcraft, the system must comply with para 6.1.4 for the first engine followed by two attempts plus one successful start on the remaining engines, but for this phase cross-bleed of electrical power or compressor bleed as appropriate may be utilised.

6.1.6 Starter oil systems shall be self contained and be capable of using all oils approved for use on the engine.

6.1.7 Gas turbine starters shall be capable of operation on all fuels supplied for use by the main engine.

6.1.8 Exhaust from starters shall not cause contamination of the air conditioning system nor cause damage to rotorcraft components or structure.

6.2 RELIGHTING

6.2.1 Each propulsion system must be equipped with an ignition system providing a satisfactory relight capability over the operating envelope of the rotorcraft.

6.2.2 The procedure for engine shut down in flight and relight must be simple.

6.2.3 The relight envelope shall apply irrespective of the time the engine has been shut down but the altitude and air speed may take account of the effect on rotorcraft performance of having had engine shut down.

6.2.4 The design aim shall be that there is the sane relight capability with one igniter inoperative, but a reduced relight envelope may be accepted subject to agreement of the Rotorcraft Project Director.

6.3 NEGATIVE ACCELERATIONS

6.3.1 No hazardous malfunction or damage shall be caused to the engine or any propulsion system when the rotorcraft is operated at the negative accelerations defined in Chapter 202 including the resultant accelerations arising from any practical combinations of normal, lateral and longitudinal positive or negative accelerations. In particular, the system shall be capable of :

(i) Three consecutive periods of normal acceleration less than 1g, each of 4 sec duration with an interval of 3 sec between them.

(ii) A normal acceleration into negative g combined with any practical combination of longitudinal and lateral acceleration.

6.3.2 Propulsion systems limits shall not be exceeded during application of the defined negative accelerations and no unacceptable distractions shall be caused to the flight crew by abnormal functional or instrumentation characteristics.

6.3.3 The effects of negative accelerations beyond the specified limits shall be investigated to determine whether suitable warnings and advice are necessary in the Aircrew Manual.

6.4 INFRA-RED RADIATION SUPPRESSION

6.4.1 The rotorcraft specification will define the applicable level of infra-red radiation suppression and the sources of such radiation.

6.4.2 The contractor will reach early agreement with the Rotorcraft Project Director for the means to be adopted to meet the requirements and the test programme to demonstrate compliance.

6.4.3 The Contractor shall, following consultation with the Director RAE, submit to the Rotorcraft Project Director infra-red emission polar diagrams and acquisition range diagrams using typical minimum detectable irradiance (MDI) values taking into account atmospheric attenuation.

6.5 EMISSIONS

6.5.1 If other emission requirements (e.g., visible smoke, noise) are applicable, these will be defined in the rotorcraft specification. The compliance demonstrations necessary must be agreed with the Rotorcraft Project Director.

7 MAINTENANCE AND SERVICING

7.1 ENGINE CHANGE

7.1.1 Rotorcraft shall be designed so that engine unit removal and replacement can be accomplished with minimum disconnections or disturbances of individual components. This is particularly important for buried engine installations.

7.1.2 The propulsion system shall be designed so that the engine change unit provisioned as a spare can be regarded as a single item defined by a Mark Number for supply, repair and modification purposes.

7.1.3 For modular engines, the requirements of DEF STAN 00-971 shall apply.

7.1.4 Removal, replacement and ground testing for flight clearance of an engine change unit shall be accomplished within the time given in the rotorcraft specification. The applicable time must include rectification of any likely damage to special rotorcraft finishes.

7.1.5 Sufficient ground clearance shall be available to permit engine removal from the underside of the rotorcraft if applicable without using pits or rotorcraft jacking and with representative oleo and tyre compression.

7.1.6 On rotorcraft which may be called on to operate from ships, it shall be possible to change engines with the rotors folded and while subject to ±5° roll and pitch movements, unless some other suitable arrangements can be made such as rotor blade removal.

7.1.7 Compliance with para 7.1 shall be demonstrated on an early representative rotorcraft. Service conditions shall be represented as far as reasonably possible including the equipment and techniques utilised.

7.1.8 Particular attention shall be paid to provision of adequate space and access to break points utilised during engine removal and replacement. Engine change unit mounting devices shall be such as to ensure ease and rapidity of disassembly and reassembly.

7.1.9 Break points for controls, auxiliary drives and pipes shall be of approved quick release types. Fluid carrying pipes, except drain pipes, shall be of an approved self sealing type.

7.1.10 Suitable means shall be provided for slinging the engine change unit during installation and removal from the rotorcraft and the stand. If it is necessary to vary the slinging angles during engine change, this must be possible while the engine change unit is suspended from the sling.

7.2 UNIT CHANGE

7.2.1 The propulsion system shall be designed so that it is possible to remove and replace a defined list of engine accessories (e.g., fuel pump(s), fuel control unit, starter motor, igniters, igniter boxes etc.,) without having to remove the engine change unit from the rotorcraft. The removal and replacement times will be agreed with the Rotorcraft Project Director and demonstrated on an early representative installation.

7.2.2 During such removals and replacements there shall be no harmful straining or distortion of any relevant pipelines and controls.

7.3 FILTERS

7.3.1 Filters in the propulsion system shall be accessible for cleaning and shall have a screen or element which is easily removable.

7.3.2 It shall be possible to retain the sediment trapped by the filter for examination and analysis as required.

7.4 ROUTINE SERVICING

7.4.1 Provision shall be made for easy access to all replenishment points for consumables in the propulsion system and for examination of all relevant contents indicators. Provision shall be made for interface points to inject cleaning fluid for compressor washing. It shall be possible to carry out such tasks wearing cold weather and NBC protective clothing and without special tools and without removing engine cowlings. On shipborne rotorcraft these actions must be possible without use of external ladders or platforms.

7.4.2 Provision shall be made for easy access and to defined adjustment points on the propulsion system and to special engine health monitoring provisions and techniques (e.g., intrascope/boroscope, magnetic chip detectors).

7.4.3 The design aim shall be that visual examination of the blading of early compressor stages and final turbine stages is possible without removing panels or provision of special equipment.

7.4.4 Periodic in situ examination and lubrication as applicable shall be possible with a minimum interference with personnel likely to be working on the rotorcraft concurrently.

7.4.5 Suitable provision shall be made for injecting compressor cleaning fluid during routine servicing if this is specified for the engine.

7.5 COWLINGS AND ACCESS PANELS

7.5.1 Engine cowling panels shall be hinged and secured by quick release fasteners which do not require special equipment for locking and unlocking. The lock/unlock position of such fasteners on cowlings and access panels shall be clearly marked.

7.5.2 The number of access panels shall be minimised. Inadvertent non-locking of panels shall be readily detectable by simple visual means. In flight loss of non locked panels shall not cause hazardous damage to the rotorcraft.

7.5.3 Integral supports shall be provided for cowlings when opened to permit maintenance. These supports shall maintain the cowling position in adverse weather conditions e.g., in winds gusting up to 26.8 m/sec (88 ft/sec).

7.5.4 The design aim is that engine control adjustments can be made during engine ground running with cowlings open.

7.5.5 Cowling and access panels shall be robust enough to withstand likely service handling.

7.6 AUTOMATIC RECORDING OF MAINTENANCE DATA

7.6.1 Consideration shall be given to the provision of built in test (BIT) equipment and/or systems which automatically record propulsion system performance data in flight to provide information for subsequent maintenance action.

7.6.2 Likely failures or malfunctions in the equipment provided shall not cause adverse effects upon the instrumentation available to the flight crew.

8 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE (Chapter 112)

8.1 The design aims and requirements of Chapter 112 are applicable to the engine installation and its systems. (For applicability to the Engine see specification DEF STAN 00-971).

8.2 The effects of turbine blade failure and of disc fragmentation in any engine shall be taken into account in relation to the crew, to structure, and to flight and mission critical equipment.

9 CROSS REFERENCE TO OTHER CHAPTERS

9.1 A number of requirements directly related to propulsion systems appear elsewhere in this publication. The most important of these are listed in Table 1 below.

TABLE 1

LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	Paragraph	Subject
100	2	Use of standard equipment
100	7	Prevention of incorrect assembly
100	8	Conditions of operation
107	13	Engine Instruments
107	Para 7 and Table 5	Engine power controls
407	-	Precautions against corrosion
702	-	Fuel Systems
704	-	Hydraulic systems
706	-	Electrical installations
708	-	Bonding and screening - engine installations
711	2.3	Ice protection of engine air intakes
712	2.1	Fire precautions - engine installations
805	-	Interchangeability of components
1001	-	Flight tests
Leaflet 713/2	2.1	Auxiliary power units (fire zones)

TABLE 2

HAIL

In evaluating the effect of hail as prescribed in para 3.3 the following design conditions are applicable.

Max Hail Size (in)	**Altitude (ft)**	**Rotorcraft Speed**	**Number of Hail Impacts per Square Ft of Frontal Area**
2.0	From sea level up to max cruise altitude or 8,000 ft which-ever is lower	Max	2 Hailstones

The hailstones shall be assumed to be spheres with ice having a Specific Gravity in the range of 0.8 to 0.9.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	-	3368

LEAFLET 700/0

PROPULSION SYSTEM INSTALLATIONS

REFERENCE PAGE

DEF STAN 00-970 Volume 1 Leaflets

710/5	.. ..	The effects of firing air weapons on the behaviour of turbine-engined aeroplanes

MOD Specifications

DERD 2024	.. ..	Interchangeability requirements for engine and accessories.
DERD 2036	.. ..	Starters engine, 28V and 112V direct current electric for aircraft gas turbines.
DERD 2171	.. ..	Starters, low pressure pneumatic, for aircraft gas turbine engines.
DTD (RDI) 3009	.. ..	Protection of power plants from corrosion
DTD (RDI) 3900	.. ..	Engine air filters (wet and dry types)
DTD (RDI) 3901	.. ..	Installation requirements for engine air filters.

NGTE Memoranda

M280	.. ..	Current position of engine malfunctioning caused by the use of air weapons.

Defence Standards

00-971	.. ..	General Specification for gas turbine aero engines.
16-10	.. ..	Connections for low pressure air starting of aircraft engines.
47-19	.. ..	Aircraft pipe fittings (inch series) Screw threads hexagon sizes, tapped holes, and '0' ring seals
53-94	.. ..	Protective plugs, and caps for aircraft pipelines.
61-12	.. ..	Wires, cords, and cables electrical (metric units) Part 20: Braids, wire.

CHAPTER 701

REFUELLING AND DEFUELLING SYSTEMS

1 INTRODUCTION

1.1 The requirements of this Chapter govern the design, construction and installation of refuelling and defuelling systems in turbine-engined and piston engined rotorcraft.

1.2 Refuelling and defuelling systems comprise those parts of the rotorcraft fuel system concerned with the on-loading and off-loading of fuel into and out of the rotorcraft tanks.

1.3 The requirements of this chapter primarily relate to ground and shipboard fuelling operations, including helicopter in-flight refuelling from ships (H.I.F.R.)

1.4 Unless specifically excluded from the requirements of the Rotorcraft Specification provision shall be made for:

(i) refuelling with rotors both spread and folded and with engine running and rotors turning, and

(ii) defuelling with rotors both spread and folded. (See also Chapter 802, para 1.4).

1.5 Recommendations relevant to the satisfactory interpretation of the requirements of this chapter are given in the following leaflets:

Leaflet 701/1 Open Orifice Refuelling and Defuelling.

Leaflet 701/2 Pressure Refuelling and Defuelling.

Leaflet 701/3 Testing of Pressure Refuelling and Defuelling Systems.

Leaflet 701/4 Helicopter In-Flight Refuelling from Ships (H.I.F.R.)

2 FUELLING/DEFUELLING METHODS

2.1 Two methods of supplying and removing fuel from the fuel tanks during ground fuelling and shipboard operations are generally featured in Rotorcraft Specifications:

(i) Open orifice fuelling and gravity draining,

(ii) Closed-circuit pressure refuelling and defuelling.

3 OPEN ORIFICE FUELLING

3.1 ROTORCRAFT REQUIRING OPEN ORIFICE REFUELLING

3.1.1 Open orifice refuelling shall be provided on all rotorcraft designed for using aviation gasoline (AVGAS). Open orifice refuelling, if required for other types of fuel will be stated in the Rotorcraft Specification.

3.1.2 The filling arrangements shall be such that it shall be impossible to fill the fuel expansion space in the tanks inadvertently when the rotorcraft is within the normal range of attitudes expected, taking into account ground slope and ship motion.

3.1.3 The internal diameter of the fuel filling orifice shall be:- AVGAS: 55mm min, 60mm max. Turbine fuel: 75mm min. The filler cap, when secured in position, shall provide a fluid tight seal and shall be designed to minimize the probability of incorrect fitting and possible leakage. It shall be so designed and installed so as to eliminate completely the need for tools for its removal and replacement.[2]

3.1.4 The filler cap shall be permanently attached to the rotorcraft by a suitable means of retention. [3]

3.1.5 Means shall be provided for draining away any fuel spilled at the filling points. Spilled fuel shall be drained overboard and not into the rotorcraft structure. Precautions shall be taken to prevent fumes from spilled fuel entering the crew, passenger or freight compartments of the rotorcraft.

3.1.6 Pipe lines, equipment items and structure, within tanks, shall either be protected, or shall be located sufficiently clear of the filler orifice position, so as to preclude damage when the gravity filling nozzle is inserted at all possible angles directly into the tank or filler neck. If a fuel strainer mesh is provided, as an integral item in the filler orifice, its location shall be such as to preclude possible damage by contact with the filler nozzle.

3.1.7 Each filling point shall have clearly marked provision for electrically bonding the rotorcraft to the ground fuelling equipment filling nozzle. [4] (See also para 5.3).

3.1.8 Filler caps, mounting flanges and their attachments shall be designed and located to prevent an electrostatic discharge from the filler cap, mounting flange and attachments within the fuel tank, with particular reference to lightning strikes, either direct strikes or swept strokes. [6]

3.1.9 Gravity refuelling installations shall be capable of receiving fuel from a nozzle relevant to the fuel type being dispensed at flow rates up to the maximum required by the Rotorcraft Specification, without blow back. [1]

3.1.10 The rotorcraft shall be capable of being refuelled within the overall time required by the Rotorcraft Specification for gravity filling, using the specified refuelling equipment and utilising the services of not more than one man per filling point.

4 PRESSURE REFUELLING AND DEFUELLING

4.1 DEFINITIONS

4.1.1 Pressure refuelling is a method of on-loading fuel to the rotorcraft tanks, using an external pumping unit which delivers fuel at a positive pressure from an external source, through a closed line to a refuelling connector on the rotorcraft from which it flows via a pipe system to the rotorcraft fuel tanks.

4.1.2 Suction defuelling is a method of extracting fuel from the rotorcraft tanks through a pipe system to a defuelling connector on the rotorcraft, from which it flows through a closed line to a receiver, when a negative pressure is applied to the defuelling connector by an external pumping unit.

4.1.3 Pressure defuelling refers to the off-loading of fuel by means of the rotorcraft fuel system pumps or by pressure applied to the fuel tank ullage.

4.2 ROTORCRAFT REQUIRING PRESSURE REFUELLING AND DEFUELLING

4.2.1 Pressure refuelling and defuelling shall be provided on all rotorcraft, unless required otherwise by the Rotorcraft Specification.

4.2.2 Pressure refuelling and defuelling systems shall serve all fuel tanks, both internal and external. Drop tanks and other tanks not part of the permanent fuel system shall be refuelled and defuelled from the main rotorcraft system, using the same refuelling connector(s) as the permanent tank(s). However, when the size and location of the tanks are such that this requirement leads to undue complications and expense, reference shall be made to the Rotorcraft Project Director for agreement to fit a connector directly to the tank.

4.2.3 The arrangement of the pressure refuelling and defuelling system shall be such that the design conditions shall be satisfied with rotors spread or folded, and the rotorcraft at any angle relative to the centre line of a ship experiencing the ship motion, amplitudes and accelerations described in Chapter 309, para 2.3.1 (ii) and subject to adequate safeguards, that the distribution of the fuel load is within acceptable limits.

4.2.4 Refuelling connectors shall comply with the requirements of BS C14.[7]. Each refuelling connector shall be capable of accepting the design flow rate defined in the Rotorcraft Specification but each connection shall be capable of accepting not less than 682 litres (150 Imperial gallons/180 US gallons) of fuel per minute. [7]

4.2.5 Provision shall be made at each pressure refuelling and defuelling point for ensuring that electrical bonding between the rotorcraft and the ground fuelling equipment is achievable. [5]

4.3 REFUELLING

4.3.1 The rotorcraft shall be capable of being refuelled within the overall time required by the Rotorcraft Specification, using the specified refuelling equipment and utilizing the services of not more than one man per pressure refuelling connector. The refuelling fuel flow must be substantially greater than the maximum fuel consumption in hover flight thus permitting nett replenishment of fuel tanks during in-flight refuelling. (See also Leaflet 701/4, para 3.1.5.)

4.3.2 The rotorcraft shall be able to accept fuel through the pressure refuelling system, without damage or derangement being caused to the fuel system and fuel tanks, at all flow rates including overflow obtained when fuel is supplied at a controlled steady pressure of 345 kPa ± 34kPa, measured at the inlet to the refuelling connector. (See also Leaflet 701/2, para 5).

4.3.3 When more than one refuelling point is provided to satisfy the maximum overall refuelling rate, all fuel load controls for use by ground personnel shall be situated so as to provide a centralised external control position for fuel loading and unloading. Refuelling/defuelling control from an internal position shall be provided if required by the Rotorcraft Specification.

4.3.4 When not in use each pressure refuelling system entry point connector shall have means to prevent the escape of hazardous quantities of fuel from the system, including an additional sealing cap, where necessary, against failure of the means provided. 5

4.3.5 Fuel flow shut-off devices for pressure refuelling systems shall be designed normally to fail closed on the ground. However, consideration shall be given, when appropriate, to operational aspects of this stipulation, should it be required to adapt the system for in-flight refuelling, when the failure modes shall not endanger the rotorcraft under any operational condition.

4.3.6 Pressure refuelling connectors, which are recessed behind access panels, shall incorporate positive provisions whereby the access doors cannot be physically secured unless the sealing cap is properly installed.

4.3.7 Panels, provided in the rotorcraft skin for access to pressure refuelling connectors and their sealing caps, shall be so designed and installed as to eliminate completely the need for tools for their removal and replacement. 2

4.3.8 Each fuel tank, or tank group, shall be provided with the means to automatically shut off the fuel supply, to prevent the quantity of fuel present from exceeding the maximum quantity approved for that tank.

4.3.9 Adequate safeguards shall be provided to prevent any design condition being exceeded resulting from failure of this means. A means for checking for correct functioning of any parts of the safety provisions must be available for use before or during refuelling.

4.3.10 The design of any safeguard provision shall be such as to preclude the risk of malfunctioning of any of its parts as a result of ice accretion.

4.3.11 The fuel shut-off device(s) shall not produce a surge pressure in the refuelling pipe lines greater than 827 kPa.

Note: If the design of the system limits the surge pressure to a lower value than the maximum quoted above, then the lower value can be used for design purposes.

4.3.12 The design of the surge prevention, or surge arresting means, shall be such that its failure will not cause the fuel shut-off device for any tank to fail to operate, or, in any other manner, cause over pressurisation of any of the fuel tanks.

4.4 DEFUELLING [8]

4.4.1 Provision shall be made for suction defuelling as rapidly as practicable, with a steady pressure of 21kPa below atmospheric maintained at the outlet of the defuelling connector. Alternatively defuelling shall be achieved by pressure generated within the rotorcraft fuel system.

4.4.2 The defuelling system shall allow the draining of any selected combination of tanks. This shall be achieved by either:

(i) a selective defuelling system. (In this case the defuelling connector shall be in accordance with BS C.14 [7] (see para 4.2.4) or

(ii) use of the standard tank drains and suitable ground equipment.

4.4.3 An alternative means of defuelling, accessible after a crash landing, without lifting the rotorcraft shall be provided if the normal means would be inaccessible in this condition.

4.4.4 Where the defuelling connector is not also a pressure refuelling connector and the rotorcraft can be damaged by attempts to pressure refuel at the defuelling point, or that fuel can be inadvertently passed into the tanks through the defuelling pipe lines, means shall be provided to prevent pressure refuelling at this point.

4.4.5 No damage shall occur to the fuel system and tanks or derangement of tank cell flexible liners during defuelling when a controlled steady negative pressure not exceeding 34 kPa below atmospheric, measured at the outlet to the defuelling connector, is applied to the defuelling connector.

4.5 ACCESSIBILITY

4.5.1 The system shall be such that refuelling and defuelling are possible without the necessity for access to the upper surface of the fuselage.

4.6 DRAINING OF REFUELLING AND DEFUELLING PIPELINES

4.6.1 Except where the pressure refuelling, defuelling and engine supply pipelines are largely common, provision shall be made for draining residual fuel from the rotorcraft refuelling and defuelling pipelines, after refuelling or defuelling, unless the quantity of fuel involved is so small that it does not constitute a hazard or increase the vulnerability of the rotorcraft.

4.7 FUEL CONTENTS INDICATION

4.7.1 For the convenient use by ground personnel, an indication shall be provided (in addition to the contents gauges in the cockpit) of when each tank (including drop-tanks and other tanks not part of the permanent fuel system) is filled to the required level.

4.8 SELECTIVE FUEL LOADING

4.8.1 When required by the Rotorcraft Specification means shall be provided for selective fuel loading to any of the fuel loading conditions specified under useful loads in the Rotorcraft Detail Specification. Internal transfer of fuel shall not be necessary to meet this requirement. Where one refuelling installation is utilized to fill several tanks, the arrangement shall be such that repeated topping-up is not required to fill the affected tank to design capacity.

4.9 FUEL MASS LOADING TO A LEVEL CORRESPONDING TO A MASS VALUE

4.9.1 The pressure refuelling and defuelling system shall provide for fuel level control to the design maximum-weight fuel-loadings specified in the Rotorcraft Detail Specification and for any intermediate loading from empty fuel tanks to the maximum fuel-loading. The level control system shall also satisfy the requirements of para 4.3.8, 4.3.9 and 4.3.10, with respect to safety provisions.

4.10 PRESSURE REFUELLING ELECTRIC CIRCUITS

4.10.1 For maximum safety during ground refuelling and defuelling operations, means shall be provided to supply power only to the pressure refuelling and defuelling circuits. These are circuits which must necessarily be energised in order to operate electrical components in the system. Such components include shut-off devices, fuel level sensors and, where necessary, fuel control and gauging devices.

4.10.2 Pressure refuelling circuits shall operate from the rotorcraft batteries except where the total demand during refuelling exceeds 5 per cent of the minimum battery capacity.

4.10.3 Where the demand exceeds 5 per cent of the installed battery capacity, the pressure refuelling circuits shall be energised, where appropriate, form the supply generated by an on-board auxiliary power unit, or through a ground supply socket. In the latter case the socket shall be an approved quick-disconnect explosion-proof type and shall be positioned remotely from the refuelling connector.

4.10.4 In all cases, whether auxiliary power unit-supply, ground-supply or battery-operated, when the pressure refuelling circuits are switched on, they shall be independent of all other rotorcraft circuits so that no other circuits need be energised during refuelling, except in the case of APU supply. When the pressure refuelling circuits are switched off, those components also required during flight shall be automatically transferred from the pressure refuelling circuit to the flight circuit.

4.10.5 On rotorcraft where the pressure refuelling circuits are energised by the rotorcraft batteries, or auxiliary power unit, disconnection of the pressure refuelling nozzle, or closing the access panel, if fitted, shall switch off the pressure refuelling circuits.

4.11 STRENGTH

4.11.1 The pressure refuelling system (excluding fuel tanks and fuel tank vents, see Chapter 702), with the tank shut-off devices closed, shall have proof and ultimate factors not less than 1.5 and 2.0 respectively on the loads produced by the maximum surge pressures that occur during refuelling (see para 4.3.11).

Note: The possibility of failure due to fatigue effects shall be taken into consideration.

4.11.2 The suction defuelling system (excluding fuel tanks and fuel tank vents), with the tank shut-off devices closed, shall have proof and ultimate factors not less than 1.125 and 1.5 respectively on the loads produced by a negative pressure of not more than 34 kPa below atmospheric measured at the defuelling connector.

5 FIRE PRECAUTIONS (see also Chapter 712)

5.1 Adequate precautions shall be taken in the design of rotorcraft systems and components to minimize the risk of fire during:

(i) refuelling and defuelling in hardened shelters and in hangars on board ship,

(ii) refuelling and defuelling in the open when other servicing operations are proceeding involving the use of electrical circuits and apparatus, the enclosures of which may not be of explosion proof design,

(iii) operation of an on-board auxiliary power unit, the proximity of which to the refuelling connector(s) could result in spilled fuel being ignited by it, or ingested into its air intake.

5.2 Pressure refuelling and defuelling electrical circuits and components shall be incapable of creating an ignition source.

5.3 When open-orifice refuelling is provided for, a grounding socket, in accordance with the requirements of BSG175 (see also SBAC Drawing AS6322), shall be fitted at each refuelling point to enable effective bonding to be achieved between the rotorcraft structure or main bonding system, as appropriate, and the refuelling equipment [4].

REFERENCES

Ref	ASCC Air Standard	STANAG Standard	British		MIL-Spec
1	25/11	3212	BS C13		MIL-C-38373
2	-	3294	-		MIL-C-38373
3	-	-	-		MIL-C-38373
4	25/26	3632	BS G175		MIL-C-83413
5	-	-	-	)	MIL-F-38363
6	-	3659	-	)	
7	25/17	3105	BS C14		MS24484
8	-	3334	-		-

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REFUELLING AND DEFUELLING SYSTEMS

REFERENCE PAGE

<u>British Standards</u>

C13	..	..	Specification for sizes of aircraft gravity filling orifices and associated replenishment nozzles (metric series).
C14	..	..	Coupling dimensions for aircraft pressure refuelling connections.
3G100	..	..	Specification for general requirements for equipment in aircraft.
G175	..	..	Aircraft fuel nozzle grounding plugs and sockets

LEAFLET 701/1

REFUELLING AND DEFUELLING SYSTEMS

OPEN-ORIFICE REFUELLING AND DEFUELLING

1 INTRODUCTION

1.1 This leaflet contains recommendations for the design of facilities for open-orifice, gravity filling of tanks in amplification of Chapter 701.

1.2 Although open-orifice refuelling is principally featured on small rotorcraft with relatively simple fuel systems, gravity-filling can provide a readily available safe back-up refuelling method for rotorcraft equipped with a closed circuit pressure refuelling system.

1.3 Access into fuel tanks through standard filler orifices also facilitates defuelling by means of a suction defuelling tube inserted into the tank. This feature of the rotorcraft can prove useful in the case of salvage activity, following a crash landing, when access to the normal suction defuelling connector and tank drains may not be possible.

2 REFUELLING

2.1 The number of open-orifice refuelling points should be kept to a minimum. Where a filling orifice is utilized to fill more than one tank, the interconnecting arrangement should be such that repeated topping-up is not required to fill the affected tanks to design capacity.

2.2 Where the filler orifice incorporates a fuel strainer, the mesh of the strainer, when clean, should allow 1.25 x required maximum flow without any blow back.

2.3 The latching and/or locking mechanism of the filler cap, when secured, should, through its orientation, tend to be held in the latched/locked position by positive normal acceleration and the rotorcraft slipstream in forward flight.

3 DEFUELLING

3.1 For use on rotorcraft provided only with open-orifice fuelling facilities, a special defuelling suction tube is available, designed to avoid damage to the tank linings when inserted into a tank. Attention should be drawn to the use of this equipment in rotorcraft servicing instructions.

LEAFLET 701/2

REFUELLING AND DEFUELLING SYSTEMS

PRESSURE REFUELLING AND DEFUELLING

1 INTRODUCTION

1.1 This leaflet contains recommendations for the design of pressure refuelling and defuelling systems in amplification of Chapter 701.

2 REFUELLING

2.1 The number of refuelling points should be kept to a minimum.

2.2 The standard refuelling connector(s) should:

(i) be located so that connection of the ground fuelling equipment nozzle can be made with relative ease by an average height man while standing on the ground or ship's deck. The use by the operator of Service standard Arctic Gloves and NBC (Nuclear Biological Chemical) clothing should be taken into consideration.

NOTE: Consideration of MIL-STD-1472, Human Engineering Design Criteria for Military Systems should be employed in the design layout of the ground refuelling connector(s) and fuel loading control panel.

An optimum location is considered to be between 1 and 1.5 metres above the ground or deck; the acceptability of more extreme locations will depend on the connector orientation. In the case of naval rotorcraft particular consideration should be given to access when the rotorcraft is parked with rotors spread, or folded, at any angle relative to the centre line of any specified ship, close to the edge of the flight deck.

(ii) be located so as to be reasonably safe from damage liable to be incurred from a crash landing,

(iii) be mounted to the rotorcraft structure such that loads applied to the connector, during operation of the refuelling ground equipment, are not transmitted to the rotorcraft refuelling pipe system.

2.3 As a safety provision, in the event of the refueller being driven away negligently before disconnection is made between the equipment and the rotorcraft, a frangible link may be specified in the Rotorcraft Specification. Where this requirement exists, consideration should be given for the connector mounting provisions to be adequate to react the hose break-away loads in such a manner that separation occurs at "the weak link", without damage to the adjacent fuel pipes and rotorcraft structure. The characteristics of the break-away loads, to be considered, will normally be specified in the Rotorcraft Specification.

2.4 The manner in which the sealing cap for the pressure refuelling connector is retained, or made captive, should be such that:

(i) it does not impede coupling and uncoupling of the refuelling nozzle.

(ii) Where the sealing cap also forms the access panel, or cover, to the connector, the latching mechanism should be such that a positive lock is provided that is readily able to be checked for security. In the event of the sealing cover becoming disconnected during flight, an analysis of the likely effect of the slipstream on the actions of the released sealing cover should be conducted, with the aim of avoiding possible damage to adjacent structure, or hazard to the rotorcraft.

2.5 It should be possible to isolate the engine fuel system from the rotorcraft fuel system during refuelling; and, where pipes are common to both refuelling and engine feed and crossfeed systems, it should be possible to isolate the engine fed system section during refuelling.

2.6 Pressure refuelling systems should be arranged so that the fuel entry point is at, or near, the bottom of the tank, so as to reduce the degree of fuel misting and the level of electro-static charge in the tank during refuelling.

2.7 The fuel entry point into the tank should be so arranged that the flow does not impinge directly on the tank wall, but is adequately diffused to minimise the risk of damage to the fuel cell or structural sealant.

2.8 Pressure refuelling systems, with more than one refuelling point, should be able to supply fuel to all tanks from any one connector. The fuel distribution, in the case of using only one of the available connectors, may result in extending the overall refuelling time. The acceptability to the Rotorcraft Project Director of any reduced rate, in this case, should be verified.

2.9 Where the fuel tank arrangements are such that a given sequence of fuel loading is necessary, or that certain sequences should be avoided, to ensure that the centre of gravity of the rotorcraft is maintained within acceptable limits, an adequate warning placard should be permanently displayed adjacent to the fuel load control panel.

2.10 Fuel loading controls for routine ground servicing should be so arranged that the action of closing the access panel appropriately:

(i) cancels any pre-selected mode of operation,

(ii) isolates the refuel or defuel function mode,

(iii) operates any changeover devices in the rotorcraft fuel system necessary for flight operation.

2.11 Consideration should be given to the effects of possible failure of the pressure control means of the external pumping unit, particularly if its maximum uncontrolled output pressure is likely to exceed the pressure given in Chapter 701, para 4.3.11.

3 DEFUELLING

3.1 When possible, suction defuelling should be designed to be via the same connector(s) as pressure refuelling. Where this is not adopted, the design and location of the defuelling connector(s) should conform to the recommendations given in para 2.1 and 2.2 of this Leaflet.

3.2 In cases where the provision of selective defuelling is by means of the standard tank drains, these should be sufficiently accessible to allow easy use, by ground personnel, of an adaptor hose assembly, one end of which is designed to mate with the tank drain, while the other end can be fitted with a connector to B.S. C14.

3.3 Where a pressure refuelling/defuelling pipe manifold also forms part of the internal fuel transfer system of the rotorcraft this may be capable of receiving fuel under pressure from the rotorcraft fuel transfer pumps. In this case, the rotorcraft pumps and transfer system may be used for defuelling subject to approval of the Rotorcraft Project Director.

3.4 To defuel the rotorcraft after a crash landing, it is acceptable that removal of a reasonable amount of panel, or cowling, may be necessary to reach the defuelling connector; alternatively, break-in panels may be provided for access, if subsequent repair or replacement is simple. Where the normal defuelling connector is not accessible, a second connector complying with B.S. C14 should be provided.

3.5 Consideration should be given to the effects of possible failure to the pressure control device associated with the defuelling ground equipment, which normally limits the negative pressure to 34kPa, malfunction of which could cause negative pressures approaching 100 kPa to be applied to the defuelling connector.

4 FUEL CONTENTS INDICATION

4.1 The indicators required by Chapter 701, para 4.7 should be located adjacent to the pressure refuelling control point, where the display will provide an instantaneous indication to ground personnel of the fuel quantity status, thus allowing them to monitor the operation of an automatic shut-off system, or to manually operate the refuel shut-off devices, as appropriate, and to know when to disengage the pressure refuelling equipment. The fuel level sensing and signalling devices may operate in conjunction with the automatic fuel level shut-off system, which, in turn, may be incorporated in the fuel contents gauging system.

4.2 When the Rotorcraft Specification requires an indication of fuel tank contents, derived from the sensors of the gauging system used for flight, to be displayed at the ground refuelling point, consideration should be given to the effect of any difference in tank configuration between flight and ground conditions, since the gauging system will have been optimized for the flight case. Ground standing conditions, e.g., rotorcraft pitch attitudes may vary from those for which the flight system has been characterised, requiring a difference in gauge calibration to indicate true quantity.

4.3 FUEL CONTENTS BACK-UP INDICATORS FOR GROUND USE

4.3.1 For use by ground personnel, as a back-up for ground servicing contents gauges, the Rotorcraft Specification may require the provision of mechanical fuel level indicator devices. These should be completely independent of other fuel contents gauges. Suitable devices comprise inverted, sealed, dipsticks which, when released from their latched positions, protrude from the bottom of the tank. If these mechanical fuel level indicators are required by the Rotorcraft Specification, the following should be considered:

(i) Number of units.

(ii) Location of units.

(iii) Locking and latching of units should be flush fitted to the rotorcraft skin.

(iv) Calibration and overlap between adjacent units.

(v) Sensitivity.

(vi) Operation without tools.

5 SAFETY PRECAUTIONS

5.1 The specified refuelling pressure of 345 kPa at the inlet to the rotorcraft is potentially dangerous, since in most cases it greatly exceeds that which the tank can withstand. The danger becomes real should the flow of fuel during refuelling be stopped, or reduced, by a back pressure within the tank itself; the most likely cause being overfilling of the tank through failure of the fuel shut-off device. Consideration should also be given to the provision of adequate venting and draining of structural bays, both inside and outside the fuel tanks, through which refuelling pipes pass, to prevent damage to the rotorcraft in the event of pipe system failure.

5.2 Adequate safeguards against failure of the fuel shut-off device are required by Chapter 701 para 4.3.8. Possible methods of providing these safeguards are:

(i) the provision of a system capable of discharging fuel from the tank, should overfilling occur, or

(ii) the employment of a secondary, or back-up, device for fuel shut-off with provision for verifying its serviceability immediately prior to refuelling.

NOTE: Provision of a functional pre-checking system of the primary fuel shut-off device does not itself constitute an adequate safeguard. A secondary device for fuel shut-off, or a discharge system, is also necessary.

5.3 The simpler of the alternatives is a discharge system. Providing the precautions described below are observed, this can be made to give complete structural safety, whatever malfunctioning of the rest of the system occurs (including the failed refuelling pipe condition, referred to in para 5.1 above). However, difficulties may arise should it be required to adapt the system for in-flight refuelling. The other alternative, introducing a secondary device for fuel shut-off, avoids the drawbacks of the discharge system, but leads to complications, since both the fuel level sensing device and the fuel flow shut-off device are involved.

5.4 OVERFILLING FUEL DISCHARGE METHOD (see para 5.2(i))

5.4.1 The discharge system may consist of the normal tank venting system, supplemented, if necessary, by an additional means of fuel discharge. The pressure drop of the system, as a whole, should be such that unacceptable tank pressures cannot arise if overfilling occurs when refuelling at the normal maximum rate, or at an abnormally high rate into the rotorcraft, such as might be associated with a failed pipe system, which is situated within a tank, upstream of a shut-off valve. Such a system can however fail for the following reasons which may not be immediately obvious:

(i) In a rotorcraft which operates in arctic conditions the fuel can become very cold. Following descent, comparatively warm moist air can enter through the normal vent system and deposit dew or form hoar frost, which melts when warm fuel is added. The water, thus formed, can coat the discharge system and freeze on subsequent flights. Should the discharge system contain a valve, U-bend, or other liquid trap, it may then fail through icing.

(ii) Where the discharge system contains a spring loaded valve it can, even if mechanically freed before refuelling, re-seize quickly if sufficient cold fuel remains in the tank after landing.

(iii) Where a removable cover is fitted over a discharge system orifice for flight purposes, the danger exists that it may not be removed before refuelling commences.

(iv) The refuelling rate specified in Chapter 701, para 4.3 can be exceeded in certain cases owing to variation in refueller performance, variation in the pressure drop of nominally identical rotorcraft systems, and the reduced viscosity of warm fuel in hot climatic conditions; significant increase in refuelling rate could result from failure of a pipe upstream of a refuelling shut-off valve, or failure open, of a common surge relief valve, discharging into a convenient tank. Moreover such failures could be dormant. These possible excess rates of refuelling should be considered when the acceptable pressure within the tank is determined.

NOTE: It is possible also that the inward relief valve, or venting system, provided to prevent tank depressions during defuelling, may freeze under similar conditions described above.

5.4.2 The foregoing considerations lead to the following safety recommendations:

(i) when a discharge system is used, the outlet should be positioned to minimise any possible fire hazard,

(ii) the flow capacity and pressure drop of the discharge system should prevent tank pressures exceeding 75% of proof, should overfilling of the tanks occur at any refuelling rate up to the maximum,

(iii) the discharge system should be so arranged that fuel supply to the refuelling system cannot commence unless the discharge system is fully open, or is shown to be capable of fully opening when required,

(iv) precautions should be taken to ensure that, when the fuel supply is cut off, the inertia of the moving fuel in the discharge system and any siphon effect, resulting from the discharge system geometry, cannot cause a dangerous depression within the tank,

(v) where applicable, means should be provided to ensure that the discharge system cannot inadvertently be left open when the refuelling operation is completed,

(vi) the inward venting system, if not permanently open, should be so arranged that fuel removal from the tanks during refuelling cannot commence until an adequate inward vent is open, and

(vii) the effect of deviations from nominal ground attitude, as given in the Rotorcraft Specification, should be considered when positioning the internal orifice of the discharge system, so that spillage during normal refuelling can be avoided. (Naval rotorcraft are subject to the requirements of Chapter 802, para 1.4 in this respect).

5.5 SECONDARY MEANS FOR FUEL SHUT-OFF METHOD (see para 5.2 (ii))

5.5.1 Where any single failure of a component, sub-component or auxiliary system may result in failure to shut off the refuelling flow, a secondary system of shut-off, involving a secondary component, sub-component or basic system, as appropriate, should be provided.

5.5.2 The following should be features of this method of safety provision:

(i) the secondary shut-off means should, if possible, provide the same shut-off level as the primary system,

(ii) where any single failure of the secondary fuel shut-off provision may not be readily evident to the refuelling operator, means should be provided for the operator to check for any single failure prior to, or during, refuelling.

5.6 FUNCTIONAL PRE-CHECKING FACILITIES (see para 5.2 (ii) Note)

5.6.1 The capability to pre-check the functioning of a primary fuel shut-off device is a useful aid in establishing confidence that there is only a remote likelihood of tank vent system flooding, or of fuel spillage from the vents, or that other means of tank protection from overpressure will be required to function.

5.6.2 The need to pre-check the functioning of a secondary, or back-up, fuel shut-off device, however, is an essential provision where the fuel tank overpressure protection method relies on this to function correctly.

5.6.3 Both these facilities should:

(i) incorporate the means for pre-checking, at the commencement of and during each refuelling of each tank, for satisfactory operation all functional parts of the fuel shut-off device for stopping the flow of fuel at the maximum approved level.

(ii) provide evidence, at the refuelling control point, of satisfactory operation of the shut-off device to stop the flow of fuel at the maximum approved level, and also provide indication of any mal-functioning of the fuel shut-off device.

6 REFUELLING THROUGH THE DEFUELLING SYSTEM

6.1 It is recommended that it should be impossible to refuel when defuel is selected.

LEAFLET 701/3

REFUELLING AND DEFUELLING SYSTEMS

TESTING OF PRESSURE REFUELLING AND DEFUELLING SYSTEMS

1 INTRODUCTION

1.1 In order to verify that the pressure refuelling and defuelling systems satisfy the requirements of Chapter 701 it is recommended that the test procedures specified in this Leaflet be carried out. As many of these tests as possible should be performed initially on a fuel system functional mock-up; complete tests should then be performed on the prototype rotorcraft.

2 REFUELLING SYSTEM

2.1 PROOF PRESSURE TESTS

2.1.1 Pipes which are subject to maximum refuelling pressure i.e., upstream of fuel shut-off devices.

(i) Apply suitable blanks at each refuelling system connection to the tanks, so that no fuel can flow into the tank. The manner in which such blanks are fitted should not fundamentally, or significantly, disturb or alter the supports of the pipelines and components.

(ii) Apply the proof pressure, as required by Chapter 701, para 4.11.1, and, whilst maintaining the pressure, examine the pipe system for leaks and distortion in the pipes and their supports.

NOTE: The rotorcraft should be in its nominal ground standing condition, on a level surface. Where the test is performed on a functional mock-up, this condition should be simulated as near as possible.

2.1.2 Pipes situated downstream of fuel shut-off devices.

(i) Pressures generated in refuelling pipes downstream of the shut-off devices are not normally of a high order. Where the need for a proof pressure test is regarded as appropriate, apply a proof pressure based on the maximum internal pressure, calculated to be likely in these pipes. While maintaining this pressure examine the pipes for leakage and distortion of their supports.

2.2 FUNCTIONING TESTS

2.2.1 Normal Refuelling.

NOTE: The rotorcraft should be in the nominal ground standing position on a level surface supported solely on its undercarriage. (In the case of a functional mock-up, this condition should be simulated as near as possible). The tank vent system should be complete, have been subject to functioning tests and be functionally acceptable. Gravity filling caps should be secured.

(i) Starting with all tanks empty, refuel all tanks, initially simultaneously, by supplying fuel at a pressure of 345 kPa maintained at the inlet to the refuelling connector. Whilst the fuel is flowing into the tanks, monitor and record:

(a) pressure generated in each tank (including fuel head where appropriate),

(b) overall time to refuel the rotorcraft,

(c) time between first tank and last tank to fill,

(d) surge pressure produced in the pipelines when each tank shut-off device operates.

(ii) If adjustment or modifications are made to the system to alter the times measured at (b) or (c) above, the whole test procedure should be repeated.

(iii) After all tanks have been filled and shut-off devices have operated to the closed position, maintain the refuelling inlet pressure at 345 kPa for sufficient time to conduct an examination for any leakage past the shut-off devices.

(iv) Starting with all other tanks partially filled to an appropriately representative fuel load condition with respect to acceptable c of g range requirements, and with a fuel pressure of 345 kPa maintained at the refuelling connector, refuel each tank separately, in turn, from empty to full, selecting all other refuel devices shut, or, where necessary, by keeping these devices closed by some artificial means, or by maintaining the other tanks full. Whilst fuel is flowing into the tank monitor and record:

(a) pressure generated in the tank (including fuel head where appropriate),

(b) the quantity of fuel delivered to each tank up to the instant of its refuel shut-off device closure,

(c) surge pressure produced in the pipelines when each tank shut-off device operates.

2.2.2 Overfilling and/or Functioning of Safety Provisions.

(i) Simulate failure of each refuelling shut-off device to close by some artificial means.

(ii) During the test, all shut-off devices, other than the one being tested, should be kept closed by some suitable method. Maintain the fuel supply to the refuelling connector at 345 kPa pressure. Allow fuel to be supplied to the shut-off device and tank being tested until operation of the safety provision for protecting the tank operates.

(iii) Monitor the functioning of the safety provision and record the tank pressures throughout the test procedure, including the maximum peak pressure and pressure during steady overfuelling condition in the case of a fuel overboard discharge protection system.

3 SUCTION DEFUELLING SYSTEM

3.1 PROOF SUCTION TESTS

3.1.1 Blank off the inlet ends of the pipes from the tank(s) to the defuelling system, so that no fuel or air can flow from the tanks to the defuelling pipe system. Where appropriate, operate the defuelling shut-off devices to the open position.

3.1.2 Apply the proof negative pressure, as required by Chapter 701, para 4.11.2, and, whilst maintaining this negative pressure, examine the system for leaks and distortion of the pipes and their supports.

3.2 FUNCTIONING TESTS

3.2.1 Starting with all tanks full (including where appropriate and significant, fuel flooded vent surge tank(s)) and, using a defuelling pump at least equal in performance to the defuelling equipment specified in the Rotorcraft Specification, perform the following defuelling tests:

(i) of the whole tank system,

(ii) of those combinations of tanks, as specified in the Rotorcraft Specification.

NOTE: The rotorcraft should be in the nominal ground standing condition, on a level surface, supported solely by its undercarriage. (In the case of a functional mock-up this condition should be simulated as near as possible). The tank vent system should be complete, have been subject to functional tests and be functionally acceptable. Gravity filling caps should be secured.

3.2.2 While maintaining a steady negative pressure at the defuelling connector not exceeding 34 kPa, operate the defuel devices to the open position, as appropriate, and record:

(i) negative pressure in each tank,

(ii) negative pressure at the outlet to the defuelling connector,

(iii) minimum overall time to defuel the rotorcraft, or selected combination of tanks, as applicable,

(iv) residual fuel in each tank when air enters the system and significant fuel flow ceases.

3.2.3 In performing these tests care should be taken not to damage the tank structure, or derange tank cells, by negative pressure. The tests should start with low defuelling rates being induced, which should then be increased in suitable increments until the tank negative pressures fall to the safety limit, or the negative pressure at the rotorcraft defuelling outlet reaches 34 kPa (should this occur before the maximum performance flow rate of the defuelling pumping equipment is reached). The relationship between negative pressure at the rotorcraft defuelling outlet and fuel flow can then be obtained and related to the defuelling performance curves for service ground fuellers The expected defuelling time, using Service equipment, can then be determined.

LEAFLET 701/4

REFUELLING AND DEFUELLING SYSTEMS

HELICOPTER IN-FLIGHT REFUELLING FROM SHIPS (H.I.F.R.)

1 INTRODUCTION

1.1 This leaflet contains recommendations for the design of pressure refuelling systems when required to be compatible with the operational conditions associated with refuelling rotorcraft in hover flight from naval vessels.

1.2 Helicopter In-Flight Refuelling (H.I.F.R.) Equipment comprises a special hose assembly carried on board a ship, the upper end of which, when raised by the rotorcraft rescue hoist to a point adjacent to the rotorcraft pressure refuelling connector and coupled to it, connects the ship's aviation fuel supply point to the rotorcraft pressure refuelling system.

2 REFUELLING EQUIPMENT DEPLOYMENT

2.1 In operational use the special hose assembly of the H.I.F.R. equipment is shackled adjacent to its lower end to a suitable deck attachment point on the ship, and the lower end of the hose connected to an outlet from the ship's aviation fuel supply. Adjacent to the upper end of the special hose assembly is a suitable lifting eye and the upper end of the hose carries a pressure refuelling coupling and manually controlled ON/OFF valve suitable to mate with the rotorcraft refuelling connector. Other features of the special hose assembly include a self-sealing emergency break-away coupling between the lifting eye and the attachment point on the ship.

2.2 In order to receive fuel from the ship the rotorcraft is positioned above the ship's deck and the rescue hoist cable and hook lowered by the flight crew to be caught by the ship's deck crew, using a grounding wand to dissipate any static charge between the rotorcraft and the ship. The hook is then connected to the lifting eye of the special hose assembly by the ship's deck crew. The hose is raised and captured by the flight crew, who manually connect the hose end coupling to the rotorcraft pressure refuelling connector. When the manually controlled ON/OFF valve is opened fuel can be received into the rotorcraft refuelling system.

2.3 When receiving fuel the rotorcraft is repositioned at a maximum of 15m (49 feet) above the ship's deck and up to a maximum lateral distance of 20m (65 feet) from the port side of the ship.

2.4 Normal pressure refuelling procedures associated with bonding the special hose assembly to the rotorcraft, prior to connection and after disconnection of the hose end coupling, apply.

2.5 On completion of fuel loading the hose end coupling is disconnected from the refuelling connector and the H.I.F.R. equipment lowered to the ship's deck by the rotorcraft rescue hoist, the cable and hook of which is then recovered by the flight crew.

3 COMPATIBILITY OF ROTORCRAFT REFUELLING SYSTEM

3.1 Consideration should be given to the following features to provide adequate compatibility and satisfactory interface with the H.I.F.R. equipment:

3.1.1 The refuelling connector and rescue hoist should be located sufficiently adjacent to one another to enable satisfactory connection of the hose end coupling. In the case of a side mounted external hoist the refuelling connector should preferably be on the same side of the rotorcraft. Where the hoist is internally mounted the refuelling connector should preferably be adjacent to the floor hatch to assist in drainage of any fuel spillage within the crew compartment.

3.1.2 To provide optimum visibility for the rotorcraft pilot (normally seated on the starboard side of the rotorcraft flight deck and in conformity with established operating procedures) an externally mounted rescue hoist, together with the refuelling connector should be located on the starboard side of the rotorcraft.

3.1.3 The pressure refuelling connector and its associated grounding socket should be accessible from the rotorcraft crew compartment doorway or hatch, when open, and within a distance of the rescue hoist which is compatible with the H.I.F.R. equipment hose assembly.

3.1.4 The location of the pressure refuelling connector used for normal ground and shipboard refuelling should be such that it is capable of satisfactory interface with the H.I.F.R equipment. Where this is not practical an additional connector should be provided.

3.1.5 Refuelling flow rates from the ship's supply system, when receiving fuel by means of the H.I.F.R. equipment, are affected not only by the additional pressure loss along the hoses or ship's underdeck pipelines to the deck attachment point, but also by the head loss due to the hover height of the rotorcraft. During initial design of the rotorcraft refuelling system, or when providing an additional connector for compatibility with the use of the H.I.F.R. equipment, account should be taken of this inherent loss of delivery pressure and the necessity to minimise the pressure drop characteristics of the rotorcraft refuelling system in order to achieve adequate flow rates during refuelling in hover flight.

CHAPTER 702

FUEL SYSTEMS

1 INTRODUCTION

1.1 The requirements of this chapter govern the design, construction and installation of airframe fuel systems in turbine-engined rotorcraft and those with rotor tip propulsion units (other than rockets), unless otherwise stated in the paragraph concerned.

1.2 In this chapter the term 'engine' shall include any rotor-mounted propulsion unit requiring a fuel system.

1.3 A fuel system comprises all those items, including fuel tanks and instrumentation required for fuel system management which are needed to meet the full range of fuel flow of the engine(s) and auxiliary power unit(s) using the fuel carried in the rotorcraft fuel tanks.

2 GENERAL

2.1 The fuel system shall be designed and constructed so as to ensure an adequate supply of fuel to each engine at flow rates, pressures and temperatures within the ranges agreed with the engine manufacturer(s) in all ground operations and flight conditions relevant to the Rotorcraft Specification including periods under negative increments of normal acceleration as defined in Chapter 700 para 2.

2.2 The fuel systems shall be suitable for F34 and any of the fuels/additives stated in the Rotorcraft Specification [1]. The possibility of interaction of the fuels (and additives if used) with the materials used in the system shall be investigated and where necessary tests shall be made to establish compatibility.

2.3 In multi-engined rotorcraft, all usable fuel including fuel for power augmentation, shall be available for any engine or combination of engines (see Leaflet 702/1, para 5).

2.4 In multi-engined rotorcraft, the design shall be such that no single failure of any part of the system shall result in loss of power in more than one engine. In addition each fuel system must meet the requirements of Chapter 700 para 2.2 and Chapter 1005.

2.5 In rotorcraft with powered rotors, a single failure and on single-engined rotorcraft stoppage of one booster pump, at any stage of flight, including take off, shall not cause such loss of power as would necessitate a forced landing.

2.6 The malfunction of any auxiliary system which draws fuel from the main system shall not adversely affect the main system.

2.7 Each fuel system must be capable of sustained operation throughout its flow and pressure range with fuel initially saturated with water at 27°C and having 0.2cc of free water per litre added and cooled to the most critical condition for icing likely to be encountered in operation (see Chapter 1005).

2.8 Where operational requirements expose the rotorcraft to the possibility of fuel waxing or solidification, due to low temperature, provision shall be made in the system to safeguard against such waxing/solidification.

2.9 The fuel system shall be arranged so as to prevent the formation of airlocks and vapour locks that could case malfunction of the engine.

2.10 The fuel feed system to each engine shall be arranged so as to permit the supply of fuel to each engine through a system independent of the system applying fuel to the other engine(s).

2.11 The fuel system and its components shall be designed so that the occurrence of failure condition, caused by malfunctions within the system or caused by failures of components in other systems, which would prevent the continued safe flight and landing of the rotorcraft is extremely improbable.

2.12 Magnesium rich alloys shall not be used in the construction of fuel systems (ie, tanks, fittings etc.,). Cadmium and copper shall not be used in direct contact with fuel.

2.13 If tank pressurising is used for fuel delivery to engine without the need for fuel transfer pumps, air evolution from the fuel shall be minimised and any free air generated removed before it reaches the engine pump. The air or gas used for pressurising shall be prevented from entering the feed line to the engine in normal operation or with any one component failure with any permitted manoeuvre

3 FUEL SYSTEM ANALYSIS AND TEST

3.1 A safety assessment shall be made for the fuel systems including its associated control and instrumentation system. This assessment shall show the effects of faults on the integrity of component parts and on the functioning of the complete system including engine feed.

3.2 A zonal analysis shall be carried out to show that the occurrence of any failure condition of the fuel system, caused by failure of equipment in other systems, which would prevent continued safe flight and landing of the rotorcraft is extremely improbable.

3.3 A fireworthiness analysis shall be carried out in those areas where flammable fluids or vapours might be liberated by leak of the fuel system or its components to ensure that there are means to prevent ignition and means to minimise the hazard in the event that ignition does occur. Each area shall be analysed in detail to ensure it complies with the requirements of Chapter 712 (see also Leaflet 702/5).

3.4 Proper fuel system functioning under all probable operating conditions, including probable component failures, must be shown as required by Chapter 1005. The tests to show compliance with this chapter shall be carried out on:

(i) the working rig that the constructor is required to provide (see DEF STAN 05-123).

(ii) the rotorcraft.

These tests shall be demonstrated to the satisfaction of the Rotorcraft Project Director.

4 FUEL SYSTEM LIGHTNING PROTECTION

4.1 The fuel system must be designed and arranged to prevent the ignition of fuel vapour within the system (see Chapter 726) by:

(i) Direct lightning strikes

(ii) Swept lightning strokes to areas where swept strokes are highly probable and

(iii) Corona and streamering at fuel vent outlets.

5 FUEL FLOW

5.1 Each fuel system must provide the fuel flow required under each intended operating condition and manoeuvre. Compliance must be shown as follows:

(i) Fuel must be delivered to each engine at a pressure within the limits specified by the engine manufacturer.

(ii) The quantity of fuel in the tank shall not exceed the minimum necessary to show compliance with this para plus the unusable fuel defined in para 7.1.

(iii) Each individual booster pump or other means of pressure feeding shall be capable of supplying fuel to the engine at the required pressure under all conditions of flight, excluding power augmentation, at a rate of 1.25 times the maximum normal flow required. Where power augmentation is provided, the complete fuel system shall be such that, with the booster pump(s) operating the pressure flow at the inlet connections to the engine and power augmentation system shall be adequate for satisfactory operation at maximum power augmentation (see Leaflet 702/1, para 5).

(iv) If a flowmeter is provided any moving parts of the meter shall be held in a position that will produce the maximum pressure loss when determining compliance with the flow test.

5.2 If an engine can be supplied from more than one tank the fuel system shall be such that there is no possibility of any interruption of the fuel supply to the engine(s) when changing the source of supply.

5.3 When automatic fuel selection is required it will be called for in the Rotorcraft Specification. However, its adoption is recommended in all cases where mass penalty for inclusion is acceptable.

5.4 If an automatic fuel system control, a single failure of which would jeopardise the safety of the rotorcraft, is fitted, there shall be provided:

(i) an indication to the pilot of malfunction of the system, and

(ii) an alternative means of complete control

5.5 Suction systems shall give full power under all the most demanding combination of conditions likely to arise on the rotorcraft.

6 FLOW BETWEEN INTERCONNECTED TANKS

6.1 If fuel can be transferred from one tank to another in flight, the fuel tank vents and the fuel transfer system shall be designed:

(i) so that no structural damage to the tanks can occur because of overfilling

(ii) to prevent an overflow from the tank vents when the tanks are full

(iii) to avoid any significant unwanted change in c g position

(iv) to provide the flight crew with indication when fuel transfer is taking place

(v) to provide fuel delivery to the engine(s) feed tank(s) at a rate not less than the rate of fuel consumption of the engine(s) in the maximum thrust (power) condition for all altitudes. For rotorcraft with power augmentation, the fuel transfer system and the engine(s) feed tank(s) shall be designed such that the feed tanks(s) will continue to supply fuel to the engine(s) at not less than the rate of fuel consumption at maximum power, at all altitudes for the maximum time specified for augmented power. A single functional failure which causes the loss of the transfer system shall not result in a hazardous condition (see Leaflet 702/1, para 5).

7 UNUSABLE FUEL

7.1 The unusable fuel quantity for each tank and its fuel system components shall be established. The unusable quantity in each tank is that quantity at which first evidence of engine malfunction occurs when fed from that tank. The flight condition used to establish the unusable fuel quantity shall be discussed and agreed with the Rotorcraft Project Director. Component failures need not be considered.

Note: Consideration shall be given to the possibility of fuel starvation in all Reasonably Probable attitudes (e.g., after engine failure or stability augmentation system failure).

7.2 The effect of a failure of each pump on the unusable fuel quantity shall also be determined for each tank and the results declared.

8 FUEL SYSTEM HOT WEATHER OPERATION

8.1 There shall be no evidence of airlocks, vapour locks or other malfunction when the rotorcraft is operated with an approved fuel at a temperature of at least 45°C. Any allowance for solar radiation which may result in a higher temperature than 45°C shall be discussed and agreed with the Rotorcraft Project Director.

8.2 The test conditions for hot fuel trials will be detailed in Chapter 1005. Until such time as these are defined in detail further information can be obtained from the Rotorcraft Project Director.

8.3 Compliance with para 8.1 and 8.2 must be shown in flight or on a simulated ground installation that closely simulates flight conditions. Prior agreement to the test conditions and the method of compliance must be obtained from the Rotorcraft Project Director.

9 CRASHWORTHINESS

9.1 Fuel tanks and the associated fuel lines and components shall be designed, located and installed so as to render the liberation of fuel unlikely in otherwise survivable crash conditions.

9.2 Each fuel tank and its installation shall be designed or protected so that the fuel tank will retain fuel without leaking under the inertia loads arising from the specified crash conditions (see Chapter 307).

9.3 Fuel tank installations shall be such that the tank will not be ruptured by the rotorcraft rolling over, nor by a collapsed landing gear, nor by the tearing away of the mountings for a landing gear or an engine or other high-inertia item.

9.4 The fuel tanks of communication and training rotorcraft shall be crash resistant in accordance with DEF STAN 15-2.

9.5 Fuel tanks shall be installed so that the tanks will not be ruptured by scraping action with the ground if the rotorcraft slides with some or all of its landing gear units retracted or collapsed.

9.6 Each fuel line, that might be susceptible to damage caused by deformation of the surrounding or supporting structure in a potentially survivable crash, shall either be designed and installed to allow likely deformation and elongation without leakage or shall incorporate a means of preventing the escape of hazardous quantities of fuel in the event of rupture of the line (see Leaflet 702/5).

9.7 Each fuel line shall be constructed and routed to withstand the inertia loads arising from the Crash Landings condition specified in Chapter 307 without hazardous leakage.

9.8 The fuel venting system shall be designed to minimise spillage of fuel through the vents in the event of a rollover during landing or ground operation.

9.9 Demonstration of compliance with these paras shall be by analysis and test as agreed with the Rotorcraft Project Director (see Chapter 307).

10 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

10.1 The requirement for the protection of fuel systems from enemy weapon effects will be stated in the Rotorcraft Specification which will also define its combat role.

10.2 The design aim shall be that, following any single strike by one of the Defined or Specified Threats (Chapter 112), sufficient fuel will be retained so that the rotorcraft will be recoverable from any point on the specified mission profile(s). The Designer shall inform the Rotorcraft Project Director at an early stage of the extent to which this design aim may not be achieved.

10.3 A Vulnerability Analysis as discussed and agreed with the Rotorcraft Project Director shall be carried out to show compliance with the combat survivability requirements as detailed in the Rotorcraft Specification, Chapters 112 and 712. See also Leaflets 712/3, 702/4 and 702/5.

11 FUEL TANKS

11.1 All rotorcraft shall have at least two fixed internal tanks unless otherwise agreed with the Rotorcraft Project Director. These may be built as a single unit provided that the divisions between them are fuel tight and that each has an independent inlet, outlet and venting connection.

11.2 Each fuel tank as installed, must be able to withstand, without failure, the vibration, inertia, fluid and structural loads that it may be subjected to in operation.

11.3 Each metal tank must be protected against corrosion and the possibility of corrosion resulting from micro-biological contamination of fuel (see Chapter 407).

11.4 Flexible fuel tanks must be approved in accordance with the requirements of DEF STAN 15-2.

11.5 For pressurised fuel tanks, a means with fail safe features must be provided to prevent the build up of an excessive pressure difference between the inside and outside of the tank.

11.6 Where inert gas is used to pressurise the fuel tanks in addition to creating an inert environment an alternative shall be provided using air. The air shall be automatically supplied if the inert gas system is not in use or the inert gas pressure falls below its design pressure.

11.7 When air or inert gas is used for pressurising fuel tanks, precautions shall be taken to ensure that during any flight condition or servicing operation fuel cannot enter other systems pressurised from the same supply.

12 FUEL TANK INSTALLATION

12.1 Non integral tanks shall be arranged so that they can be removed and replaced easily with the least possible disturbance to other parts of the rotorcraft.

12.2 Means must be provided to prevent chafing between the tank and its supports. Material used for this purpose shall be non absorbent or treated to prevent the absorption of fluids.

12.3 Each fuel tank must be supported so that tank loads (resulting from the mass of fuel in the tanks) are not concentrated on unsupported tank surfaces. The installation design of flexible tanks shall be such that when replacement is required, it can be carried out conveniently either in the field or at remote aerodromes.

12.4 If a flexible tank is used, it must be supported so that it is not required to withstand fluid loads and each interior surface of the tank compartment must be smooth and free of projections that could cause wear of the flexible tank, unless:

(i) Provisions are made for protection of the liner at these points or

(ii) The construction of the liner itself provides that protection.

12.5 Spaces adjacent to tank surfaces must be ventilated to avoid fume accumulation due to minor leakage. If the tank is in a sealed compartment, ventilation may be limited to drain holes large enough to prevent excessive pressure resulting from altitude changes. These drain holes shall be at the lowest point to prevent accumulation of fluid. In addition the drain holes shall be arranged so that no hazard is likely to result from fuel leaking into the surrounding space.

12.6 Means shall be provided to prevent excess suction within any tank becoming sufficient to cause it to collapse or be damaged or foul any internal component under any normal condition of operation.

12.7 The location of each tank must meet the requirements of Chapter 712. In addition:

(i) Each fuel tank must be isolated from personnel compartments by a flameproof and fuelproof enclosure. The fuel proof enclosure shall be drained and ventilated.

(ii) There shall be adequate airspace between each tank and each firewall or shroud isolating a designated fire zone such that the possibility of ignition of liquids or vapours is minimised in the event of a fire in the designated fire zone (see also Chapter 712).

12.8 Rotorcraft skin which lies immediately behind a major air outlet from the engine compartment, shall not act as a wall of an integral tank without the agreement of the Rotorcraft Project Director.

12.9 Rotorcraft skin forward of an engine/engine compartment air intake shall not act as wall of an integral tank if it can be damaged by enemy action.

12.10 Provision shall be made for the easy internal inspection and cleaning of every metal tank and for adequate access to permit repair of the complete interior of integral tanks as agreed with the Rotorcraft Project Director.

13 FUEL TANK STRENGTH REQUIREMENTS

13.1 Each tank (or tank compartment structure where flexible tanks are fitted) shall, with fuel at maximum permitted specific gravity, satisfy the strength requirements for the rotorcraft as a whole, at all levels of fuel in the tank from empty to full: each of the remaining tanks in turn shall be considered as damaged and incapable of providing support to the one under consideration. In addition each tank shall have sufficient strength to meet the requirements of para 11.2 at Limit Load conditions without leakage or detrimental deformation and at Ultimate Loads without unacceptable leakage or structural failure.

13.2 Each tank (or tank structure where flexible tanks are fitted) shall have proof and ultimate factors not less than 1.5 and 2.0 respectively on the loads resulting from pressures in each of the following cases considered separately:

(i) The combination of the internal pressure applied to transfer fuel or to prevent fuel boiling and the local external pressure in straight and level flight under maximum cruising conditions at all heights up to the maximum attainable.

(ii) The pressure developed during refuelling and defuelling.

13.3 There shall be no leakage from or permanent distortion of the tanks at any proof load or combination of loads.

14 ENGINE REFRIGERANT INJECTION SYSTEMS

14.1 Each refrigerant injection system must provide a flow of fluid at the rate and pressure established for proper engine functioning under each intended operating condition. If the fluid can freeze, fluid freezing shall not damage the rotorcraft or adversely affect rotorcraft performance.

14.2 The system including tanks and components shall be constructed from materials resistant to corrosion by refrigerant fluids (see Chapter 407) and shall meet the relevant requirement, as agreed with the Rotorcraft Project Director, of this Chapter.

14.3 A contents gauge is not required for the tank, but means shall be provided to give clear indication when the tank is full.

14.4 The engine refrigerant tank capacity, available for the use of each engine must be large enough to allow operation of the rotorcraft under the approved procedures for use of liquid-augmented power. The computation of liquid consumption must be based on the maximum approved rate, appropriate for the desired engine output and must include the effect of temperature on engine performance as well as any other factors that might vary the amount of liquid required.

14.5 In order to permit ground testing, a self-sealing coupling shall be provided in the pipe line between the pressure regulator and the engine, the male half of the coupling being fitted on the engine side.

14.6 The tank gravity filling orifice shall be not less than 38mm (1.5 in) effective internal diameter. 2 If a pressure replenishment connection is fitted, it must meet the requirement of BS.C14. 3 Means shall be provided for collecting and draining away any liquid spilt at the filling points.

14.7 Each tank filler cap opening must be marked to identify the fluid at or near the filler cover.

14.8 If the refrigerant fluid is subject to freezing and the fluid can be drained in flight or during ground operation the drains must be designed and located to prevent the formation of hazardous quantities of ice on the rotorcraft as a result of the drainage.

15 FUEL TANK EXPANSION SPACE

15.1 Each fuel tank shall have an expansion space of not less than 2 per cent of the tank capacity so that the system can stand a rise in fuel temperature, after refuelling, due to solar radiation, without causing fuel to be spilt on the ground. It must be impossible to fill the expansion space inadvertently when the rotorcraft is within the normal range of attitudes expected, taking into account ground slope and asymmetric loading. For pressure fuelling systems, an automatic shut off means shall be provided to prevent the quantity of fuel in each tank from exceeding the maximum quantity approved for that tank (see Chapter 701).

16 FUEL TANK TEMPERATURE

16.1 The highest temperature allowing a safe margin below the lowest expected auto ignition temperature of the fuel in the fuel tanks must be determined and agreed with the Rotorcraft Project Director.

16.2 No temperature at any place inside any fuel tank where fuel ignition is possible may exceed the temperature determined under para 16.1. This must be shown under all probable operating, failure and malfunction conditions of any component in any system whose operation, failure or malfunction could increase the temperature inside the tank.

17 FUEL TANK SUMP AND FUEL SYSTEM DRAINS

17.1 Each fuel tank must have a sump with an effective capacity, in the normal ground attitude of not less than the greater of 0.10 per cent of the tank capacity or one-quarter of a litre. The sump capacity must be effective with the rotorcraft in any normal attitude and must be located so that the sump contents cannot escape through the tank outlet opening.

17.2 Each fuel tank or tank group must allow drainage of any hazardous quantity of water from any part of the tank or tank group to the lowest point with the rotorcraft in the ground attitude.

17.3 Each fuel tank sump must have accessible drains that:

(i) Allows complete drainage of the sump on the ground.

(ii) Discharges clear of each part of the rotorcraft and

(iii) Has manual or automatic means for positive locking in the closed position.

17.4 Drainage of the fuel system must be accomplished by the use of fuel strainer and fuel tank sump drains.

17.5 Each drain required by para 17.4 must:

(i) Discharge clear of all parts of the rotorcraft.

(ii) Have manual or automatic means for positive locking in the closed position and

(iii) Have a drain valve that is readily accessible and which can be easily opened and closed.

17.6 Each drain valve required by para 17.4 must be either located or protected to prevent fuel spillage in the event of a landing with landing gear retracted.

17.7 In the case of rotorcraft engines which employ means to dump burner line fuel to a common drain tank on engine shut down, no failure of automatic valves or other devices shall permit other components in the common engine drain system to be subjected to over-pressurisation when the engine is restarted.

18 FUEL TANK VENTS

18.1 Each fuel tank must be vented from the top part of the expansion space so that venting is effective under normal flight conditions.

Note: The design of the venting of a series (cascade) transfer system which may not be able to comply with para 18.1 shall at an early stage of the design be discussed and agreed with the Rotorcraft Project Director.

18.2 The venting arrangement shall be such that no significant amount of fuel is lost under any flight condition. The venting system shall be self draining and free from traps in which moisture can collect.

18.3 Venting shall be such that no fuel is lost from the full tanks when the rotorcraft is parked or manoeuvred on the ground.

18.4 On naval rotorcraft, the venting arrangements shall be such that no fuel is lost from the tanks when the rotorcraft is parked, with rotors spread or folded, at any angle relative to the centre line of any specific ship which is rolling 20^{o} to port and 20° to starboard.

18.5 The venting capacity and vent pressure levels must maintain acceptable difference in pressure (i.e., within design limits) between the interior and exterior of the tank during:

(i) Normal flight operation,

(ii) Maximum rate of ascent and descent and

(iii) Refuelling and defuelling (where applicable). (Refer to Chapter 701, para 4.3.2 and Leaflet 701/2, para 5).

18.6 Airspaces of tanks with interconnected outlets must be interconnected.

Note: The design of series (cascade) transfer systems which may not be able to comply with para 18.6 shall at an early stage of design of the system be discussed and agreed with the Rotorcraft Project Director.

18.7 No vent or drainage provision may end at any point:

(i) Where the discharge of fuel from the vent outlet would constitute a fire hazard (see Chapter 712).
(ii) From which fumes could enter personnel compartments.

18.8 Each vent system must have means to avoid stoppage by dirt or ice formation or other causes. Particular attention must be taken to ensure that mud, dust, dirt etc., thrown up during take-off and taxi shall not enter the vent system.

19 FUEL TANK OUTLET

19.1 There must be a fuel strainer for the fuel tank outlet or for the booster pump to prevent the passage of any object that could restrict fuel flow or damage any fuel system component.

19.2 The clear area of each fuel tank outlet strainer must be at least five times the area of the outlet line.

19.3 The diameter of each strainer must be at least that of the fuel tank outlet.

19.4 Each strainer must be accessible for inspection and cleaning.

20 FUEL TANK FILLER CONNECTIONS

20.1 Each fuel tank filler connection shall meet the requirements of Chapter 701 in addition to the following:

20.2 Each fuel tank filler connection must prevent the entrance of fuel into any part of the rotorcraft other than the tank itself.

20.3 Each filler connection must be marked as prescribed in Chapter 806.

20.4 Each recessed filler connection that can retain any appreciable quantity of fuel must have a drain that discharges clear of the entire rotorcraft.

20.5 Each filler cap must provide a fuel tight seal under the pressure expected in normal operation.

20.6 Each fuel filling point must have a provision for electrically bonding the rotorcraft to ground re-fuelling equipment (see STANAG 3212).

21 PREVENTION OF INCORRECT ASSEMBLY

21.1 The system shall comply with the requirements of Chapter 100.

22 NON-RETURN VALVES

22.1 Non-return valves, or some other fully automatic means, shall be fitted in the fuel pipe lines of systems having two or more tanks, to ensure that no fuel is lost from the remaining tank(s), if one tank is holed or otherwise damaged. The non-return valves shall be situated as close as possible to the point from which the return flow to a damaged tank could otherwise commence, in order that they may isolate the maximum possible amount of pipe line with each tank.

23 FUEL SYSTEM LINES AND FITTINGS

23.1 Pipe lines, fittings and components shall be installed and supported to prevent excessive vibration and to withstand loads due to fuel pressure and accelerated flight conditions. They shall also be designed to achieve an adequate fatigue life in regards to loads imposed by peak transient pressures.

23.2 Each fuel line connected to components of the rotorcraft between which relative motion could exist shall have provision for flexibility.

23.3 Each flexible connection in fuel lines that may be under pressure or subjected to axial loading shall use flexible hose assemblies or equivalent means.

23.4 Flexible hose must be approved or shown to be suitable for the particular application.

23.5 No flexible hose that might be adversely affected by exposure to high temperatures may be used where excessive temperatures exist during operation or after engine shut down.

23.6 Each fuel line within the fuselage must be designed and installed to allow a reasonable degree of deformation and stretching without leakage.

23.7 If a rise in temperature of the fuel in the fuel lines can result in a rise in pressure, means shall be provided to prevent the pressure becoming excessive.

23.8 Fuel lines shall be routed, as far as practical, through tanks to minimise the risk caused by leaking pipe (joints) (see Chapter 712).

Note: If fuel pipes are routed through tanks then it must be shown that the capability of the vent system is such that the possibility of back pressures from major leaks causing structural damage is extremely improbable.

23.9 Absorbent materials close to fuel line joints must be covered or treated to prevent the absorption of hazardous quantities of fluid.

23.10 Fuel system pipe lines passing through personnel or pressurised compartments shall comply with Chapter 712, para 14.1.

23.11 See Leaflet 703/5 for recommendations about V-Flange couplings.

24 FUEL SYSTEM L.P. AND CROSS FEED VALVES

24.1 Valves shall be provided to enable the appropriate flight crew member to isolate promptly the flow of fuel to each engine individually. Closing the fuel valve for any engine shall not make any of the fuel supply unavailable to the remaining engine(s) (see Chapter 712).

24.2 Valves provided to meet para 24.1 requirement shall be installed in a protected location and shall be separated from the engine by a firewall or its equivalent. It shall be shown that no dangerous quantity of fuel will drain into the engine compartment after a valve has been closed.

24.3 The operation of any shut-off valve shall not interfere with the later emergency operation of any other equipment, such as the means for declutching the engine from the rotor drive.

24.4 Valves shall be provided in all fuel cross feed pipes.

24.5 Valves shall be provided with positive stops at, or locating provisions in, the "ON" and "OFF" positions; they shall be so supported that loads resulting from their operation, or from accelerated flight conditions are not transmitted to the lines connected to them.

24.6 Mechanically operated valves shall be installed so that they will be positive and easy to operate and will not change their position due to vibration. Electrically operated valves shall give a positive indication of their position to the appropriate flight crew member.

24.7 Means shall be provided to guard against inadvertent operation of the valves and to make it possible for the flight crew personnel to re-open the valves rapidly after they have been closed.

24.8 Valve control knob shape and position shall be in accordance with the requirements of Chapter 107.

25 FUEL FILTER CONTAMINATION

25.1 Airframe fuel system components (e.g., pumps and valves) shall be type tested with contaminated fuel in order to demonstrate their ability to function reliably in the presence of contaminants to be found in rotorcraft fuel tanks (see Leaflet 702/6).

25.2 There must be a filter between the fuel tank outlet and the inlet of either the engine control unit or an engine driven positive displacement pump whichever is nearer the fuel tank outlet.

25.3 This fuel filter must:

(i) be accessible for draining and cleaning and must incorporate a screen or element which is easily removable.

(ii) have a sediment trap and drain, except that it need not have a drain if the strainer or filter is easily removable for drain purposes.

(iii) be mounted so that its weight is not supported by the connecting lines or by the inlet or outlet connections of the filter itself, unless adequate strength margins under all loading conditions are provided in the lines and connections.

(iv) have the capacity (with respect to operating limitations established for the engine) to ensure that engine fuel system functioning is not impaired with the fuel contaminated to a degree (with respect to particle size and density) that is greater than that established for the engine.

(v) An indicator to show contamination of the filter before it reaches the capacity established by the operating limitations established for the engine.

26 FUEL PUMPS

26.1 Each fuel pump required for proper engine operation, or required to meet the fuel system requirements of this chapter (other than those in para 26.2) is a main pump. There must be at least one main pump for each turbine engine.

Note: If more than one pump is required in order to provide proper engine operation over the whole flight envelope (other than those in para 26.2) then such pumps are also main pumps. In which case there must be one set of main pumps for each turbine engine.

26.2 There shall be an emergency pump immediately available to supply fuel to the engine if any main pump fails or by any other acceptable method agreed by the Rotorcraft Project Director. The power supply for each emergency pump, if fitted, shall be independent of the supply for each corresponding main pump.

26.3 If both the normal pump and emergency pump operate continuously there must be a means to indicate to the appropriate flight crew members a malfunction of either pump.

27 FUEL JETTISONING SYSTEM

27.1 Provision for fuel jettisoning shall be made when specified in the Rotorcraft Specification. On rotorcraft having a water take-off capability it shall also be possible to jettison fuel while the rotorcraft is floating.

27.2 The design rate of the fuel jettisoning system shall be based on exhaustion of 98 per cent of all jettisonable fuel within 5 minutes or as specified in the Rotorcraft Specification.

27.3 Means shall be provided to ensure that after jettisoning sufficient fuel is still available for ten minutes flight at full throttle or as agreed with the Rotorcraft Project Director.

27.4 It shall be possible to terminate and, if so desired, re-start jettisoning at any time during the jettisoning operation.

27.5 Control of the fuel jettison system shall be independent of the main rotorcraft system as far as possible and not rely on the engine(s) being operative.

27.6 If the system involves jettisoning fuel from feed line rather than from a fuel tank, then the loss of pressure occasioned by the fuel being jettisoned shall not affect the engine(s) under any power/flight condition.

27.7 The fuel jettisoning system shall safely discharge fuel clear of all parts of the rotorcraft: the extended core of the jettisonable fuel shall not come in contact with the jet efflux.

27.8 Fuel or vapour shall not enter any part of the rotorcraft as a result of fuel jettisoning.

27.9 The jettisoning operation shall not adversely affect the control of the rotorcraft.

27.10 The fuel jettisoning system, and its operation must be free from fire hazard. In addition when jettisoning is terminated and the system is no longer in use an explosion in any part of the jettisoning system downstream of the shut-off valve will not spread into the fuel tanks; the valves in the closed position shall be designed to withstand such an explosion.

27.11 The jettisoning discharge pipe must be designed to prevent any dangerous corona discharge.

27.12 If an extendible jettisoning outlet pipe is provided, means shall be provided to indicate to the appropriate flight crew member when the pipe moves from the retracted position. If a jettisoning pipe is extended inadvertently, or following a defect, means shall be provided for retracting the pipe if a hazard could otherwise be caused in any flight condition or subsequent landing.

27.13 If the jettison system is required to operate with the outlet submerged under water, then the system shall be designed to prevent contamination of the main fuel system by salt water.

27.14 Each fuel jettisoning system control must have guards to prevent inadvertent operation. No control shall be near any fire extinguisher control or other control used to combat fire.

27.15 Tests shall be conducted in accordance with Chapter 1005 to demonstrate compliance with the requirements of this para 27.

28 AUXILIARY POWER UNIT FUEL SYSTEM

28.1 Each fuel system shall allow the supply of fuel to the Auxiliary Power Unit;

(i) through a system independent of each part of the system supplying fuel to the main engines or

(ii) by any other acceptable means.

Note: The fuel supply to an A.P.U. may be taken from the fuel supply to the main engine if provision is made for a shut-off means to isolate the A.P.U. Fuel Line.

28.2 If an auxiliary power unit is provided for Essential Services for use in flight, there shall be sufficient capacity of fuel to ensure continuous operation for the period of duration for which certification is requested.

29 FUEL SYSTEM INSTRUMENTATION

29.1 FUEL CONTENTS INDICATORS

29.1.1 There must be means to indicate to the appropriate flight crew members the quantity in kilograms of usable fuel:

(i) as a total in the rotorcraft (unless otherwise agreed with the Rotorcraft Project Director).

(ii) in each internal tank or group of tanks.

(iii) in each external tank or group of external tanks (unless otherwise agreed with the Rotorcraft Project Director).

29.1.2 The pilot shall be provided with a cautionary warning when the usable fuel falls below a datum setting. The Rotorcraft Specification shall state whether the datum setting is to be fixed or variable. If fixed, the datum setting shall be specified and if variable, the range through which the datum shall be adjustable, shall be specified.

29.1.3 Errors in the fuel quantity indicators, when the rotorcraft is in normal cruise attitude, shall not exceed 2 per cent of the usable fuel remaining in the tank, plus 2 per cent of the full usable capacity of the tank. Similar requirements shall be met for groups of tanks which use a single indicator. When the rotorcraft departs from the cruise attitude, the additional error shall not exceed 3 per cent of usable fuel remaining in the tank (or group) between climb and descent or as defined by the Rotorcraft Project Director.

29.1.4 It shall be possible to trim the indicator(s) to give the minimum possible error at "full and 20 per cent full" fuel capacities and the attitude error at these two points shall not be greater than ±1 per cent. The system shall be such that any errors which occur below the 20 per cent capacity level are in a negative and not a positive sense, i.e., the indicator(s) shall not over-read.

29.1.5 A suitable calibration to permit measurement of fuel contents to within ±1 per cent of the full scale deflection during ground refuelling shall be provided.

29.1.6 There shall be no loss of accuracy of fuel quantity indication due to temperature changes throughout the range quoted in Chapter 101 when using any of the fuel types on which the rotorcraft is required to operate.

29.1.7 For pressure refuelling systems as specified in Chapter 701 an indicator shall be provided (in addition to the contents gauges in the cockpit) for use by ground personnel or when each tank (including any other tanks not part of the permanent fuel system) is filled to the required level.

29.2 FUEL FLOWMETERS

29.2.1 When called for in the Rotorcraft Specification fuel flowmeters shall be fitted to record the instantaneous flow to each engine in kilograms per hour. Flowmeters shall incorporate an automatic or manual means of compensating for specific gravity variations between fuels. On rotorcraft with powered rotors, the engines supplied by an individual fuel line shall be treated as a single engine for the purpose of this requirement.

29.2.2 The fuel pressure drop at the engine attributable to the flowmeter shall have no effect on engine performance, and failure of any part of the flowmeter installation shall not restrict fuel flow to the engine.

29.3 FUEL PRESSURE INDICATORS

29.3.1 Indicators shall be provided to show if the fuel pressure in the low pressure system is adequate.

29.3.2 Indicators shall be provided to show the appropriate flight crew member:

(i) that transfer pressure in the fuel lines from auxiliary tanks is adequate before take-off

(ii) that fuel is being transferred and

(iii) when transfer has ceased.

30 FUEL TANK TESTS

30.1 A typical fuel tank of a new design shall be tested to demonstrate that the design will meet the proof strength requirements of para 13.

30.2 Each metallic tank with large unsupported or unstiffened surfaces whose failure or deformation could cause fuel leakage must be able to withstand the following tests or their equivalent without leakage or excessive deformation of the tank walls:

(i) Each complete tank assembly and its supports must be vibration tested while mounted to simulate the actual installation.

(ii) Except as specified in para 30.2 (iv) the tank assembly must be vibrated for 25 hours at an amplitude of not less than 0.8 mm (unless another amplitude is substantiated) while 2/3 filled with water or other suitable test fluid.

(iii) The test frequency of vibration must be as follows:

(a) If no frequency of vibration resulting from any r.p.m. within the normal operating range of the engine or rotor system speeds is critical, the test frequency of vibration must be 2,000 cycles per minute.

(b) If only one frequency of vibration resulting from any r.p.m. within the normal operating range of engine or rotor system speeds is critical, that frequency of vibration must be the test frequency.

(c) If more than one frequency of vibration resulting from any r.p.m. within the normal operating range of engine or rotor system speeds is critical, the most critical of these frequencies must be the test frequency.

(iv) Under para 30.2 (iii) (b) and (c) of this para the time of test must be adjusted to accomplish the same number of vibration cycles that would be accomplished in 25 hours at the frequency specified in para 30.2 (iii) (a) of this para.

(v) During the test, the tank assembly must be rocked at the rate of 16 to 20 complete cycles per minute, through an angle of 15° on both sides of the horizontal (30° total) about the most critical axis for 25 hours. If motion about more than one axis is likely to be critical, the tank must be rocked about each critical axis for 12.5 hours.

30.3 Except where satisfactory operating experience with a similar tank in a similar installation is shown, non-metallic tanks must withstand the test specified in para 30.2 (v) of this para with fuel at a temperature of 43°C. During this test, a representative specimen of the tank must be installed in a supporting structure simulating the installation in the rotorcraft.

30.4 For pressurised fuel tanks, it must be shown by analysis or tests that the fuel tanks can withstand the maximum pressure likely to occur on the ground or in flight.

30.5 All production metal tanks, integral tanks and flexible bag tanks when installed, shall be subjected to a static pressure test of not less than one third of the appropriate design proof pressure without leakage.

31 FUEL CONTENTS TESTS

31.1 Tests shall be made on the rotorcraft to determine the calibration of the fuel contents gauges. The accuracy of the fuel contents gauges shall be established for the type of fuel on which they will normally operate in Service and a list of corrections shall be provided for the fuels specified in the Rotorcraft Specification.

32 CROSS REFERENCES

32.1 A number of requirements directly related to the fuel system appear elsewhere in DEF STAN 00-970, Volume 2 and the most important of these are listed in Table 1.

TABLE 1
LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	Paragraph	Subject
100	2	Standard Items
100	7	Prevention of Incorrect Assembly
100	8	Conditions of Operation
101	1	Temperature Limits
107	8 & Table 6	Fuel System Controls
112	-	Reduction of vulnerability
201	-	Fatigue
202	3 & Fig. 1	The Flight Envelope
307	4.2	Fuel System
400	9	Use of Magnesium Alloys
404	4.6	Fuel Tanks
700	2.1	Flight Conditions
700	2.2	System Failure
700	5.2	Cocks and Control Valves
701	-	Re-fuelling and de-fuelling
726		Lightning Strike Protection
708	3	Bonding
712	-	Fire Precautions
800	-	General Maintenance Requirement
802	-	Routine Servicing
804	-	Replacement of Components
806	-	Marking and Notices
1005	-	Flight Tests
117/1		Definitions of terms associated with probability and effects
117/2		General Recommendations and means of compliance

REFERENCES

Reference	ASCC Air Std	STANAG	BS
1	-	4270	-
2	25/11	3212	C13
3	25/14	-	C14

*TSS - Transport supersonic standard issue 4 March 1976.

Prepared by Anglo/French authorities for certification of Concorde. Full Standard can be obtained from:

CAA Printing and Publication Services
Greville House
37 Gratton Road
Cheltenham

Tel: 0242 35151

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REFERENCE PAGE

Defence Standards

00-971	.. ..	General specification for aircraft gas turbine engines
15-2	.. ..	Flexible tanks for use in aircraft fuel and methanol/ water systems
17-2	.. ..	Replenishment equipment for aero-engine refrigerant injection systems
47-12	.. ..	Polytetrafluoroethylene (PTFE) hose assemblies for medium and high pressure fluid systems in aircraft
47-23	.. ..	Bead profile for aircraft piping and ducting (flexible connections)
53-68	.. ..	'V' flange couplings for aircraft piping and ducting (metric)
53-70	.. .. ..	Dimensions of elastomeric toroidal sealing rings (inch series) for aircraft purposes
91-2	.. ..	Petroleum fuels

British Standards

C4	.. ..	Coupling dimensions for aero-engine refrigerant pressure replenishment connections
C13	.. ..	Sizes of gravity filling orifices on aircraft
C14	..	Specification for coupling requirements for aircraft pressure fuelling connection
3G 100	.. ..	Specification for general requirements for equipment for use in aircraft

LEAFLET 702/1

FUEL SYSTEM

GENERAL RECOMMENDATIONS AND DEFINITIONS

1 INTRODUCTION

1.1 Demonstration of compliance with the requirements of Chapter 702 may be made by a combination of analysis, calculations, rig and flight testing. The requirements of Chapter 702 are applicable to the rotorcraft fuel system. They do not include that part of the system which is mounted on the engine and is certificated as part of the engine. Such requirements are covered by R.D. Specification 2300 Chapter 12. They do not include the requirements for refuelling and defuelling. These requirements are given in Chapter 701. There are also a number of other chapters which include requirements which are relevant to the design and construction of the fuel system. The more important of these are listed in Chapter 702, Table 1.

1.2 Even though the requirements of the rotorcraft fuel system are contained in different chapters of DEF STAN 00-970 Chapter 702 treats the fuel system as whole in that it requires the severity of effects of all conceivable important failures to be taken into account. In this respect, failure includes failure of components considered separately and in combination. The effect of human error and external circumstances should be included in the analysis. The constructor should also consider the effects on the fuel system of failures in other systems which are installed in the same area.

1.3 In carrying out tests on both the rig and the rotorcraft the constructor should include in his demonstration programme those failures which the analysis shows have an important effect on the system. In some cases, the effect of failure on the rotorcraft may be so minor that it need be only a paper study. In other cases, it may be sufficiently important that the effect has to be checked on either the rig or during flight test. If there is any doubt where the failure should be checked, the Rotorcraft Project Director should be consulted.

1.4 It is for the above reasons that the analysis of the fuel system should be a continuous process during the design and development period. The depth of the analysis will of course depend on the complexity of the fuel system. However, the fuel system is a critical one, it includes the storage, measurement, delivery and control of the fuel used in the engines and the re-fuelling and de-fuelling of this fuel. Such systems often require complex electrical systems to control the delivery of the fuel. All these sub-systems should be considered as part of the whole fuel system and should be treated by the analysis accordingly.

2 SAFETY ASSESSMENT

2.1 The primary objective of the safety assessment is to identify critical features of the system. It should make clear what the critical features of the system are and upon which special manufacturing techniques, inspection, testing, crew drills and maintenance practice they are critically dependant. In addition to highlighting those aspects upon which safety depends, the safety assessment should help ensuring that the design is practical and economical and likely to prove reliable in service.

2.2 The safety assessment for the fuel system should include the following information:

(i) A definition of the system and its function.

(ii) A statement of Airworthiness Objectives which lists possible effects which the system can produce on the flying qualities of the rotorcraft.

(iii) A list of the equipment of which the system is comprised.

(iv) Analysis of failure conditions and their effects. This analysis should be carried out to identify all failure conditions which could lead to any effects listed in the Airworthiness Objectives other than minor effects.

(v) Statistical Analysis. The critical functions identified in (iv) for further analysis should be analysed in detail to determine whether they comply with the Airworthiness Objectives.

3 ZONAL ANALYSIS

3.1 The Zonal Analysis examines the location of critical parts of the system in a particular zone and determines the effect of failure in other systems on the fuel system in the vicinity. It should also examine the consequences of failure not only of other systems but of other airframe parts, engines etc.

3.2 To carry out the analysis, the rotorcraft is divided into zones (see para 4) and all the equipment, cable runs, pipe runs in each zone are listed. A study is then made of the effect of failure of the equipment and parts on the functioning of the fuel system. During the analysis it is important to check the segregation between the systems and the installation of the parts to ensure that the design requirements relating to segregation and installation have been met. Threats from outside the zone should also be considered.

4 FIREWORTHINESS ANALYSIS

4.1 To ensure that the fuel system meets requirements laid down in Chapter 712 and that all necessary precautions, (listed in Chapter 712) against possible risk of fire are taken, the constructor has to carry out a methodical analysis of the component parts of the system in relation to the environment in which they are installed in the rotorcraft.

4.2 One method of ensuring that the work is kept to manageable packages is to divide the rotorcraft into regions or zones (see para 3.2) Some of these will be fire zones (see Chapter 712, para 2.1). However, even those zones/regions not containing flammable fluids but containing such items as cable runs etc., which form part of the control of the fluid systems should be examined.

5 GLOSSARY OF TERMS

TERM	DEFINITION
Failure	The inability of an item to perform within previously specified limits. As far as practicable when failure results from enemy action the spirit of the requirement should be met.
System	A combination of inter-related items arranged to perform a given function.
Component	Any self contained part, combination of parts, sub-assemblies or units, which perform a distinctive function necessary to the operation of the system.
Minor Effect	See Volume 1 Leaflet 117/1 for definition.
Major Effect	See Volume 1 Leaflet 117/1 for definition.
Probable	May occur once or several times during the total operational life of each rotorcraft of the type.
Extremely Improbable	So extremely remote that it may be discounted.
Defect	Any confirmed abnormal condition of an item whether or not this could eventually result in a failure.
Hazardous Effect	An effect which can result in: (i) A rare reduction in safety margins (ii) Physical distress on a workload such that the flight crew cannot be relied upon to perform their tasks accurately or completely or (iii) Serious injury to one or more of the crew.
Malfunction	The occurrence of a condition whereby the operation of an item is outside of specified limits.
Usable Fuel	Is that part of the total fuel capacity which is capable of being delivered to the engines at flow rates and pressures within the range agreed with the engine manufacturers, with fuel pumps operating and the rotorcraft in the cruise attitude.
Unusable Fuel	That quantity of fuel which is established when complying with the requirements of Chapter 702, para 7.1.
Fuel Tank Sump Capacity	The fuel tank sump capacity is equal to the difference between the unusable fuel capacity and the amount of undrainable fuel in the tank.

<u>TERM</u>	<u>DEFINITION</u>
Auxiliary Tanks	Those fuel tanks which are additional to the normal rotorcraft fuel system tank installation and which cannot be jettisoned in flight.
Power Augmentation	Definition given in DEF STAN 00-971.

LEAFLET 702/2

FUEL SYSTEMS

FUEL JETTISONING INSTALLATIONS

1 INTRODUCTION

1.1 This leaflet gives points which should be considered in the design of the fuel jettisoning installations, where appropriate.

2 GENERAL

2.1 It should be possible to jettison fuel from selected individual tanks, but where use of Avgas is permitted, some fuel should remain in each tank, to prevent the formation of an explosive mixture. The volume of this fuel should not be less than 1 per cent of the total volume of the tank.

2.2 Where the siphon system of jettisoning is used, it should be possible to start the jettisoning with a minimum of 25 per cent of fuel in any tank.

2.3 Satisfactory jettisoning of fuel should be possible under the following conditions:

(i) Climb at the maximum rate of climb with operating power-units at the maximum rating approved for climb.

(ii) Descent. The minimum rate of descent should be approximately 500 ft/minute unless otherwise agreed with the Rotorcraft Project Director.

(iii) Level flight at the maximum speed for jettisoning.

3 FIRE RISKS

3.1 The danger of fire may arise from the following:

(i) electrostatic discharge by sparking in flight,

(ii) corona of brush discharge,

(iii) electrostatic discharge on landing,

(iv) flames and carbon particles from the engine exhaust,

(v) lightning or

(vi) spontaneous ignition on hot surfaces.

4 INSTALLATION

4.1 As fuel jettisoning is an emergency service, a mechanical or self-contained hydraulic or pneumatic means of operation is preferred.

4.2 Rigid jettison pipes which project below the fuselage are preferred. The pipes should be inclined at an angle of about 60° to the horizontal and be sufficiently long to project the fuel well clear of the rotorcraft. To reduce drag, the pipes may be made retractable, e.g., by the use of hinged joints or telescopic pipes, but when in the discharge position they should be rigid. There should be no leaks in the pipes themselves. Jettison pipes which could strike the ground if they fail to retract should be made of non-metallic material.

4.3 Differences of electrostatic potential and the consequential risk of sparking may arise on a rotorcraft from thunderstorms, hail or dust storms, and from the electrostatic effects of the jet itself. All parts of the discharge pipes should be electrically connected to the main ground system of the rotorcraft.

4.4 The charge on the rotorcraft as a whole may be sufficient to cause corona or brush discharge. The main safeguard against this is the prevention of fuel touching the rotorcraft or passing near projections. It is also advisable to avoid sharp metallic edges and points on parts of the rotorcraft near to the path of the discharged fuel. On the discharge pipe itself it is essential that there should be no sharp edges in contact with the fuel or near it.

4.5 The jettison valves should be designed for rapid opening and closing. It should be possible to close the valves quickly not only when jettisoning is completed but also at any intermediate stage, since it is dangerous to land while fuel is emerging.

4.6 It is not always possible to prevent a local explosion in the discharge pipe on landing, but this is not usually serious provided suitable precautions are taken. The closure valve and the discharge pipe should be able to withstand the explosion, as required by Chapter 702, para 27.10.

LEAFLET 702/3

FUEL SYSTEMS

PROPERTIES OF AVIATION FUELS

1 INTRODUCTION

1.1 This leaflet gives information on the properties of aviation fuels for design purposes.

2 SPECIFIC GRAVITY AND CO-EFFICIENT OF VOLUMETRIC EXPANSION

2.1 The following table gives the average values of the specific gravity and co-efficient of volumetric expansion of aviation fuels which should be used for design purposes, (e.g., weight and c.g., estimates). For evaluating the strength of fuel tanks and their attachments, however, the maximum permitted specific gravity quoted in the relevant fuel specification should be used (see Chapter 702, para 13.1).

Types of Fuel	Specification	Joint-Service Nomenclature	Nato Code Number	Specific Gravity at 60°F/60°F	Coefficient of volumetric expansion (/°C)
Gas turbine engine fuels	DERD 2452[1] DERD 2453[1] DERD 2454[1] DERD 2494[1] DERD 2498	AVCAT/FSII* AVTUR/FSII* AVTAG/FSII* AVTUR AVCAT	F-44 F-34 F-40 F-35 F-43	0.82 0.79 0.77 0.79 0.82	0.00086 0.00094 0.00101 0.00094 0.00086
Aviation gasoline	DERD 2485[2]	AVGAS 100LL (100/130)	F-18	0.72	0.00118
Diesel fuel oil	DEF STAN 91-4 DEF STAN 91-4	DIESO (47/0) DIESO (47/20)	F-75 F-76	0.84 0.84	0.00082 0.00082
MT gasoline	DEF STAN 91-30	MTGAS (80)	F-50	0.75	0.00108

*Fuel System Icing Inhibitor (FSII) to Specification DERD 2451

[1]These specifications implement the requirements of STANAG 3747

[2]This specification implements the requirements of STANAG 3824

LEAFLET 702/4

FUEL SYSTEMS

PROTECTION OF FUEL SYSTEMS FROM ENEMY WEAPON EFFECTS

1 INTRODUCTION

1.1 This leaflet gives recommendations on the methods of complying with the requirements for the protection of fuel systems against enemy weapon effects.

2 PROTECTION AGAINST ENEMY WEAPON EFFECTS

2.1 The requirements for protection of the fuel system of combat rotorcraft will be stated in the Rotorcraft Specification. These requirements are stated in principle only and leave entirely for settlement between the rotorcraft designer and the Rotorcraft Project Director the methods to be adopted in a particular case. The methods are in three classes, based on:

(i) protection against fuel fire in voids outside the fuel tanks,

(ii) protection against fuel vapour explosion within fuel tanks,

(iii) prevention of excessive fuel loss due to perforation of fuel tanks.

2.2 FUEL FIRE SUPPRESSION

2.2.1 In general, integral fuel tanks are preferred since they reduce the number of structural voids adjacent to tanks to a minimum and fuel fire outside the structural envelope is unlikely to be sustained at speeds above 100 knots. Protection of structural voids may be achieved by active or passive devices. The choice will depend on the dimensions of the void, the presence of system components and the requirement for access, etc., since these factors determine the effectiveness and mass of particular methods of protection.

2.3 EXPLOSION SUPPRESSION

2.3.1 The generation of disruptive pressures due to ignition of fuel vapour/air mixtures in a tank ullage can be prevented by:

(i) installation of a suitable amount of reticulated polyurethane foam within the tank, or

(ii) purging the ullage with an inert gaseous mixture, or

(iii) maintaining a non-explosive fuel vapour pressure in the ullage.

The most suitable method for a particular installation will depend upon the total tank capacity, operating temperature and logistics.

2.4 PREVENTION OF EXCESSIVE FUEL LOSS

2.4.1 Self-sealing fuel tanks. The effectiveness of self sealing materials is currently limited to the sealing of slit type wounds produced by bullets of up to 12.7mm calibre and wounds produced by small shell fragments. A considerable time (up to 5 min) is required to develop a seal and these materials do not, therefore, contribute significantly to fire suppression. To ensure that these tanks develop their self-sealing qualities it is important that they are housed in properly designed compartments. This is also necessary to realise their inherently good crash resistant properties. Guidance on this subject is given in RAE Technical Note Mech Eng 120 and Specification DTD/RDI 3967.

2.4.2 Tank compartmentation. This is the only method currently available for the prevention of excessive loss of fuel from integral tankage and from internal tanks damaged beyond the capacity of self-sealing material. In general, it is necessary to distribute the fuel amongst a number of separate tanks and to arrange the fuel system so that there is a controlled distribution of fuel between these tanks in flight. The capacity of the individual tanks will be determined by the permissible quantity of fuel lost due to a single hit.

LEAFLET 702/5

FUEL SYSTEMS

CRASH RESISTANT FUEL SYSTEMS

1 INTRODUCTION

1.1 The overall objective of designing for crashworthiness is to eliminate unnecessary injuries and fatalities in an otherwise survivable accident.

1.2 A recent study carried out in the United States of America into survivable and partially survivable accidents and incidents to large civil transport aeroplanes showed that, of the 85 cases in the period examined, fire occurred after impact in approximately 42 per cent of the cases. This study also showed that a high percentage of the cases where some fatalities occurred involved a post crash fuel fire.

1.3 This leaflet presents some fuel system design guide lines which if followed should help in minimising the risk of fuel fed post crash fire.

2 GENERAL

2.1 The installation details of all fuel system components should be evaluated in relation to other system components in the same area and the strength of the component mountings should be checked for inertia crash induced loads. Possible or likely ignition sources should be examined in relation to predicted fuel spillage from broken fuel lines. Probable migration paths caused by spilt fuel should be evaluated in relation to possible ignition sources.

2.2 Although the effect of a crash landing cannot be precisely determined, past experience suggests that a reasonable effort to protect fuel system components from damage has helped in reducing, the risk of post crash fuel fires. Such design effort may result in the risk of post crash resistant fuel tanks, flexible fuel lines, self sealing break away fuel line couplings, drip fences, drainage troughs, non-conducting shrouds for electrical wiring and explosion proof electrical equipment.

3 FUEL SYSTEM ANALYSIS (see Chapter 712, para 4.2)

3.1 A crashworthiness analysis of the fuel system should be carried out to show compliance with the various requirements. The analysis should consider all systems and equipment the area of concern and it should evaluate the possible interaction between the various equipments and parts. In addition, it should take account of the effect of large deflections of structural members and the various modes of structural failures which could result from the large inertia forces, (specified in Chapter 307), which occur during a crash landing.

3.2 The fuel system analysis should be integrated with the work carried out to show compliance with Chapter 712. This combined fuel system and fire precaution analysis which should list all relevant combinations of fuel migration due to damage and hazardous potential ignition sources could be used to help assess the risk of ignition of fuel from combat action (see Chapter 702, para 10).

3.3 The analysis should consider the various crash cases specified in Chapter 307 and 702. The analysis should highlight the design precautions taken to protect the fuel tank installation against ruptures leading to fuel spillage. The more important of the crash cases are as follows:

(i) Crash resulting in the rotorcraft rolling over or the landing gear collapsing or the tearing away of the mountings for the landing gear or an engine or other high inertia item.

(ii) The effect of scraping action with the ground of the fuel tank installation if the rotorcraft slides along the ground after the initial impact.

(iii) Any other crash landing case, not covered by the above cases, where predicted survival conditions exist for the occupants, yet major structural failure occurs. For example, rotorcraft with fuel tanks mounted under the floor are vulnerable if the rotorcraft crashes, by descending at a high sink rate in a near level flight attitude, or rough ground. It is obvious that tanks mounted low in the fuselage will contact the around early in the crash sequence and will be exposed to possible penetration by rocks, stumps and other such ground irregularities. Such predicted cases should be discussed with the Rotorcraft Project Director before being included in the fuel system analysis.

4 STRUCTURAL ANALYSIS

4.1 If a structural analysis has been undertaken it will list the structural deflections and modes of failure. It can be used to:

(i) highlight areas of possible damage to fuel system components and fuel lines.

(ii) indicate those areas where the fuel system may be exposed to scraping action on the ground.

(iii) show possible migratory paths for fuel spilt or leaking from components of fuel lines.

5 LOCATION OF FUEL TANKS

5.1 Installation over the engine compartment, electrical battery or other potential ignition sources should be avoided. If this is not practical then precautions, such as shielding or use of flexible tanks, should be considered and discussed with the Rotorcraft Project Director (Chapter 712, para 11.4).

5.2 The installation should allow as much rotorcraft structure as possible to crush or be torn away before the tanks themselves are liable to damage or exposed to direct contact with the ground (Chapter 702, para 9.2 and 9.5).

5.3 If it is predicted that structure surrounding a tank may collapse due to compressive loads during the crash, areas into which the tank and its contents could expand should be provided if practical. Care should also be exercised in the design to ensure that when structural failure occurs, sharp edges and major structural members are unlikely to penetrate the tank and cause leakage.

5.4 Fuel tanks mounted externally to the rotorcraft, for example, pylon tanks, are particularly vulnerable to scraping on the ground during a crash landing. A suggested method of compliance with the need to reduce the risk of fire under these conditions is to design the tank so that it breaks away from the rotorcraft under the crash loads. In these circumstances the fuel lines should be equipped with self sealing couplings and the tank should continue to contain fuel without leakage following ground impact.

6 FUEL LINES AND COMPONENTS

6.1 Although the complete fuel system should be designed to contain its fuel both during and after an accident, release of fuel does sometimes occur due to severed or punctured fuel lines. Such spillage or leakage should be diverted or excluded to the maximum possible extent from spreading to likely ignition sources.

6.2 Fuel components and lines should be located and routed as far as is practicable from likely impact areas and from areas where structural deformation may cause crushing, severing, punctures, or high tensile loads in the lines. If possible, the fuel lines should be routed along the heavier structural members as these members are less likely to deform or separate in an accident.

6.3 Where the deformation of the structure might have an adverse effect on the fuel lines, flexible and stretchable hoses should be used or the fuel lines should be designed to allow stretching or movement with the deformed structure up to an amount likely to be required to prevent failure under high tensile or shear loads. If flexible hoses are employed it is important that the hose has space in which it can deform if necessary. If other design requirements limit the use of the protective measures discussed above, full use should be made of self sealing break-away couplings.

7 IGNITION SOURCES

7.1 Friction sparks are caused by parts of the rotorcraft structure scraping along the ground. Ignition depends on the thermal energy of the spark. Thermal energy is a function of bearing pressure, speed at which the material is scraped along the ground, hardness of the material and the temperature at which the metal particles will burn. Research has been conducted by NASA to determine the minimum conditions under which friction sparks from material typically used in rotorcraft structures will ignite. Some results of this research are give in the following table:

Material	Minimum Bearing pressure lb/sq.in.	Drag Speed m.p.h.
Titanium	21 - 23	Less than 5
Chrome-molybdenum steel	30	10
Magnesium	37	10 - 20
Stainless steel	50	20
Aluminium	1455+	40

+Note: Ignition was not obtained with aluminium.

7.2 There are two solutions for reducing the risk of a post crash fuel fire resulting from friction sparks. One is to ensure that fuel is not present in areas where sparks are possible. The other solution is to construct those areas, which might slide along the ground in a crash, of material which is unlikely to produce sparks capable of igniting spilt fuel. The best results can be produced by adopting the principles of both solutions.

7.3 Electrostatic sparks result from the discharge of an electrostatic charge accumulated on parts during normal operation. The discharge may be triggered by the separation of the parts. Fortunately the combination of a crash causing the relevant parts to separate and the environmental conditions which result in sparking only occur infrequently.

LEAFLET 702/6

FUEL SYSTEMS

ROTORCRAFT FUEL SYSTEM - CONTAMINATED FUEL QUALIFICATION TEST PROCEDURE

1 INTRODUCTION

1.1 This leaflet gives details of laboratory tests suitable for demonstrating the ability of components of fuel systems to meet the requirements of Chapter 702, para 25.1.

1.2 In-service experience and specific equipment reliability test programmes have demonstrated the desirability of confirming that fuel systems' components will function correctly in the presence of contamination, (typical of that found in military rotorcraft fuel tanks and capable of being transmitted through the fuel system pipework and equipment under normal operating conditions).

2 SCOPE

2.1 This test provides requirements for demonstrating the resistance of rotorcraft fuel system components to resist typical in-tank contaminants, complementing the existing DERD 2153 specification. To enable this to be achieved, requirements for the manufacture of contaminants, typical of military rotorcraft fuel systems having a variety of design features, are given, and recommendations made for fuel system component testing and test rig design.

2.2 Whilst the types of contaminant specified are considered to be typical of those found in rotorcraft fuel systems, this test is designed as an accelerated test to pinpoint equipment that is susceptible to contaminated fuel and the levels of contamination described herein are not to be taken as representative of those likely to be achieved in service.

3 APPLICATION

3.1 The test may be used alone or in conjunction with DERD 2153 for the qualification of equipment likely to be subject to contamination. This equipment will be, primarily, in fuel transfer and engine feed lines as well as in refuelling and vent system lines. The applicability of the test and contaminants to be considered will be determined by the rotorcraft designer.

3.2 The test has been prepared to allow application to any military rotorcraft project.

4 MANUFACTURE OF REPRESENTATIVE CONTAMINANTS

4.1 The contaminants to be used in the testing of fuel system components should be selected by the rotorcraft designer from the appropriate sections of Table 1, according to the design features of the fuel system concerned. Table 1 makes provision for the presence or otherwise of pump inlet strainers, and the presence or otherwise of various in-tank explosion suppressant materials. These contaminants are those which, on the basis of experience, can be expected to be found in the fuel tanks of a military rotorcraft in service, in spite of measures adopted to maintain cleanliness during build and maintenance, and which can be expected to be passed into fuel transfer and feed pipework during normal system operation. The size of contamination particulate which can be passed into the system will depend upon whether mesh screens have been used at the pumps and/or at pipework inlets, and the size of such screens, if used.

4.2 Recommended techniques for the manufacture of representative samples of contaminants are given in Table 1, which also defines the concentration of contaminant to fuel to be used for testing.

4.3 Para 8 lists the information required to define the contaminants likely to be found in the fuel tanks of any specific rotorcraft type.

5 TEST RIG DESIGN

5.1 A suitable test rig should have the following features (although the detail arrangements will be dependent upon the type of equipment to be tested by the equipment supplier, and must be approved by the rotorcraft designer):

(i) Capacity: Minimum test tank capacity 150 litres for flow rates up to 2 litres/sec. A larger capacity is desirable for equipment passing higher fuel flow rates.

(ii) Equipment mounting. Provision should be made for the in-tank or line mounting, in a representative manner, of the equipment under test.

(iii) Agitation. Return fuel or other means, should be used to provide sufficient agitation to maintain contaminants in suspension in the fuel.

(iv) Fuel circulation. A pump capable of providing a fuel flow rate equivalent to the maximum operating flow of the equipment under test should be provided. (This pump may be the equipment under test). Where appropriate, provision should be made for varying the flow through the equipment under test during the test cycle.

(v) Contaminant sampling. Means should be provided to check that contaminant circulation is being maintained during a test cycle (e.g., filter loop).

(vi) Contaminant removal. Means should be provided for removal of the contaminant from the test tank following completion of a test cycle, to prevent accumulation of contaminant in the test rig, thus ensuring consistency of testing .

(vii) Flow rate. A flow meter should be incorporated to ensure that the component can be tested over the full range of expected rotorcraft fuel flows. To avoid the effects of the contaminants on the calibration of the flow meter, the flow control valve should be calibrated against the flow meter prior to the test, using clean fuel. If doubt exists as to whether the flow meter could affect the free circulation of contaminant, then the flow meter should be located as shown in the filter loop and switched out during the actual test.

(viii) Fig 1 presents a schematic of a suitable rig for these tests.

6 TEST PROCEDURE

6.1 CONTAMINANTS

6.1.1 The specified contaminants should be mixed with fuel in the proportions of 1 unit of each contaminant to 25 litres of fuel, (see Table 1 for definition of "unit" of contaminant).

6.2 TEST CYCLE

6.2.1 Contaminated fuel should be circulated through the equipment during a number of agreed duty cycles totalling a minimum of 10 hours, with the equipment operating in its normal mode. The equipment operating cycle should be specified by the rotorcraft designer.

6.3 FUEL TEMPERATURE

6.3.1 In tests using the contaminants as defined in Table 1, fuel may be circulated at room temperature. However, where the characteristics of the contaminants of the equipment under test change with temperature, testing at both high and low fuel temperatures should be considered. The appropriate temperatures and the necessity for temperature testing should be specified by the rotorcraft designer.

6.4 CONTAMINANT CIRCULATION

6.4.1 This should be checked at intervals during the test cycle by means of a filter loop, sight glass or sampling point at the outlet of the equipment under test. Any samples taken from the equipment outlet or filter during the test cycle should be returned to the test tank, to ensure that the contaminant concentration is maintained. Agitation should be sufficient to keep the contaminants in suspension (see para 5.1(iii)).

6.5 FUNCTIONAL CHECKS ON EQUIPMENT UNDER TEST

6.5.1 The equipment under test must pass a manufacturers' acceptance test prior to, and after, contamination resistance testing, and should perform correctly during a test cycle. The final acceptance test should be followed by a strip examination to establish the existence of any damage resulting from the test.

6.6 EFFECTS OF ADJACENT EQUIPMENT

6.6.1 Consideration should be given to the effects of adjacent equipment on the flow through the item being tested, (e.g., swirl induced by pipework or other equipment immediately upstream or downstream). In certain cases, it may be necessary to test a complete sub-system of a few components linked together by the appropriate pipework. However, in general, it will not be cost effective to test a complete fuel system unless one has been built for other reasons.

7 PRODUCTION OF TANK SEALANT CONTAMINANT SAMPLES

7.1 The process described below will permit the manufacture of a reproducible sample of sealant particles giving an adequate representation of actual rotorcraft tank contamination, when tank sealants of the P.R.C. type, or similar, are used.

7.2 Sealant material of the type or types used for wet assembly and oversealing should be mixed and poured into suitable small containers to cure. From the cured materials, small blocks of approximate dimensions 15 x 5 x 5 mm and mass approximately 7 to 8 grams, should be cut. One of these will produce 5 units of contaminant, as defined in Table 1, when processed as follows. The block sealant is fed into a 'mincing' machine, having a very coarse screw feed and a rotating cutter with 10 x 6.5 mm dia holes. The minced sealant must be collected in a suitable container, and any particles remaining in the mincer, subsequently extracted and added to the collected material.

7.3 For fuel systems using screens, the sealant particles produced as above are to be sieved through a screen mesh, of corresponding size, before use in component evaluation tests. In this case, 1 unit of the sieved material shall be the amount derived from 1 unit of the un-sieved material.

Notes: 1 A mix of sealant types may be used, to adequately represent likely tank contamination on any particular rotorcraft.

2 A suitable mincing machine is a domestic kitchen hand operated mincer (e.g., 'Spong' type).

8 INFORMATION REQUIRED TO DEFINE CONTAMINATION TEST REQUIREMENTS FOR A SPECIFIC ROTORCRAFT TYPE

8.1 The following, lists the information required to define the contaminants likely to be found in the fuel tanks of any specific rotorcraft type:

(i) Tank structure - integral tanks.
Predominant structure material type.

(ii) Tank sealants - integral tanks overcoating sealant type.
Wet assembly sealant type.

(iii) Electrical cable types - general electrical cable type appropriate to rotorcraft (not special screened or co-axial types unless widely used and likely to get into rotorcraft tanks).

(iv) Explosion suppressant foam.
If used, specify type of in-tank explosion suppressant foam material.

TABLE 1

FUEL CONTAMINANT CONCENTRATION AND MANUFACTURE

	Drilling swarf	**Tank sealant**	**Electrical cable**	**Explosion suppressant**
Fuel systems with boost/ transfer pump/ tank pickup screens fitted	1 unit consists of the swarf generated by drilling 2.4, 3.2 and 4.0 mm holes in 1.2 mm thick aluminium alloy material of the type used in the tank and adjacent structure of the rotorcraft. The resulting swarf is to be sieved through a mesh identical to that used in the fuel system to collect a sample of mass 40 milligrams. (To produce the required sample mass, sufficient <u>sets</u> of all three holes sizes given above should be drilled). Concentration - 1 unit to 25 litres of fuel.	1 unit consists of particles of cured tank sealant, of irregular shape and a range of sizes, produced by a 'mincing' process. The 'minced' material is to be sieved through a mesh screen identical to that used on the fuel system, to collect a sample. For details of the contaminant sample production procedure, see para 7. Concentration - 1 unit to 25 litres of fuel.	1 unit consists of 1 piece of cabling 3 to 4 mm long taken from the range of cable types used on the rotorcraft, and of a small enough diameter over the insulation to pass through a mesh identical to that used in the fuel system. Note: Cable should only be included if it is possible for a sample as defined above to pass through the screens into the system's pipework/components. Concentration - 1 unit to 25 litres of fuel.	Experience has shown that currently available in-tank explosion suppressant foams, in general, have a tendency to 'strand', the strands then being likely to pass through any fuel system screens. For both reticulated polyurethane and melded polyamide foams, therefore, 1 unit of contaminant is a small piece of foam 10 mm long, which will not fall through a 5 mm hole, but can be pushed through it with care. These particles are to be separated manually from bulk material. Concentration - 1 unit to 25 litres of fuel.
Fuel systems without inlet screens	1 unit consists of the swarf generated by drilling a 4 mm hole through 1.2 mm aluminium alloy material of the type used in the tank and adjacent structure of the rotorcraft. Concentration as above.	1 unit consists of the particles of sealant resulting from 'mincing' a total mass of sealant of 1.5 grams. For details of the contaminant sample production see para 7. Concentration as above	1 unit consist of 1 piece of cable 3 to 4 mm long taken from the range of cables used on the rotorcraft. Diameter over the insulation to be in the range of 1.0 to 1.5 mm. Concentration as above	As above. Concentration as above.

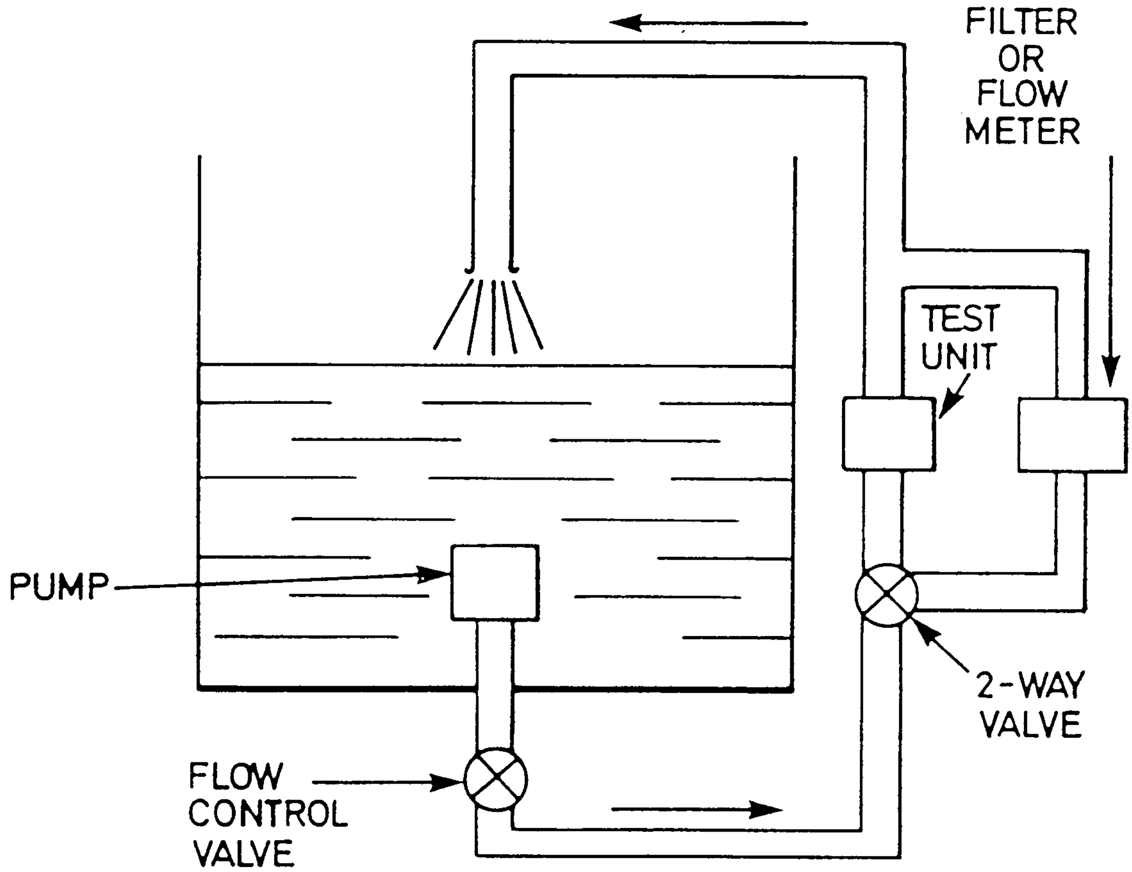

FIG.1 FUEL CONTAMINANT RIG

CHAPTER 703

PNEUMATIC SYSTEMS

1 INTRODUCTION

1.1 The requirements of this chapter apply to all pneumatic systems in Rotorcraft both static (low mass flow) systems used for power transmission and dynamic (high mass flow - total loss) systems used mostly for environmental control. Alternative requirements are given where necessary.

1.2 The requirements apply to all components of the pneumatic system but the requirements of Chapter 719 (Pressurised Gas Storage Vessels) may also apply to some. Any component, or part of the system, or combination of these, having a medium or high energy rating (see Leaflet 719/1) shall meet the relevant requirements of Chapter 719.

1.3 The applicable temperature/altitude requirements will be found in Chapter 101 or the Rotorcraft specification.

1.4 See Leaflet 703/1 for recommendations about the protection of Pneumatic Systems against the ingress of foreign matter.

1.5 See Leaflet 703/2 for recommendations on how to reduce the explosion hazard.

1.6 See Leaflet 703/3 for recommendations about Hoses.

1.7 See Leaflet 703/4 for recommendations about Tubing.

1.8 See Leaflet 703/5 for recommendations about V-Flange Couplings

1.9 See MIL-SPEC Mil-P-5518 for further background information and equivalent American requirements.

2 GENERAL

2.1 The capacity and output of the storage vessel(s) and/or compressor(s) shall be sufficient to operate all normal services which may need to be used simultaneously and all combinations of normal and emergency services which could reasonably be expected to be needed at the same time, against the appropriate external loads.

2.2 All pneumatically operated services, which are essential to safety in flight or landing shall be provided with an alternative source of power, not necessarily pneumatic, for use in an emergency. Emergency systems shall be completely independent of the main systems up to, but not necessarily including, the actuator. Any part of either system which is also part of the power plant shall comply with power plant design requirements (see Chapter 700).

2.3 The system shall be so designed and installed as to ensure its satisfactory functioning under all expected conditions.

2.4 Consideration shall be given to the effect of the ambient temperatures to which the various components may be subjected (see para 3) and to other environmental effects, such as vibration, abrasion, corrosion and mechanical damage in service.

2.5 When two sub-systems are powered by a common source, and one is essential to safety in flight and the other is essential to safety in landing each shall have an emergency system (para 2.2) and, a vulnerability analysis (Chapter 112) shall be done to show whether the two systems should be fully isolated.

2.6 Any part of the system that would be adversely affected by foreign matter, or oil or water contamination, shall be adequately protected by filters or other devices which, if they require routine servicing, shall be readily accessible (see Leaflet 703/1).

2.7 Where differential motion or vibration exists between any two points in a pneumatic line compensation shall be provided by the use of flexible connections or other suitable devices.

2.8 Adequate indication of system and sub-system condition shall be provided to the crew to ensure safety in flight and landing and to ensure that correct action can be taken in the event of any malfunction.

2.9 External test connections from which all services can be recharged and tested for correct operation while the rotorcraft is on the ground, shall be provided. These connections shall be readily accessible and incapable of incorrect assembly (see Chapter 100 para 7). Adequate charging pressure and temperature data shall be placarded on the structure near the connections.

3 TEMPERATURE EFFECTS

3.1 Blockage caused by freezing shall not occur on the ground or in the air. Where necessary anti-freeze and/or pressure relief devices shall be provided.

3.2 Where necessary drains shall be provided at low points in the system to permit removal of condensation.

3.3 Provision shall be made, where necessary, to cool the compressors, under all conditions of flight and ground operation, such that the design life and reliability targets are met.

3.4 Pressure limiting devices shall not be installed where they are liable to freeze up. If their design is such that they are liable to be adversely affected by heat they shall not be fitted in positions where their operating temperature, either on the ground or in flight is expected to be high.

3.5 Consideration shall be given to relative expansion and contraction between the system components and the airframe. See Leaflets 703/3 and 703/4.

4 SAFETY AND VULNERABILITY

4.1 The system shall be designed to comply with the requirements of Chapter 100 para 9. 1 and of Chapter 112.

4.2 When duplicate lines are provided they shall be so located as to minimise the probability of both being damaged by a single threat effect (see Chapter 112), tyre burst, non-containment of an internal engine failure, local structural failure or other hazardous event.

4.3 As far as possible all components shall be so installed that their bursting would not be likely to cause catastrophic failure of any part of the rotorcraft or any injury to any occupant.

4.4 No component which would be liable to explode if subject to a fuel fire shall be mounted within an engine bay or other designated fire zone.

4.5 Means shall be provided (e.g., valves, fuses), as far as practical, to isolate ruptured circuits and prevent complete loss of power in both the normal and emergency systems.

4.6 Shuttle valves shall not be used in installations in which a force balance can be obtained on both inlet ports simultaneously which may cause the shuttle valve to restrict flow from the outlet port. Where shuttle valves are necessary to connect an actuating cylinder with the normal and emergency systems, the shuttle valve unit shall be built into, or attached to, the appropriate cylinder head.

5 DEFINITIONS

5.1 P_W. The normal working pressure for which a particular part of the system is designed, and which must not be less than the minimum pressure necessary for efficient functioning of that part of the system.

5.2 P_r. The relief pressure associated with a particular part of the system and fixed at a value which allows a reasonable margin above P_W. A value of 1.33 P_W is implied by the strength requirements of this chapter and covers the effects of variability in maximum delivery pressure in service (nominally 10%), supply control failure, ingress of foreign matter, filter blockage (Leaflet 703/1), and temperature changes but not transients. In some projects a value of P_r greater than 1.33 P_W may be necessary to prevent excessive loss of gas from the system following a cold soak at altitude and a rapid descent to ground level in a high temperature.

5.3 TPA (Transient Pressure Allowance). The pressure allowance above Pr for short duration increases in pressure, arising from solenoid operation or similar causes, which do not last long, enough to cause the relief valve to crack. The allowance should be based on experimental evidence if available. If no relevant evidence is available the allowance should not be less than 50% of P_r.

5.4 P_c. The maximum permissible charging pressure at 20°C for which the system is designed.

5.5 R. Design Pressure Ratio. The ratio of pressure at the maximum design temperature to the pressure at 20°C, obtainable from standard tables for the gas used.

5.6 P_d. The design pressure for a component or part of the system.

For static systems $P_d = P_r + TPA$

For dynamic systems $P_d = (P_c \times R) + TPA$ or $P_d = P_w + TPA$ as appropriate.

6 PRESSURES

6.1 The system may be designed to operate at any pressure(s) consistent with the particular needs of the services operated.

6.2 The pneumatic power supply shall incorporate, or work in conjunction with, pressure regulating devices such that within their design tolerances the pressure supplied to any part of the system, both during and on completion of the normal operation of any service, is not greater than P_w.

6.3 Means shall be provided, which within their design tolerances, will prevent the pressure in any part of the system exceeding the relevant P_r, both during normal operation and during ground charging.

6.4 The system shall be so designed that the intended functioning of the components will not be adversely affected by the highest back pressure resulting from operation of any part of the system.

6.5 Transient pressures shall be damped wherever possible. Any residual transients, shall be included in the fatigue analysis for the parts affected.

7 STRENGTH

7.1 The strength of each part is defined in terms of its design pressure P_d as defined in para 5.6.

7.2 COMPONENTS

7.2.1 All pneumatic components shall be designed to withstand, at most adverse working temperature:

(i) without leakage (but see para 7.4 below) or permanent distortion, a design proof pressure not less than $1.125P_d$.

(ii) without fracture or bursting, an ultimate pressure not less than $1.5P_d$.

together with, in each case, the most adverse loads, similarly factored, that can occur at the same time from the operation of any service or from flight, take-off or landing.

7.2.2 When a pneumatic component forms part of a mechanism which has to withstand externally applied loads during flight, take-off or landing, the required proof and ultimate factors shall be realised in the current strength cases when the most adverse effects of pressure and acceleration which can occur at the same time are included in the loading conditions.

7.3 PIPES AND PIPE COUPLINGS

7.3.1 All pipes and pipe couplings shall be designed to be capable of withstanding at most adverse working temperature, without leakage (but see para 7.4 below) or permanent distortion, a proof pressure of $1.125P_d$ and without fracture or bursting an ultimate pressure of $2.25P_d$. (See Leaflet 703/5 for recommendations about V-Flange Couplings)

7.4 LEAKAGE AND DISTORTION

7.4.1 In addition to the conditions of Chapter 200, para 4.3 such leakage or distortion as might directly prevent the operation of any part or would in the course of one flight be likely to render the system inoperative, shall be regarded as a failure. On the other hand, in dynamic systems, (see para 1.1 above) some leakage may be permitted from joints, and from some components. The standard shall be agreed with the Rotorcraft Project Director.

7.5 FATIGUE

7.5.1 Where any part of the system is subject to fluctuating or repeated external or internal loads, due allowance shall be made for fatigue. (See Chapter 201). The effects of variability of compressor delivery pressure and of transient pressure changes shall be included.

8 STRENGTH TESTS

8.1 SAFETY PRECAUTIONS

8.1.1 For safety reasons pressure tests on pneumatic components shall first be done hydraulically. The hydraulic fluid used shall be such that it will not cause damage to, or deterioration of, any part of the system with which it will come into contact.

8.1.2 Where it is considered that tests at high temperature are too hazardous, an additional test factor shall be applied to the required pressure.

8.2 STATIC TESTS - COMPONENTS

8.2.1 Prototypes of all pneumatic components shall be tested to establish compliance with the requirements of para 7.2.

8.2.2 In cases where the shape of the component is such that reliable calculations of ultimate strength can be made, and if failure is unlikely to cause damage additional to that caused by the loss of pneumatic pressure, tests to ultimate conditions need not be made unless called for. But see also para 1.2.

8.3 STATIC TESTS - PIPES AND COUPLINGS

8.3.1 Prototype pipe couplings in conjunction with the appropriate piping shall be tested to establish compliance with the requirements of para 7.3. But see also para 1.2.

8.4 PROTOTYPE SYSTEM TESTS

8.4.1 The prototype system shall be tested at a pressure of P_d to show that functional and leakage requirements have been met.

8.4.2 The test of 8.4.1 shall be continued to $1.125P_d$ without deformation of any part of the system that would prevent it from performing its intended function. Clearance between parts of the system and the structure must be adequate and there shall be no permanent detrimental deformation. Such leakage as occurs shall meet the requirements of para 7.4.

8.4.3 For the purpose of these tests, valves in components shall be adjusted where necessary to obtain correct local pressure.

8.5 FATIGUE TESTS

8.5.1 Each pneumatic component, pipe or coupling, shall have a safe life at least equal to the specified life of the rotorcraft unless otherwise agreed. This shall be demonstrated by fatigue tests which shall be on lines similar to those of Leaflet 704/2 for hydraulic systems. Where possible these tests should be integrated with the Endurance and Environmental tests of para 9 to improve simulation of true service conditions.

8.5.2 Fatigue tests may not be required on components such as compressors for which a separate type test schedule is specified, nor on some components, such as certain items in an emergency circuit which are not subject to pressure applications during normal flying.

9 FUNCTIONING, ENDURANCE AND ENVIRONMENTAL TESTS

9.1 The satisfactory operation of the complete pneumatic system shall be demonstrated on a prototype or development rotorcraft or on a representative ground rig of the system, or appropriate sub-systems, to a schedule agreed with the Rotorcraft Project Director. Tests shall be included to demonstrate compliance with paras 6.2, 6.3 and 6.4, and also to show that, at all operating temperatures, oil, water or other impurities in the system do not adversely affect the operation of the system including the functioning of the relief valves (see para 2.6). Consideration shall be given to the possibility of ice formation.

9.2 The functioning tests shall include simulation of pneumatic system failure conditions. The tests should also include relevant flight loads, ground loads and pneumatic system working loads, relief loads, limit loads, and transient pressures expected during normal operation. Vibration loads and loads caused by temperature effects may be included where relevant subject to the agreement of the Rotorcraft Project Director.

9.3 Endurance tests shall simulate the repeated complete flights that could be expected to occur in service. Elements which fail during the tests shall be modified in order to have the design deficiency corrected and where necessary must be sufficiently retested.

9.4 Tests simulating operating and environmental conditions shall be done on elements and appropriate portions of the pneumatic system to the extent necessary to evaluate the environmental effects.

10 PRODUCTION TESTS

10.1 The drawings of all pneumatic components and pipe coupling assemblies shall specify a test pressure equal to P_r for the relevant part of the system, and shall state that all parts shall function correctly, that no permanent distortion shall occur when the test pressure is applied, and that the leakage requirements of paras 7.2, 7.3 and 7.4 shall be met.

10.2 The drawings of the complete system shall specify a test pressure P_w or P_c as appropriate for the main high pressure part of the system and shall state that the system shall function correctly, and that such leakage as occurs when this pressure is applied shall meet the requirements of para 7.4. The resulting pressures in the low-pressure sections of the system, with the pressure reducing valves correctly set, will be considered to be the proof test pressure for these portions of the system.

LEAFLET 703/0

PNEUMATIC SYSTEMS

REFERENCE PAGE

RAE Technical Notes

SME 377	Flow of high pressure air in small pipes.
Chem 1108	The cubical coefficients of expansion of rubbers used in hydraulic and pneumatic systems.
Chem 1118	The compounding of natural and synthetic rubber to control the cubical coefficients of expansion

R.A.E. Specifications

AD 139	Rams, pneumatic

British Standards Specifications

C 9	Coupling dimensions for high pressure air charging valves for aeroplanes
C18	V-Flange couplings for aircraft piping and ducting - metric series.
M23	Specification for an identification scheme for pipelines
M48	Dimensions of elastomeric toroidal sealing rings for aerospace use - inch series
2917	Specification for graphical symbols used on diagrams for fluid power systems and components

Defence Standards

05-56	Graphical symbols for aircraft hydraulic and pneumatic systems.
16-7	Compressed non-breathing air characteristics, supply pressure and hoses for aircraft systems.
16-26	Charging valves for aircraft high pressure nitrogen and hydraulic fluid systems
47-12	Polytetrafluoroethylene (PTFE) hose assemblies for medium and high pressure fluid systems in aeroplanes.

47-22	End fittings for flexible hose assemblies for aeroplanes - metric.
47-25	Pipelines and pipe couplings for aeroplane fluid systems (metric).
53-68	V-Flange couplings for Aircraft piping and ducting (metric).
81-24	Identification marking of cylinders, compressed gases

American Mil-Specs

MIL-P-5518	Pneumatic systems, Aircraft, Design, Installation and Data Requirements for.

LEAFLET 703/1

PNEUMATIC SYSTEMS

PROTECTION AGAINST THE INGRESS OF FOREIGN MATTER

1 INTRODUCTION

1.1 This leaflet describes the possible sources of contamination of pneumatic systems which should be taken into account when complying with the requirements of Chapter 703, para 2.6.

1.2 In servo systems where close tolerances between working parts are critical, protection by filters or other devices is essential. In other systems, however, where large air flows occur, fine filtration may not be necessary.

2 FLUID CONTAMINATION

2.1 Oil may enter a pneumatic system:

(i) When the normal air supply is via a tapping from the rotorcraft engine compressor, the working parts of which are oil lubricated, and

(ii) During ground checks and/or testing in cases where an external air supply is used, particularly if the ground support equipment concerned incorporates an engine-driven compressor, and more so if the servicing of the ground support equipment is suspect, i.e., away from a fixed, properly supervised base, or in the field.

2.2 Water may enter a pneumatic system during ground checks and/or testing under the conditions of para 2.1 (ii) above and also possibly when workshop high pressure air supply is used.

3 SOLID FOREIGN MATTER

3.1 Solid foreign matter may enter a pneumatic system:

(i) By being ingested into the rotorcraft engine compressor thereafter passing by way of the tapping into the pneumatic system.

(ii) Through external connections used during ground servicing and/or testing. This can occur when using ground support equipment in circumstances similar to those referred to in para 2.1(ii) above. It can also occur through breakdown of the internal surfaces of ground support equipment, e.g., due to flaking or rusting of the pipelines of workshop high pressure air supply pipelines.

4 FILTER BLOCKAGE

4.1 Filters used to guard against contamination may, depending on the type of contamination, be liable to complete blockage, or to restriction such that the performance of the system is affected. Suitable relief and by-pass arrangements should, therefore, be included to meet this eventuality.

4.2 Alternatively a clogging indicator to ensure timely changing of the filter may be provided if agreed with the Project Director.

4.3 In either case suitable instructions must be included in the servicing handbook.

5 DEHYDRATORS

5.1 Dehydrators may be used where necessary to remove moisture introduced by the compressor(s). They may also be used to remove all but traces of compressor lubricating oil.

LEAFLET 703/2

PNEUMATIC SYSTEMS

EXPLOSION HAZARD

1 Under certain conditions of pressure and temperature, the combination of air with lubricant combustible materials may cause damaging ignitions and explosions. The designer should attempt to so design and install the system that:

(i) The pressure and temperature conditions are not inductive to combustion or explosion.

(ii) Percentages of lubricant and combustible materials tending to cause or sustain combustion are controlled and held to a minimum.

(iii) Large cavities, in which high energy combustions or explosions could occur, are avoided as much as possible. See also Chapter 719.

(iv) Excessively high compression ratios, and resulting temperature peaks, are avoided, particularly in the compressor, if one is used; and pressure or temperature responsive devices are used to temporarily suspend the compressor activity whenever there is a danger of approaching dangerous temperatures.

(v) Lubricating oils or greases which are (basically or because of additives) prone to induce combustions or explosions are avoided.

(vi) All materials used in the system are chosen after taking into consideration their merits in not sustaining combustion or enhancing explosion; and in not wearing out, eroding, or weakening in service in a manner that might cause substantial weakening of any part of the system.

2 Where the possibility of combustion or explosion cannot be precluded, all steps should be taken to minimise the effects of combustion or explosion on personnel and equipment, such steps may include flame arresters and blowout disks.

LEAFLET 703/3

PNEUMATIC SYSTEMS

HOSES

1 JOINT TYPES

1.1 Preferably, where there is relative motion between two connections, hoses should be used. However when design conditions permit, swivel joints or flexible steel lines may be used, subject to the approval of the Project Director.

2 INSTALLATION

2.1 Hoses should be so installed that they will not be subject to torsion under any condition of operation; and there should be no tendency for their connecting fittings to loosen.

3 HOSE SUPPORT

3.1 The support of a flexible line should be such that it will never tend to cause deflection of the rigid lines under any possible relative motion that may occur. Flexible hose between two rigid connections may have excessive motion restrained where necessary, but should never be rigidly supported, as by a tight clamp around the flexible hose.

4 HOSE BEND RADII

4.1 The minimum radius of bend of hose assemblies will be a function of hose size and flexing range to which the hose installation will be subjected, and will be specified by the designer.

5 HOSE PROTECTION

5.1 Hoses should be suitably protected against chafing wherever this is expected to occur.

6 PROVISION FOR HOSE ELONGATION AND CONTRACTION

6.1 Hose assemblies should be so selected and installed that elongation and contraction under pressure, within the hose specification limits, will not be detrimental to the installation either by causing strain on the end fittings or by binding or chafing of the hose.

LEAFLET 703/4

PNEUMATIC SYSTEMS

TUBING

1 MATERIALS

1.1 Only corrosion-resistant steel tubing should be used in pressure lines and for all tubing mounted on shock struts. Proposals to use alternative materials should be discussed with the Project Director.

2 SYSTEM TUBING CORROSION PROTECTION

2.1 Aluminium-alloy tubing in exposed areas such as landing gear wheel wells and missile bays, should be protected against corrosion, particularly under the sleeve and nut couplings.

3 DESIGNED MOTION IN TUBING

3.1 Looped or straight aluminium-alloy tubing should not be used between connections where there is designed relative motion. Relative motion is allowable between the ends of steel tubing if the combined calculated stress resulting from Bourdon effect, torsion, tension, and compression, as applicable, due to the relative motion, is less than 10 per cent of the ultimate strength of the tubing and vibration of the tubing mass does not have a detrimental effect. Endurance tests of such installations may be required at the option of the Project Director.

4 STRAIGHT TUBE LINES

4.1 The use of straight tube lines installed between two rigid connections should be avoided wherever possible. Where such straight lines are necessary, provision should be made in the mounting of the units or rigid connections to ensure that no excessive strains will be applied to the tubing and fittings. Semiloops may be provided in the tubing, as necessary, to ensure proper alignment on installation and to take care of vibratory motion.

5 TUBING IDENTIFICATION

5.1 All pneumatic lines should be permanently marked in accordance with Chapter 806, para 5, and BS M23. A sufficient number of pneumatic lines should be marked in conspicuous locations throughout the rotorcraft so that each run of line may be traced. This marking should indicate the unit operated and the direction of flow. These markings should be repeated as often as necessary, particularly on lines entering and emerging from closed compartments.

6 LOCATION OF PNEUMATIC TUBING

6.1 Insofar as practicable, pneumatic lines should not be installed in the cockpit or cabin and should be remote from crew stations. All tubing should be so installed that accumulated moisture will drain to the reservoirs or to specially provided drain points.

7 MISCELLANEOUS

7.1 Provision should be made in clamp location to provide for change in tubing length due to contraction and expansion. Removal and replacement of tubing should be possible without removal of components.

LEAFLET 703/5

PNEUMATIC SYSTEMS

V-FLANGE COUPLINGS

1 INTRODUCTION

1.1 This leaflet draws upon in-Service experience to outline the factors that should be considered when designing aircraft systems which incorporate V-Flange couplings.

1.2 V-Flange couplings are used widely in aircraft pneumatic and fuel systems. They are most commonly used within bleed air and environmental control systems and provide a strong, lightweight alternative to heavy bolted flange joints.

1.3 In-Service experience has shown that the most significant factors in failures of V-Flange couplings in aircraft systems are mal-alignment of rigid pipe flange mating faces and failure to seat the coupling correctly over the mating faces.

2 PIPE FLANGE ALIGNMENT

2.1 The designer should consider the alignment of the pipe mating faces, particularly where tolerances can build up through the system, to ensure force does not have to be applied to one or both pipe mating flanges to align them to allow the coupling to be fitted.

3 COUPLING ACCESS

3.1 The designer should consider maintainability of couplings including provision of adequate access for fitting and removal. Where inadequate space is available consideration should be given to the installation of overheat detectors.

4 COUPLING MARKINGS

4.1 Coupling markings should be in accordance with chapter 404 of this document and should not include clamp nut torque figures.

CHAPTER 704

HYDRAULIC SYSTEMS

1 INTRODUCTION

1.1 The requirements of this Chapter apply to all hydraulic systems.

1.2 A hydraulic system comprises a source of hydraulic power, distribution lines and components, control devices, actuating units, accumulators, fluid reservoirs, and filters.

2 HYDRAULIC SYSTEM TYPES

2.1 PRIMARY SYSTEMS

2.1.1 Hydraulic systems which are dedicated to functions that are essential to flight safety (e.g., primary flight controls).

2.2 UTILITY SYSTEMS

2.2.1 Hydraulic systems which serve functions which are not essential to flight safety (to be agreed with the Rotorcraft Project Director).

3 HYDRAULIC SYSTEM PRESSURE/TEMPERATURE CLASSIFICATION (REF BS M51-ISO 6771)

NOMINAL PRESSURE CLASSIFICATIONS NOMINAL SYSTEM PRESSURES (Pw) SHALL BE CLASSIFIED AS FOLLOWS:

CLASS	PRESSURE
A	4000 kPa (40 bar)
B	10500 kPa (105 bar)
C	16000 kPa (160 bar)
D	21000 kPa (210 bar)
E	28000 kPa (280 bar)
F	40000 kPa (400 bar)
G	50000 kPa (500 bar)

TEMPERATURE TYPES SYSTEM OPERATING TEMPERATURE RANGES SHALL BE CLASSIFIED AS FOLLOWS:

TYPE	TEMPERATURE RANGE
I	-55 to 70°C
II	-55 to 135°C
III	-55 to 200°C
IV	-55 40 320°C
V	-55 to 400°C
VI	-55 to 650°C

4 HYDRAULIC FLUID

4.1 The type of hydraulic fluid to be used in the system will be stated in the rotorcraft specification. (see Leaflet 704/0).

4.2 All materials in contact with the fluid and in particular elastomeric sealing materials, shall be compatible with the hydraulic fluid over the temperature range, functional, service and storage conditions to which the hydraulic system will be exposed.

5 DEFINITIONS

5.1 Pw - The nominal system supply working pressure within its design tolerance, see para 6.1.

5.2 Pr - The relief pressure is the maximum pressure permitted by any pressure relief means provided in compliance with para 8.1.6 within its design tolerance.

5.3 Pd - The design pressure for any component or part of the system. This includes the working pressure (Pw) plus the effect of any intensification of pressure during operation, or from external loading, and from transient peak pressures that may occur. See para 8.1.3. and 8.14.

6 HYDRAULIC POWER SUPPLY

6.1 Means shall be provided to maintain the pressure in any part of the system without excessive fluctuations during normal operation of any service. These means shall ensure that, under any possible condition of operation, the maximum pressures never exceed the proof pressure and the minimum pressures never become less than that necessary for efficient operation. The design aim shall be to maintain the steady state system pressure at Pw ± 10%. Also see para 5.1 and 8.1.4.

6.2 The power output from the power supply, including the accumulator(s) shall be sufficient to allow all services to satisfy their respective operating requirements.

6.3 The requirements of paras 6.1 and 6.2 shall be met when the services are operated singly and also when the most adverse combination of services likely to be operated during operational flying is used (e.g., under approach conditions), including engine out conditions on multi engined rotorcraft.

7 FLUID CAPACITY AND RESERVOIRS

7.1 Provision shall be made in the reservoir for a minimum reserve of fluid equal to at least 25 per cent of the fluid between the maximum and minimum levels (or 1.0 litre if this is greater). This reserve shall be available when the fluid is at its minimum level and the rotorcraft is in the most adverse attitude, corresponding to that combination of services which produces the minimum level.

Note: This does not apply to completely self contained power units.

7.2 The system shall be designed to minimise possible changes in reservoir fluid level when operating, and the reservoir capacity shall be such that, in normal conditions, it can accept all the returning fluid without spillage or over pressurisation.

7.3 Reservoirs shall provide adequate head of fluid for the hydraulic pump, or be pressurised to meet the hydraulic pump requirements at system start up and at all operating conditions and altitudes. Pressurised reservoirs shall have a maximum pressure control device to prevent over pressurisation. Bootstrap type reservoirs shall have the equivalent low pressure relief setting slightly higher than the HP system relief pressure to avoid transient functioning of the reservoir relief valve. Non pressurised reservoirs shall also have a maximum pressure control, and a minimum pressure control shall be considered to prevent collapse of the reservoir or system cavitation.

7.4 Reservoirs shall be either self bleeding of air during filling and changes in fluid level, or a manually operated bleed valve shall be fitted. This valve is important when manual filling is required as provision must be made for the reservoir to depressurise safely before removal of the filler cap. The return flow to the reservoir must be so arranged that the pump suction does not directly receive the returning fluid and any fluid aeration is reduced to a minimum.

8 SYSTEM OPERATING REQUIREMENTS

8.1 PRESSURE AND FLOW CHARACTERISTICS

8.1.1 Consideration shall be given to pressure losses in hydraulic lines and units due to flow. The hydraulic pipe sizes selected shall take into account hydraulic losses due to pressure and flow rates and the shape of the pipe.

8.1.2 The actual working pressures available at hydraulic units shall be used for operational performance calculations. See para 15 for strength requirements.

8.1.3 Account shall be taken during design, of any intensification of pressure that could occur in the system under operational conditions of loading (e.g., that caused under certain circumstances by the difference in area between the two sides of an actuator piston or where the load assists the actuator pressure).

8.1.4 Transient peak pressures that may occur during operation of any service in the system shall not exceed the nominal system pressure (1.0 Pw) by more than 20 per cent (1.2 Pw max). In special cases where this cannot be achieved, due allowance shall be made for the surge pressures in the strength of the system (see para 15).

8.1.5 Return line back pressures which can occur in the system from reservoir pressurisation, or operational conditions shall be considered in the design of hydraulic units and transmission lines. In particular the effect which this may have on performance of hydraulic valves, actuators, wheel brakes and mechanical locks that are hydraulically actuated.

8.1.6 Pressure relief provision shall be made for preventing the steady pressure in any part of the system from exceeding, from any cause including thermal expansion of the fluid and excessive external loads, the pressure Pr, where Pr is the relief pressure for that part of the system fixed at a value which allows a reasonable margin above Pw (usually Pr = 1.33 Pw approximately but in some systems a higher relief pressure may be necessary). The pressure relief provision shall also protect the system during ground testing with external hydraulic power applied.

8.1.7 Consideration shall be given to the effect of local heating of fluid due to high pressure leakage and to possible electrokinetic damage.

8.2 TEMPERATURE

8.2.1 The hydraulic system shall be designed so that the hydraulic fluid and sealing materials in any part of the system do not exceed their temperature limitations under the most severe conditions of normal flight or on the ground.

Local heating due to any cause, including increased flow through worn components, shall not cause excessive friction or seizure.

8.2.2 The cumulative damage which may be caused to elastomeric sealing materials or flexible pipes, that may have to operate at their extremes of temperature for considerable periods, shall be considered when selecting materials.

8.2.3 When electrical or electronic controls are used integrally with hydraulic components and units in the system the combined effect of hydraulic heating with fluid temperatures shall be considered (e.g., wet solenoids) and also any fire hazard.

8.3 BLEEDING

8.3.1 The design aim shall be that systems are self bleeding, but where this is not possible, bleed screws in components shall be provided for the removal of trapped air, and if necessary in the piping system. Disconnection or slackening of the pipelines for bleeding is prohibited unless agreed by the Rotorcraft Project Director.

8.4 INDICATION AND INSTRUMENTATION (see also Chapter 107).

8.4.1 Pressure indication shall be provided in the cockpit to show the pressure available from the hydraulic system and to give indication of the failure of any system. On rotorcraft with more than one system the indication shall show the pressure available in each system.

8.4.2 When two pumps feed into one system consideration shall be given to the need for providing flow and/or pressure indication for each pump.

8.4.3 Cockpit or pressure cabin indication shall be provided by the use of remote indication type gauges to avoid locating fluid or gas transmission lines in these areas (see para 13.5). Alternatively indication may be by electronic means when the Rotorcraft Specification includes the use of data bus electronic transmission systems.

8.4.4 Pressure indication for accumulator gas charge pressure shall be installed so that it can be easily seen when servicing the rotorcraft.

8.4.5 Where mechanical pressure gauges are used in other rotorcraft locations, with the agreement of the Rotorcraft Project Director, they shall be installed so that failure of the gauge cannot cause serious loss of hydraulic fluid or gas.

8.4.6 Additional instrumentation to satisfy particular requirements for monitoring or indication of the condition of the hydraulic system (e.g., reservoir fluid level, low pressure warnings or BIT) will be stated in the Rotorcraft Specification and agreed with the Rotorcraft Project Director.

8.5 REPLENISHMENT

8.5.1 Replenishment points of hydraulic reservoirs shall be readily accessible and there shall be a simple means of determining the fluid level from the replenishment point without the use of external equipment. Fluid used for replenishment shall be filtered to the required standard for the system, and pass through a rotorcraft system, filter.

8.5.2 When gravity filling is required the internal diameter of the filling orifice shall not be less than 38mm (Ref 1) and the reservoir replenishment inlets shall contain strainers to prevent the ingress of foreign bodies when the filler cap is removed.

8.5.3 Means shall be provided for draining away clear of the rotorcraft any fluid spilled at the filling points.

8.6 GROUND TESTING

8.6.1 External Test connections from which all services can be operated by external ground servicing equipment shall be provided. These shall be of the self sealing type. Test connections must give easy access to, and be of adequate strength to accept the load from the ground equipment hoses. On small simple rotorcraft and when agreed with the Rotorcraft Project Director a hand pump which is part of the rotorcraft system can be used for ground testing.

8.6.2 Where two or more pumps feed on a system, means shall be provided for checking all pumps on the ground before take off.

8.6.3 Where more than one hydraulic system is connected to a hydraulic sub circuit it shall be possible to check the functioning of the hydraulic circuit using each hydraulic system separately. Care shall be taken to avoid possible damage to, or cavitation in, any unit due to single system working e.g., hydraulic motors with a common drive shaft where one may act as a pump in an unpressurised condition, or cavitation in the unpressurised portion of a tandem powered flight control.

8.7 INTERNAL PROTECTION

8.7.1 Internal corrosion and fretting shall be considered when designing the hydraulic system and its components and protection provided where possible.

8.7.2 Some hydraulic fluids are hygroscopic and additional water may be induced due to condensation in reservoirs. Also, other contaminants are possible, therefore, the ability to drain is necessary and suitable drain points shall be provided.

8.8 FUNCTIONING

8.8.1 The system shall comply with the requirements of Chapter 100, paras 8 and 9.

9 FILTRATION

9.1 FILTRATION STANDARDS

9.1.1 Filters shall be provided to filter all the circulating fluid in the system and shall be such as to maintain contamination levels of the system below the level of the class (as specified in DEF STAN 05-42) agreed between the Designer and the Rotorcraft Project Director. Sub circuits may require a different filtration standard to that required for the main system. When ground servicing rigs are connected to the rotorcraft hydraulic system(s) the filtration standard of the rotorcraft system must not be degraded by circulation of fluid through the ground rig.

9.2 FILTER TYPES AND LOCATIONS

9.2.1 General. Filters of the type specified shall be provided in the locations in the following paras as a minimum requirement. Additional filter locations shall be provided if required by the Rotorcraft Project Director, or if required to protect critical components such as powered flight controls.

9.2.2 Pump Circuit

(i) Pressure Line. A non bypass type filter shall be installed in each system pressure line, and so located that all fluid from the pump(s) and ground rigs will be filtered prior to reaching system components which are sensitive to contamination. In multi pump systems each pump shall have a separate filter installed with no system components between the pump and the filter.

(ii) Pump Case Drain Circuit (PCD) Pump case drain flows shall be filtered using either a separate pump case drain (PCD) filter or combined LP/PCD filter. Separate PCD filters shall normally be of the bypass type, but may be of the non bypass type with differential indicator if the proof pressure of the pump case and the PCD filter is high enough to withstand the pressure arising from a blocked PCD filter up to full system pressure.

(iii) Pump Suction Circuit. Suction filters (filters between the system reservoir and pump inlet) shall not be used unless specifically agreed with the Rotorcraft Project Director.

9.2.3 Hydraulic Sub Circuits. Screen type filter elements to protect against contaminate particles in both flow directions shall be fitted adjacent to the orifices in components or hydraulic lines where these are 1.75mm or less in diameter, or the equivalent cross section area. The total area of the screen must ensure that it does not itself cause a blockage. Other hydraulic units (e.g., sequence valves) may also require filter screens in both flow directions if the unit controls a critical function. Powered flying control valve blocks should also be fitted with a suitable filter or filter screen.

9.2.4 Return Line Circuits. A bypass type filter shall be installed in each system low pressure (LP) (return) line downstream of all system components. Consideration shall be given to providing a filter with a rated flow well above the return line flow under maximum operating conditions to minimise bypass operation. All fluid in the low pressure circuits (including reservoir replenishment fluid) shall pass through the low pressure filter prior to entering the system reservoir. Special consideration must be given to prevent unacceptable back pressures generated by return line filters in return lines from wheel-brakes. The solution to be agreed with the Rotorcraft Project Director.

9.3 FILTER ELEMENTS

9.3.1 All filter elements other than the screen type called for in para 9 shall be of the disposable 'depth' type. The pressure line filter elements including screen type elements shall have a burst strength above the nominal system working pressure for both normal and reverse flow directions (see para 15). Filters shall be positioned to give easy access for element change in minimum time, with suitable provision for the removal and replacement of the filter elements, and to prevent fluid spillage and limit the ingress of air.

9.4 DIFFERENTIAL PRESSURE AND INDICATORS

9.4.1 All non by pass type filters shall be equipped with differential pressure indicators, and it is preferred that these indicators shall be fitted to all filters. The differential pressure indicator shall signal when an element change is required and shall be clearly visible to the ground crew during routine inspections. Provision shall be made to prevent spurious indications occurring due to extreme conditions (e.g., external contamination). Where indicators are not fitted filter elements must be changed on a periodic rather than an 'on condition' basis.

9.5 FLUID SAMPLING POINTS

9.5.1 The requirement for fluid sampling points will be stated in the rotorcraft specification. DEF STAN 05-43 gives guidance on fluid sampling and the type of sampling valve shall be agreed with the Rotorcraft Project Director.

10 HYDRAULIC PIPELINES AND COUPLINGS

10.1 RIGID PIPES

10.1.1 The hydraulic tubing shall be in accordance with DEF STAN 47-25 unless otherwise agreed with the Rotorcraft Project Director.

10.2 INSTALLATION OF RIGID PIPES

10.2.1 The aim shall be to ensure that the installation of hydraulic rigid pipes, together with their components and supports, will be such that they are capable of withstanding throughout the life of the rotorcraft the worst effects of vibration, structural distortion, malhandling and temperatures likely to occur. All conditions up to the limits of the design flight envelope of the rotorcraft shall be taken into account. In addition the rigid pipes shall be so installed that they are unlikely to be used as hand or foot holds.

10.2.2 The material of the pipe supports, clips and fairleads in contact with the pipe, shall be such that it does not cause damage to the pipe through chaffing when subjected to vibration, expansion and contraction. See Chapter 407 for protection against corrosion. The need for electrical bonding shall be considered.

10.2.3 Provision shall also be made in the construction of the supports or in the layout of the pipes to permit changes in pipe length induced by expansion or contraction resulting from changes in temperature, flight or ground loads, actuator movement or hydraulic pressure. All pipes shall be adequately supported. Outrigged clipping with distance tubing and spacing bobbins shall not be used.

10.2.4 The following minimum clearances shall be maintained:

(i) between pipe lines and between pipelines and adjacent rigid structures or fixed components - 6.0mm.

(ii) between pipelines and structure where pipes are held by a fairlead on to the structure or a support bracket - 3.0mm. (This dimension can only be permitted if the structural part does not extend more than 25.0mm beyond the fairlead).

(iii) between pipelines and moving rigid parts, e.g., control rods, etc., - 13.0mm under the most adverse conditions, and

(iv) between pipelines and control cables or other flexible moving parts - 25.0mm under the most adverse conditions.

Any cases of difficulty in complying with this requirement shall be referred to the Rotorcraft Project Director for consideration at an early stage in the design.

10.2.5 The clearances of para 10.2.4 shall not be achieved by the insertion of packing materials which will be subject to deformation or deterioration which will ultimately appreciably reduce the clearance.

10.2.6 Where components and units, such as non return valves, restrictors etc., are not provided with integral mountings and are coupled directly into pipelines, provision shall be made for the pipes to be supported as close as practicable on each side of the component.

10.2.7 Adequate spanner access to all hydraulic pipe unions, connections and couplings shall be provided. Torque loading shall be specified where necessary. It shall be possible to change a pipe, component or unit with the minimum of disturbance, and if necessary adjustment shall be provided to achieve this requirement.

10.2.8 Pipe identification shall be in accordance with Chapter 806, para 5.

10.3 PIPE COUPLINGS

10.3.1 All pipe couplings shall be in accordance with DEF STAN 47-25 unless otherwise agreed with the Rotorcraft Project Director.

10.3.2 The pressure line and return line pipe coupling sizes at each junction shall be different and so arranged that there is no possibility of incorrect assembly. Direction of flow where this is important, e.g., non return valves, shall be clearly marked on the unit, but in addition different size and connections is preferred, or male and female configuration.

10.4 FLEXIBLE PIPES/HOSES

10.4.1 All flexible hoses shall conform to DEF STAN 47-12. Unless otherwise agreed (see para 10.3.1), the end fittings shall be in accordance with DEF STAN 47-22. The hose assemblies shall not be subjected to torsional deflections (twisting) when installed, or during functioning and operation of the system. The support of a flexible hose shall not cause deflection or relative motion with its associated rigid pipe. The minimum hose bend ratio is a function of hose size and type; it will be stated in the applicable hose specification.

10.4.2 Hoses shall be installed so that chaffing with adjacent parts does not occur and where necessary they shall be suitably protected. Any hose elongation and contraction under pressure shall not cause straining of the end fittings, or binding and chaffing of the hose.

10.4.3 Coiled rigid tubing may be used if required to connect the hydraulic lines of moving components or units to the airframe connections, providing the movement is relatively small, and adequate qualification testing has been completed to prove the design to a standard acceptable to the Rotorcraft Project Director.

10.4.4 Other forms of flexible connections in hydraulic circuits, such as swivel couplings, sliding sleeves, ball joints etc., may be used if required, providing approval testing has been completed to a standard acceptable to the Rotorcraft Project Director.

10.4.5 The installation of all types of flexible pipes and hoses shall be such that cross coupling between pipes or hoses cannot occur.

11 BONDING

11.1 The hydraulic system units and components, also the pipelines, shall be electrically bonded to the airframe in accordance with the requirements of Chapter 708.

12 ELECTRICAL AND ELECTRONIC UNITS AND COMPONENTS

12.1 Where electrical or electronic controls, switches, transducers, potentiometers, differential transformers, solenoids, or motors etc., are built in or form part of a hydraulic unit, their requirements will be included in the specification for the hydraulic unit. The hydraulic unit shall be tested and approved complete with the electrical or electronic components.

12.2 The electrical or electronic components must be compatible with the associated systems. They should be items approved as individual components. Where this is not possible they may be qualified and approved with the hydraulic unit, providing the testing satisfied both the hydraulic and electrical requirements. (see also Chapter 706 and 708).

12.3 EMC requirements, when applicable, shall be satisfied in accordance with the requirements of DEF STAN 59-41 and Chapter 1011

13 FAILURE MODE AND EFFECT ANALYSIS

13.1 On completion of the initial design of the hydraulic system a failure mode and effect analysis shall be prepared. This will identify the degree of hazard or emergency which can occur as a consequence of the design or failure of any component or detail part in the system. This shall be updated as changes to the system occur.

13.2 Primary hydraulic systems and any critical circuits served by utility systems must have an emergency standby system. The extent of the emergency system and the method of engagement/disengagement shall be agreed with the Rotorcraft Project Director. Malfunction or failure of any part of a hydraulic system shall not leave the rotorcraft in a hazardous flight condition. See Chapter 100, para 9.1.

13.3 Where more than one hydraulic system is installed in the rotorcraft, the transfer of fluid between systems shall be prevented, as shall any pressure or flow change due to the interaction of the systems at any hydraulic component. It shall not be possible for the failure or malfunction of one system to cause the failure or malfunction of the other system due to interaction at any point including the reservoir and return lines.

13.4 Where two or more sub circuits are pressurised by a common pressure source, one of which is essential to flight operation and the other is not essential, suitable priority circuit isolation shall be included to prevent a failure in the non essential circuit affecting the essential circuit.

13.5 The location of high pressure fluid components and transmission lines in the cockpit or pressure cabins shall be avoided or contained in sealed conduits drained out of the inhabited area.

13.6 REDUCTION IN VULNERABILITY TO BATTLE DAMAGE (Chapter 112)

13.6.1 Reduction of vulnerability to battle damage shall be in accordance with the aims and requirements of Chapter 112. (See para 15.2 for vulnerability test on gas/oil accumulators).

13.6.2 Pipes shall be designed to function despite structural distortion caused by the effects of Defined or Specified threats in Chapter 112.

13.7 FIRE PRECAUTIONS

13.7.1 Fire precautions shall be in accordance with Chapter 712 to minimise the inherent fire risk in hydraulic systems.

14 HYDRAULIC SYSTEM FUNCTIONING AND TESTING

14.1 TEST PLAN

14.1.1 A test plan shall give full details of all testing proposed including information on the components, units or systems to be used to demonstrate compliance with the requirements. This shall be agreed with the Rotorcraft Project Director in conjunction with Airworthiness Division RAE Farnborough.

14.2 FUNCTIONING TEST RIG

14.2.1 The satisfactory operation of the complete hydraulic system(s) shall be demonstrated on a functional mockup of the system(s). This shall show that the system adequately satisfies the rotorcraft operation requirements, including the most critical cases, with all reasonably foreseeable combinations of loading and failure cases. Suitable instrumentation shall be included to measure critical parameters such as load, speed of operation, temperature, pressure surges, flow rates where critical, fluid contamination etc. (Leaflet 704/1).

15 STRENGTH, FATIGUE, ENVIRONMENTAL, VIBRATION AND ENDURANCE REQUIREMENTS

15.1 STRENGTH REQUIREMENTS

15.1.1 The strength of each part is defined in terms of pressure Pw, Pr and Pd defined in para 5. This applies equally to parts subjected to internal pressure, and those subjected to suction pressure which may cause the collapse of parts. The criteria defined for Pw, Pr and Pd also applies to those components exclusively used in low pressure hydraulic systems.

15.1.2 Components

(i) A hydraulic component is defined as any separate unit which is connected by the piping or hoses within the hydraulic system. Components include all classes of valves, hydraulic actuators, accumulators, manifold blocks, hydro-mechanical devices, filters etc. The strength requirements for reservoirs are detailed in para 15.1.4.

(ii) All hydraulic components shall be able to withstand:

(a) A design proof pressure not less than 1.5 Pw, 1.33 Pd or 1.125 Pr whichever is the greater, without permanent distortion or leakage.

(b) An ultimate pressure not less than 2.0 Pw, 1.75 Pd or 1.5 Pr whichever is the greater, without fracture or bursting.

Together with, in each case, the most adverse loads, similarly factored that can occur at the same time from the operation of any service or from flight, landing or take-off conditions.

(iii) When a hydraulic component forms part of a mechanism which has to withstand externally applied loads during flight, take-off or landings, the required proof and ultimate factors shall be realised in the current strength cases for the mechanism when the most adverse effects of pressure which can occur at the same time are included in the loading conditions.

15.1.3 Pipes and Pipe Couplings

(i) Rigid pipes and couplings shall be able to withstand:

(a) A design proof pressure not less than 1.5 Pw, 1.33 Pd or 1.125 Pr whichever is the greater, without leakage or permanent distortion greater than that permitted by the material specification at the specification pressure.

(b) An ultimate pressure not less than 3.0 Pw, 2.66 Pd or 2.25 Pr whichever is the greater, without fracture or bursting.

In the case of rigid pipes subjected to return line pressures only, the proof pressure may be 0.75 Pw (the Nominal system supply pressure) with an ultimate pressure of 1.5 Pw or a proof pressure of 1.5 times maximum return line pressure with an ultimate pressure of 3.0 times maximum return line pressure, whichever is the greater.

(ii) All flexible pipes shall be able to withstand:

(a) flexible hoses, the design proof and ultimate pressures defined in DEF STAN 47-12 for the appropriate Type.

(b) coiled rigid tubing and other forms of flexible connection as defined in paras 10.4.2 and 10.4.3, the proof and ultimate pressures in para 15.1.3.

15.1.4 Reservoirs

(i) The reservoir shall have proof and ultimate factors not less than 1.125 and 1.5 respectively on the most adverse loads than can occur at the same time or separately under all conditions of flight. This requirement shall be met at all fluid levels in the reservoir from empty to full.

(ii) Reservoirs shall have proof and ultimate factors not less than 1.5 and 2.0 respectively on the maximum working reservoir differential pressure, or 1.15 and 1.5 times the maximum reservoir relief pressure, whichever is the greater.

15.2 STRENGTH TESTS

15.2.1 General - Hydraulic Equipment. Using the units allocated for qualification testing in the test plan, all hydraulic equipment shall be tested to establish compliance with the requirements of para 15.1. The equipment is defined as follows:

(i) Components in para 15.1.2.

(ii) Pipes and pipe couplings in para 15.1.3.

(iii) Reservoirs in para 15.1.4.

The tests shall demonstrate that there is no permanent distortion or leakage at proof conditions. In the case of hydraulic components where the shape of the component is such that reliable calculations of ultimate strength can be made, tests to ultimate pressure need not be carried out. Gas/oil hydraulic accumulators shall also be tested to the requirements of Chapter 719, para 5.8.

15.2.2 Prototype System. The prototype system shall be tested to show that no leaks occur at a pressure of 1.33 Pw. For the purpose of this test valves in components shall be adjusted where necessary to obtain this pressure.

15.2.3 Production Tests

(i) General - Hydraulic Equipment. The drawings and/or production test procedures of all hydraulic components, units, pipe and coupling assemblies and reservoirs shall specify a test pressure equal to:

(a) 1.33 Pw, Pd or Pr, whichever is the greater for 3 minutes, for hydraulic components, units and pipe and pipe coupling assemblies. (see paras 5.1 and 7.1.6).

(b) 1.0 Pw or Pr whichever is the greater for 3 minutes, for hydraulic reservoirs. (see para 15.14).

(c) Pipe and pipe coupling assemblies which are subjected to return line pressures only, may be tested to 0.75 (Pw or Pr) whichever is the greater, or 1.33 times maximum return line pressure; pressures to be held for 3 minutes. (see paras 5.1 and 8.1.6).

and shall state that no leakage or permanent distortion is to occur when the test pressure is applied.

(ii) Complete Rotorcraft - Hydraulic Systems. The drawings and/or production test procedures for the complete hydraulic system shall specify a test pressure equal to 1.0 Pw (see para 5.1 and 6.1) for 3 minutes and shall state that no leakage is to occur when the pressure is applied.

15.3 FATIGUE REQUIREMENTS

15.3.1 The safe fatigue life of each hydraulic component, hydraulic unit, pipe and pipe coupling assemblies, flexible pipes, reservoirs, filters etc., shall be at least equal to the specified life of the rotorcraft. Any deviation shall be agreed with the Rotorcraft Project Director in conjunction with Airworthiness Division RAE. The requirements for fatigue life determination are given in Chapter 201, and advisory material is contained in Chapter 201 supporting leaflets and Leaflet 704/2. The fatigue life of all Grade 'A' parts (see Chapter 400) shall be demonstrated by a fatigue test to an agreed spectrum and appropriate safety factors, with exceptions as defined in para 15.4.2.

15.4 FATIGUE TESTS

15.4.1 General - Hydraulic Equipment. Using the units allocated for fatigue testing, it shall be demonstrated by a fatigue test to an agreed spectrum, suitably factored, that the specified fatigue life of each component is achieved. Leaflet 704/2 provides information on the fatigue testing of hydraulic components. The factors to be used are given in Chapter 201. The endurance test in para 15.6.1 may be used as part of the fatigue test providing all conditions are satisfied. The whole fatigue test must be completed on the same component, and the test factors in Chapter 201 achieved.

15.4.2 Existing Equipment and Standard Parts. Further fatigue tests may not be required on components such as hydraulic pumps, motors, existing equipment and standard parts, for which a separate type test and test evidence is available. The existing test reports shall be compared with the fatigue requirements for the particular project and presented for approval by the Rotorcraft Project Director in conjunction with Airworthiness Division RAE.

15.5 ENVIRONMENTAL AND VIBRATION REQUIREMENTS

15.5.1 General Requirements. Information on the general working ambient temperatures for the system is given in para 3. More general information will be contained in the Rotorcraft Specification and in BS3G100 for environmental conditions and BS3G100 Sub Section 3.1 for vibration. The appropriate requirements and tests will be included in the separate specification for hydraulic equipment. The complete hydraulic system, including lines and components shall be designed to withstand the effects of vibration, pump pulsation and shock loads encountered during the service operation of the rotorcraft.

15.5.2 Environmental Tests. New designs of hydraulic equipment shall be tested to establish compliance with the environmental conditions. For all cases if there is close similarity to other components already tested and in service, the evidence shall be considered and if appropriate submitted for agreement by the Rotorcraft Project Director in lieu of further testing.

15.5.3 Vibration Tests. Using the units allocated for qualification testing in the test plan, all hydraulic equipment (components and units etc.,) shall be tested to establish compliance with the vibration requirements. An exception may be acceptable if there is close similarity in all respects with components already tested. The test evidence shall be considered and if appropriate submitted for agreement by the Rotorcraft Project Director in lieu of further testing.

15.6 ENDURANCE REQUIREMENTS

15.6.1 General Requirements. When preparing the separate specifications for hydraulic equipment, consideration shall be given to the need to demonstrate by endurance testing that the maintenance requirements of DEF STAN 00-970 Part 8 will be achieved. The required duty cycle, loads and number of cycles to be achieved will be included in the equipment specification with any temperature or other conditions. A suitable factor on the actual duty cycle is required, and normally for a separate endurance test to demonstrate maintenance requirements only, a factor of 2.0 is recommended. This shall be agreed with the Rotorcraft Project Director. However, if it is proposed to use the Endurance Test as part of the Fatigue Test, the test factors given in Chapter 201 must be achieved on parts subject to fatigue life control. See para 15.4.1.

15.6.2 Endurance Tests. Endurance tests may be carried out on units allocated for qualification testing as a separate test, but if convenient, and all conditions are satisfied, the tests may be carried out as part of the tests of the functioning test rig required in para 14.

16 SYSTEM MAINTENANCE

16.1 MAINTENANCE (See Part 8)

16.2 HEALTH MONITORING

16.2.1 Health monitoring procedures shall be considered for the hydraulic circuits and sub circuits, to enable the Service to have early warning of possible failures. This may include built in test equipment or external ground services diagnostic equipment, as required by the Rotorcraft Specification.

16.2.2 Technical data shall be provided on the principal performance parameters of the hydraulic system and its individual units, for the as new and maximum worn conditions (e.g., pressure, flow, leakage etc., as appropriate). This information will assist the User Service in assessing the airworthiness of the system and its units.

16.3 Marking of hydraulic components shall be in accordance with DEF STAN 16-21.

17 CROSS REFERENCES

17.1 A number of requirements directly related to the hydraulic system appear elsewhere in this publication and these are listed in the Alphabetical Index. In addition certain general requirements apply when hydraulic actuation is chosen for special tasks and it is the designer's responsibility to ensure compliance with all such requirements. Some of the more important requirements in both these categories are listed in Table 1.

TABLE 1

LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	**Paragraph**	**Subject**
100	2	Use of standard equipment
	6.2	Component tests
	7	Prevention of incorrect assembly
	8	Conditions of operation
	9	Power operated systems
	9.1	Independence of services
	9.2	Provision of power for use in an emergency
101	1	Climatic conditions
107	14.2	Warning of main hydraulic power failure
	10	Powered flying controls - tests
306		Undercarriages - retraction and lowering
310	-	Wheel tyres, brakes and braking systems
407	-	Precautions against corrosion
605	-	Power operated flying controls - safety precautions
712	-	Fire Precautions
800	-	Maintenance
806	4 and 5	Marking of filling points and pipe lines
1004	-	Flight tests - hydraulic systems
Leaflet	**Paragraph**	**Subject**
206/5 (VOL 1)	-	Fatigue tests on hydraulic powered flying controls

17.2 ASSOCIATED INTERNATIONAL AEROSPACE REFERENCES

ISO 6771 Aerospace Construction - Fluid systems and components - Pressure and temperature classification.

17.3 ASSOCIATED USA AEROSPACE REFERENCES

MIL-H-5440G - Hydraulic Systems Aircraft Types I and II, Design and Installation Requirements for

MIL-H-8775D - Hydraulic System Components, Aircraft and Missiles, General Specification for

MIL-A-5503D - Actuators, Aeronautical Linear Utility, Hydraulic, General Specification for

MIL-A-5498C - Accumulators Aircraft Hydropneumatic Pressure

MIL-F-8815 - Filter and Filter Element, Fluid Pressure Hydraulic Line, 15 Micron Absolute and 5 Micron Absolute Type II System, General Specification for

REFERENCES

Reference	ASCC Air Standard	STANAG	British Standard
1	25/11	3212	C13

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HYDRAULIC SYSTEMS

REFERENCE PAGE

Scientific and Technical Memoranda

16/62	.. ..	Fatigue and hydraulic tubing

RAE Technical Notes

Mech Eng 365	.. ..	The manipulation, strength and fatigue properties of titanium tubing for aircraft hydraulic systems
Chem 1108	.. ..	The cubical coefficients of expansion of rubbers used in hydraulic and pneumatic systems
Chem 1118	.. ..	The compounding of natural and synthetic rubber to control the cubical coefficients of expansion
Chem 1132	.. ..	The swelling of natural and synthetic rubber in experimental hydraulic fluids

British Standards

C9	.. ..	Specification for coupling dimensions for high pressure air charging valves for aircraft
C13	.. ..	Sizes and gravity filling orifices on aircraft
M24	.. ..	Graphical symbols for aircraft hydraulic and pneumatic systems
M51 (ISO 6771)	.. ..	Temperature and pressure classifications
M52 (ISO 6772)	.. ..	Impulse testing of hydraulic hose, tubing and fittings
M53 (ISO 6773)	.. ..	Thermal shock testing of pipes and fittings for fluid systems
M55 (ISO 7257)	.. ..	Rotary flexure testing of hydraulic tubing joints and fittings
M60 (ISO 7169)	.. ..	General specification for separable tube fittings for fluid systems
3G100	.. ..	Specification for general requirements for equipment in use in aircraft
6275 Part 1	.. ..	Hydraulic fluid power filter elements: multipass method of evaluating filter performance

Defence Standards

00-40	..	..	Achievements of reliability and maintainability
00-41	..	..	MOD practices and procedures for reliability and maintainability
01-5	..	..	Fuels, lubricants and associated products
05-42	..	..	Particulate contamination classes for fluids in hydraulic systems
05-43	..	..	Standard procedures for taking samples of hydraulic fluids for evaluation of particle contamination
05-56	..	..	Graphical symbols for aircraft hydraulic and pneumatic systems
05-123	..	..	Technical procedures for the procurement of aircraft, weapons and electronic systems
16-6	..	..	Charging valve for aircraft liquid spring undercarriage units
16-21	..	..	Marking of aircraft hydraulic components
16-26	..	..	Charging valves for aircraft high pressure nitrogen and hydraulic fluid systems
17-5	..	..	Aircraft hydraulic system ground support equipment used for international cross-servicing purposes
47-12	..	..	PTFE hoses for medium and high pressure fluid systems
47-22	..	..	End fittings for flexible hose assemblies for aircraft - metric
47-25	..	..	Pipelines and pipe couplings for aircraft fluid systems - metric

LEAFLET 704/1

HYDRAULIC SYSTEMS

FUNCTIONAL TESTING

1 INTRODUCTION

1.1 The purpose of this Leaflet is to recommend a schedule of tests to be made on a functional mock-up of the hydraulic system in order to establish that the requirements of Chapter 704 are satisfied. These tests are to check the operation of the hydraulic system as a whole and do not include tests on individual services such as undercarriage raising and lowering, bomb door opening, operation of powered flying controls, etc.

1.2 Provision by the contractor of a functional mock-up is called for in DEF STAN 05-123 (Technical Procedures for the Procurement of Aircraft, Weapon and Electronic Systems) Chapter 230. Further details necessary for the required tests are given in this Leaflet.

2 FUNCTIONAL MOCK-UP

2.1 The mock-up should represent the complete Hydraulic System including all emergency systems both physically and functionally as far as is practical. Where possible simulation of environmental extremes should be considered. It should be a replica in all details such as lengths of pipe lines and general disposition and types of components. Each service should be loaded by some means which simulated as far as possible the flight loading conditions. Pressure pickups required by Chapter 704, should be fitted at all points where surge pressures are likely to develop and temperatures should be measured at critical points in the system.

2.2 If an unpressurised reservoir, or a pressurised reservoir in which the fluid level can be affected by attitude, is used in the system means should be provided for tilting the hydraulic reservoir and suction pipe about an appropriate axis so that the fluid level at any rotorcraft attitude or acceleration can be simulated. Hydraulic pumps should be driven by a power source which can be regulated so that all engine power conditions from idling to maximum power can be simulated.

3 TEST SCHEDULE

3.1 HYDRAULIC POWER SUPPLY

3.1.1 With the power supply operating at maximum engine conditions and with no services operating check that the steady pressure does not exceed the maximum working pressure requirements defined in Chapter 704, para 6.1 in any part of the system.

3.1.2 With the power supply operating at the appropriate conditions select simultaneously all services that could be needed in any flight condition and determine the most adverse combination of services likely to be used in operational flying. Check that the times of operation with this combination of services satisfy the operating requirement for each service.

3.1.3 Simulate failure of the mechanism which off-loads the pumps and show that with pumps working at full capacity with no services operating, the pressure in the system does not exceed the relief pressure, P_r.

3.2 SURGE PRESSURE

3.2.1 Explore the complete system for surge pressures and examine the interaction of one circuit with another. Check that under all likely conditions of operation the requirement of Chapter 704, para 8.1 is satisfied. Any surge pressure which cannot be eliminated by design modification should be recorded so that account can be taken of it in the strength calculations in accordance with the requirements of Chapter 704.

3.3 TEMPERATURES

3.3.1 In the tests of para 3.1 check the functioning of each service or combination of services as far as practicable throughout its appropriate ambient temperature range as required by Chapter 101. Check that the temperature limitations of fluid and sealing materials are not exceeded under the most severe operational conditions of temperature and horsepower output.

3.4 INDEPENDENCE OF SERVICES (see Chapter 100, para 9)

3.4.1 Simulate singly, various likely failures, including leakage of fluid in each service in turn and each time check the operation of the corresponding emergency system. Each time check that all other vital services, as required by Chapter 100, para 9, can be operated and satisfy their respective operating requirements.

3.5 RESERVOIR CAPACITY

3.5.1 Select the combination of services corresponding to the fluid at minimum level and if applicable with the reservoir positioned to simulate the most adverse attitude and acceleration appropriate to that combination of services, check that the fluid level satisfied the requirement of Chapter 704, para 7. When simulating forward acceleration with any reservoir in which the fluid level can be affected by attitude the reservoir should be tilted so that the fluid level is equal to that which would be produced by the acceleration. The effect of accelerated take-off, when applicable, should be taken into account.

3.5.2 With the power supply operating at the most adverse conditions and with fluid topped up to the correct level operate those services or possible combinations of services which produce the maximum inflow to the reservoir and check that the reservoir relief valve or vent will operate satisfactorily, if required, and that the pressure in the reservoir does not exceed the design pressure. These tests should include the operation of emergency systems after a primary system failure.

3.6 NEGATIVE G CASE

3.6.1 Invert the reservoir if applicable and check that detrimental aeration of the fluid is not caused when the pump is running at all speeds from idling to maximum conditions. In systems in which powered flying controls are incorporated, no aeration should occur and full control should be maintained under negative g conditions for a period longer than the endurance of the engines for the particular installation, such that, after the engines have stopped, normal attitude can be regained from the most adverse g conditions, and engines restarted. When no special provision for negative g is made in the engine installation a period of 10 seconds should be assumed or a lesser period as agreed with the Rotorcraft Project Director.

LEAFLET 704/2

HYDRAULIC SYSTEMS

FATIGUE TESTING OF HYDRAULIC COMPONENTS

1 INTRODUCTION

1.1 This Leaflet contains information on the fatigue testing of hydraulic components and units required in Chapter 704, para 15.

1.2 The general requirements for fatigue substantiation are defined in Chapter 201 and associated leaflets. These requirements have been interpreted in this Leaflet for the particular application to hydraulic system components and units. The fatigue information in this Leaflet relates only to utility hydraulic system components and units in the pump and normal actuating circuits for the landing gear, bomb doors etc. , which are selected intermittently.

1.3 Fatigue testing of hydraulic powered flying controls is dealt with in Volume 1, Leaflet 206/5. In hydraulic circuits which are dedicated to powered flying controls and active control technology, the hydraulic circuit pressure pulsations will be determined from the test rig measurements defined in Volume 1, Leaflet 206/5, para 3. This spectrum with the factors in Chapter 201 will be used for testing units. For active control technology applications see Volume 1, Chapter 208. Where components and units are common to a number of primary and utility circuits, they should be qualified to the most severe spectrum in the interests of standardisation.

1.4 For a wheeled steering system and units refer to Chapter 303, and for wheel brakes and braking refer to Chapter 310.

1.5 For actuator mounting attachments etc., which are also loaded by system pressure pulsations in the hydraulic actuator, it is important that this fatigue damage is considered in the design of the airframe attachments.

1.6 Hydraulic components and units will be divided into 2 grades as specified in Chapter 400, para 2. These requirements apply to Grade 'A' parts only.

2 LOAD/PRESSURE SPECTRUM

2.1 The rotorcraft constructor should determine the spectrum appropriate to each hydraulic component and unit in the system, and include this, with the required fatigue safety factors, in the detail specification for the item. For safety critical Grade 'A' components and units, and in cases of doubt the spectrum should be agreed to the satisfaction of the Rotorcraft Project Director in conjunction with Airworthiness Division RAE.

2.2 Consideration should be given to the loading from external sources including any effect of transient peak loads arising from kinetic energy and the arresting of movement by hydraulic or mechanical means (e.g., hydraulic actuators in some installations). Internal system pressure pulsations, including transient peak pressure, should also be included and considered in the fatigue spectrum, together with any thermal, vibration or other environmental conditions which may affect the fatigue life. In the case of hydraulic actuators it may be necessary to allow for the rotational friction of the attachment pins or bearings.

2.3 It is most important that the fatigue spectrum for each hydraulic unit and component is determined at an early stage of design and must be available in advance of any production series fatigue testing.

3 LIFE DETERMINATION

3.1 The safe life will normally be demonstrated by fatigue tests, and as defined in Chapter 201. Fatigue test factors for safe life compliance are given in Chapter 201.

3.2 For safety critical hydraulic components or units, and those which contain major structural features, such as mounting lugs, an estimate of safe life by calculation should be made at the early design phase, based on the principles in Chapter 201 until a fatigue test is completed to confirm the design.

4 FATIGUE REQUIREMENTS FOR PARTICULAR CIRCUITS

4.1 The location of the component or unit in the system determines the spectrum for that component. An individual circuit or sub circuit is defined as the circuit containing all components and units which are operated by a particular selection. The location of non return valves in the system usually determines the pressure independence of parts of the system. The effects of all demands and the interaction between circuits should be considered.

5 TEST REQUIREMENTS

5.1 Chapter 201 and its supporting leaflets give information on the derivation of test spectra and the safe life test factors to be achieved depending on the number of samples tested.

5.2 Where two or more samples are tested the geometric mean of the results should achieve the required factored life.

5.3 In all cases if there is close similarity to other components of known fatigue life a fatigue test may not be necessary, see Chapter 704 para 15.4.2. Evidence of life determination by analogy should be agreed with the Rotorcraft Project Director in conjunction with Airworthiness Division RAE.

5.4 Where different parts of the same component (e.g., an actuator) normally operate under different spectra, each part should be subjected to the spectrum appropriate to that particular part.

5.5 To use test spectra corresponding as nearly as possible to service conditions, flight by flight loading and internal pressures should be used whenever this is possible within a realistic timescale. This is particularly important for safety critical Grade 'A' components and units.

5.6 The test cycling may be repeated as rapidly as convenient providing the external loading is achieved on the test specimen, and a close approximation to a square wave pressure curve is obtained for internal pressure cycles. Further, the rate of test cycling must allow the pressure or load (and hence the stresses in components) to reduce to the required minimum level prior to the next test cycle.

CHAPTER 705

TRANSMISSION SYSTEMS

1 INTRODUCTION

1.1 SCOPE

1.1.1 This chapter sets out the requirements for design and testing to be met by conventional mechanical transmissions. For other types of transmission, or where elements of a novel character are embodied, the contractor shall discuss and agree his design proposals with the Rotorcraft Project Director.

1.2 DEFINITION

1.2.1 The transmission systems shall be considered to include all parts* necessary to transmit power between the engine output coupling and the rotor hubs, and drives for essential accessories. Mechanical transmission systems shall be considered to include rotor drive shafts, gearboxes, inter-connecting shafts and their support bearings, couplings, clutches, freewheels, rotor brakes, lubrication and cooling systems, accessory pads and drives, monitoring sensors, components of rotor control mechanisms that are integral with gearboxes, and attachment interfaces. (* See Leaflet 705/1, Para 2.6).

1.3 DESIGN RESPONSIBILITY

1.3.1 The rotorcraft designer shall be responsible for the design of the transmission systems and their installation, and for liaising with the manufacturers of transmission components and associated equipment to ensure compatibility. (See also DEF STAN 05-123/1, Chapter 102 Para 2.3 and Chapter 104, Para 3.4).

1.4 FUNCTIONING CONDITIONS

1.4.1 Unless otherwise specified the system shall be suitable for and function over the range of operating and environmental conditions for which the rotorcraft is designed (See also Chapter 101), including emergency conditions (See Chapter 100, Para 9).

2 INSTALLATION INTERFACES

2.1 AVOIDANCE OF HAZARDOUS EFFECTS

2.1.1 Containment of High Energy Rotating Parts. The design of transmission systems and their installations shall satisfy the requirements of Chapter 100, Para 25.

2.1.2 Prevention of Vibration Damage to Rotorcraft. Precautions shall be taken in the installation to ensure that there are no significant adverse effects resulting from vibration generated by the transmission systems within the operational envelope. If flexible mountings are used to isolate such vibration, the maximum deflections of such mountings shall be taken into account in the design of the relevant transmission system components. Additional requirements relating to the rotorcraft vibration environment for airframe, equipment, and aircrew are given in Chapter 501 Para 2.

2.1.3 Space Envelope. The space envelope provided for the transmission systems shall be such as to cause no hazard to exposed shafts, oil lines, sensors, or other vulnerable parts. Full account shall be taken of possible deflections or deformations included by operations of the rotorcraft within normal limits, and of normal maintenance actions.

2.1.4 Flexibility. Where relative movement between components within the transmission systems and between such components and the rotorcraft can occur, there shall be adequate provision for flexibility. Structural distortion of the airframe or gearbox casings shall not hazard power transmission, relative phasing of the rotors or fluid lines under pressure. (See also Chapter 200 para 1.7)

2.1.5 Security of Components. Attention shall be given to means of ensuring the security of hinged and removable access panels, cowlings, and any components which might otherwise hazard the transmission systems through contact with rotating shafts or the rotors.

2.1.6 Prevention of Access Damage. Vulnerable parts of the transmission systems including surface protection coatings shall be protected from the risk of maintenance induced damage by the appropriate provision and marking of guards, footholds and handholds. (See also Chapter 100 paras 14, 15, 800 para 6, 801 Leaflet 801/0, 802 para 2.2, and 806 para 10)

2.1.7 Shielding from Contaminants. Consideration shall be given to provisions for shielding vulnerable surfaces and components of the transmission systems from the worst effects of harmful contaminants such as sand, snow, salt water spray and contaminating liquids. (See also Chapter 407 para 3, and Chapter 101 para 7).

2.1.8 Engine Overspeed Protection. In cases where the safety analysis or substantiation tests indicate that transmission damage can progress from detection to failure within the duration of one flight, consideration shall be given to interfacing transmission fracture monitoring devices with a cockpit warning to allow a controlled shut down of the appropriate engine(s).

2.2 DESIGN FOR INTERCHANGEABILITY AND MAINTENANCE (See also DEF STAN 05-123/1 Chapter 201)

2.2.1 Attachment Points. The basic requirements for interchangeability and for attachment are given in Chapter 805. The effect of structural distortion shall be considered in respect of the attachment of transmission system change units.

2.2.2 Electrical Connections. Interchangeability requirements shall be considered when determining the locations of connectors for electrical actuators, and instrumentation.

2.2.3 Accessibility of Oil Sight Gauges. In addition to the general requirements of Chapter 800 para 6, particular attention shall be given to the accessibility of oil sight glasses where fitted, both for inspection and replacement. (See also Chapter 802 para 2)

2.3 REDUCTION OF GEARBOX NOISE TRANSMISSION

2.3.1 Attention shall be given to the attenuation of structurally and acoustically transmitted gearbox noise to the cockpit and cabin in relation to the requirements of Chapters 108 and 501.

3 SYSTEM DESIGN

3.1 SYSTEM ARRANGEMENT

3.1.1 Operation following Engine Failure. The transmission system shall be so arranged that in the event of engine failure, sufficient power will be supplied, either from any remaining engines or by rotor autorotation, to enable the satisfactory functioning of rotors and other units necessary for control of the rotorcraft and for such further operation after the failure as is called for in the Rotorcraft Specification.

3.1.2 Engine Starting. It shall be possible to start and idle each engine separately without driving the rotor(s).

3.1.3 Engine Disengagement. The drive from each engine shall incorporate a freewheel or similar device which can if necessary automatically disengage the engine from the rotor system to prevent it from driving the engine and which will permit the rotor(s) to continue to rotate freely when the engine is stopped.

3.1.4 Rotor Brake. Unless otherwise stated in the Rotorcraft Specification, a rotor brake or brakes shall be provided to stop rotor rotation without it being necessary to stop the engine(s). (See also para 3.2.10)

3.1.5 Avoidance of Rotor Interaction. The operating planes of the rotors shall not interact such that loss of transmission synchronisation could result in contact between rotors.

3.1.6 Phasing of Rotors. If a rotor dephasing device is incorporated (e.g. for stowage or transportation purposes) means shall be provided to ensure that the rotors are locked in proper phase before the engines are started. Provision shall also be made for remote indication to the pilot that the rotors are locked in phase. (Further requirements relating to folding components are given in Chapter 722).

3.1.7 Rotor-Structure Clearance. Adequate clearance shall be provided between any rotor and all other parts of the rotorcraft to prevent the blades striking any part of the rotorcraft during any permitted ground or flight manoeuvre. The clearance shall also be such that the possibility of the blades striking any part of the rotorcraft during any foreseeable wind condition or manoeuvre that might inadvertently occur shall be extremely remote. (For definition see Leaflet 705/1 para 2).

3.1.8 Rotor-Ground Clearance. It must be impossible for the tail rotor to contact the landing surface during a normal landing.

3.1.9 Accessories. No accessory, other than any essential for autorotative descent and landing, shall be driven from the rotor drive system unless it can be shown that the presence of the additional drive or any failure or malfunction of that drive or accessory will not constitute a hazard to the rotor system.

3.1.10 Incorporation of the other Functions. Where rotor control systems or other functions are incorporated within transmission system components they shall not reduce the integrity of power transmission or rotor synchronisation.

3.2 SYSTEM SAFETY

3.2.1 Design Criteria. As a design aim the probability of failure* of the Rotor and Transmission systems considered together that would prevent a controlled descent and landing shall be Very Remote*, unless otherwise agreed by the Rotorcraft Project Director.
(* these terms are defined in Leaflet 705/1, para 2.)

3.2.2 Safety Assessment. Compliance with the requirements of para 3.2.1 shall be shown by analysis, supported where necessary by ground, flight, or simulation tests as agreed by the Rotorcraft Project Director. The analysis shall include:

(i) Possible modes of failure
(ii) Possible multiple failures and undetected failures
(iii) The resulting effects on the rotorcraft and aircrew
(iv) Aircrew warning cues
(v) Events and errors

The extent, depth, and form of the analysis shall be agreed with the Rotorcraft Project Director.

3.2.3 Protection from Sudden Overload. The transmission systems shall be protected from failure due to excessive sudden torques produced by the engine(s), slippage of freewheels, or resulting from control movement (see Leaflet 705/1). Where a torque limiting device is used it shall be set at least 10 percent above the declared rotor maximum overload torque value, and shall be readily amenable to calibration.

3.2.4 Protection from Cumulative Overload. If any possible torque loading can result in overload or fatigue damage to any part in the transmission systems then automatic means of monitoring torque-time exposure and peak torque values shall be implemented in accordance with the requirements of Chapter 727 and used as a basis for assessing continued airworthiness together with additional inspections, the rules governing which are to be agreed with the Rotorcraft Project Director.

3.2.5 Application of Health and Usage Monitoring. Full consideration shall be given to the application of health and usage monitoring systems (see Chapters 201 and 726 - in preparation) to support the achievement of the design requirements of para 3.2.1 in respect of high integrity (vital) parts:

(i) the failure of which cannot be related to loads or duty cycles.

(ii) for which reliance upon damage tolerant characteristics is impractical.

(iii) for which a safe life has been declared, in order to confirm that the substantiation of the declared life remains valid.

Full consideration shall be given to the results of the safety assessment of para 3.2.2 in determining the health and usage monitoring functions to be provided.

3.2.6 Assessment of Safety Devices. Where reliance is placed on safety devices, instrumentation, early warning devices, maintenance checks etc., to limit the effects of a failure, the safety analysis shall cover the safety system failure in combination with the basic transmission systems failure.

3.2.7 Provision of Instruments. A statement shall be submitted to the Rotorcraft Project Director listing the cockpit indications necessary for satisfactory operation of the transmission systems. The overall limits of accuracy required of such indications for the purpose of enabling the flight crew to control satisfactorily the operation of the systems shall also be stated so that the suitability of the instruments as installed may be assessed. Due regard shall be given to possible inaccuracies in reading the instruments, and to the need to delineate normal and abnormal values.

NOTE: In determining transmission systems operating limitations, account will be taken of the declared overall limits of instrument accuracy.

3.2.8 Control of Wear through Component Overhaul. Whilst the design aim shall be to achieve the longest wear life practicable, the preferred method of achieving the airworthiness requirements for transmission systems is by removal 'on-condition' (defined in Leaflet 705/1, para 2). However in cases where overhaul inspections at fixed intervals are agreed with the Rotorcraft Project Director to be the means of controlling wear, corrosion, and age effects then the initial Time Between Overhaul (TBO) and subsequent increases shall be supported by evidence from detailed inspections of an agreed number of sample gearboxes at the existing and proposed TBO periods (see Leaflet 705/1, para 5).

3.2.9 Safety of Lubricated Parts.

(i) Each independent assembly shall where practical have its own oil supply suitably filtered; independent of the engine lubrication system(s), with a suitable level indicator or contents gauging means that shall not be rendered ineffective by obscuration or staining. (See Leaflet 705/1 para 6.3).

(ii) Transmission systems shall continue to function for a period of 30 minutes minimum following loss of oil*. Compliance shall be demonstrated by rig tests at loads and time factors to be agree with the Rotorcraft Project Director.
(* See Leaflet 705/1 para 2.7 for definitions)

3.2.10 Safe operation of the Rotor Brake (see also para 3.1.4)

(i) Means shall be provided to minimize both the possibility of a take-off with the rotor brake applied and the possibility of inadvertent application of the brake in flight. Means shall also be provided to warn the pilot if the rotor brake(s) have not been completely released prior to take off.

(ii) The rotor speed range above which the rotor brake(s) should not be applied shall be agreed with the Rotorcraft Project Director if not given in the Rotorcraft Specification. The rates of acceleration and deceleration resulting from brake operation shall not result in damage to components.

(iii) Limitations on the use of the rotor brake shall be specified and included in the Aircrew Manual.

3.2.11 Accident Data. Data required to establish the operating status or condition of the transmission systems for Accident Data Recording purposes shall be advised to the Rotorcraft Project Director in accordance with the requirements of Chapter 100, para 21. Every effort shall be made to implement knowledge gained from transmission related occurrences and accidents to both military and civil rotorcraft (see Leaflet 705/1, para 12).

3.2.12 Fire Precautions. Transmission systems shall comply with the fire precaution requirements of Chapter 712.

3.3 SYSTEM OPERATION LIMITS

3.3.1 Normal Service Use. The Transmission System operating limitations shall be determined having regard to the values demonstrated during the type tests, due allowance having been made for the overall limits of accuracy of the instrumentation declared for use in service, and shall conform to the following:-

(a) Rotor Speed

(i) Rotor Maximum RPM (power off) autorotation shall not exceed 95% of the lesser of the maximum rpm shown during the type tests or the maximum design rpm.

(ii) Rotor Minimum RPM (power off) shall be not less than 105% of the greater of the minimum rpm shown during the type tests or the minimum design rpm.

(iii) Maximum RPM (power on) shall be the time weighted mean value demonstrated during the ground endurance test (see Leaflet 705/2, para 8).

(iv) Rotor Minimum RPM (power on) shall be not less than the maximum shown during the ground endurance tests.

(v) Where Rotor Speed limitations for ground running are outside these values, they shall be declared and substantiated.

(b) Torque. Each declared transmission torque limitation shall be based on the mean values recorded during the appropriate tests.

(c) Oil Temperature and Pressure. The operating limitations shall be based on the time weighted mean values recorded during the appropriate periods of the endurance test or equivalent rig tests.

3.3.2 Handling Flight Tests. In determining the operating limitations of the transmission systems, consideration shall be given to the requirements relating to Handling Flight Tests (see Chapter 900, paras 7.2, 7.5 and 7.6).

4 DESIGN OF COMPONENTS

4.1 STRENGTH AND GENERAL REQUIREMENTS

4.1.1 Change Units. Components in the Transmission Systems Change Units not approved as part of the Transmission shall be designed in accordance with Chapter 200 and graded as defined in Chapter 400 para 2.

4.1.2 Proof and Ultimate Strength Factors. The whole transmission system shall have proof and ultimate strength factors not less than 1.125 and 1.5 respectively under all operational conditions. This requirement shall be met within the limitations of the rotorcraft over the full range of rotational speeds, including zero, for all torques up to the maximum torque which can be transmitted in either direction including the effects of the rotor brake(s), engine 'cough', surge and restarting of an engine in the air.

4.1.3 Effect of Vibratory Loads. Adequate allowances shall be made in the design of the transmission system(s) for the effects on stress levels of vibratory loads generated by the complete dynamic system(s). Where methods of predicting vibratory modes and loads have uncertain accuracy, the allowances shall be sufficient to minimize the likelihood of early test failure due to unexpected vibratory loads. Adequate allowances shall also be made for the occurrence of transient loads arising from changes of input or of working state, (e.g. freewheel slippage or actuation or abrupt changes of power). Unfavourable dynamic couplings shall be avoided where possible.

NOTE: The above requirements draw attention to the possible need for a reserve of static strength beyond that defined in para 4.1.2. Where dominant transient loads occur only once or twice per flight, an ultimate factor of 2.0 would be considered appropriate; where such loads occur more frequently or where cycles of vibratory loads of large amplitude can occur, a static ultimate factor seater than 2.0 may be required if the loads cannot be adequately represented in the fatigue spectrum.

4.1.4 Changes in Load Distribution. Any change in the distribution and magnitude of external or internal loads as a result of deformation of structure under load shall be taken into account.

4.1.5 Fatigue. All components shall have a fatigue life at least equal to that stated in the Rotorcraft Specification, unless otherwise agreed by the Rotorcraft Project Director. For each transmission system the load spectrum used for substantiation (i.e. the input torque and speed versus time intervals during which combination of the torque and speed are assumed to be applied) shall be agreed with the Rotorcraft Project Director in conjunction with the Airworthiness Division RAE. (See also Chapter 201)

4.1.6 Stress Concentrations. Particular care shall be taken in detailed design to avoid or reduce stress concentrations in critical areas that are subjected to vibratory loading.

4.1.7 Identification of Finite Life Components. Any component having a finite fatigue life shall be identified with a serial number. Precautions shall be taken to ensure that the identities of life items in assemblies are not lost during overhaul or rectification.

4.1.8 Control of High Integrity (Vital) Parts. The standards of design, documentation, manufacture and assembly of all non-redundant components subject to fatigue loading, the failure of which would critically affect safety shall comply with the requirements for high integrity (vital) parts given in Chapter 400 para 2.2. (See also Leaflet 705/1, para 2.4 and 6.2).

4.1.9 Control of Mass Distribution. The need for control of mass, moments, and dynamic balance within suitable limits shall be considered on all parts, especially high speed parts and long shafts, to permit direct replacement. The drawings shall specify these parts and indicate how adjustments are to be made.

4.1.10 Rolling Contact Fatigue.

(i) Design action shall be taken to reduce to a practical minimum the likelihood of surface initiation of rolling fatigue in service and in production acceptance tests. (See Leaflet 705/1, para 7.2.).

(ii) Effective and reliable means shall be provided for the detection and diagnosis of rolling contact fatigue, particularly in bearings and gears, well in advance of component failure. (See Leaflet 705/1, para 7.1 and 9.1).

4.1.11 Prevention of Incorrect Assembly of Parts. The requirements of Chapter 100, para 7.2 shall also apply to Transmission Systems. (See Leaflet 705/1, para 8).

4.2 GEARBOXES

4.2.1 Lubrication and Cooling. Each gearbox shall have self-contained means of lubrication and cooling unless otherwise agreed with the Rotorcraft Project Director. (See Leaflet 705/1, para 6.2)

4.2.2 Gearbox Configuration. In determining the configuration of gears, shafts, etc., consideration shall be given to factors affecting integrity. (See Leaflet 705/1, para 9).

4.2.3 Detection of Wear.

(i) A minimum of one effectively sited magnetic plug or ferrous wear debris monitoring probe capable of removal for inspection without necessitating oil drainage shall be provided for each gearbox. Attention shall be given to the integrity of the sealing valve arrangements, and to maximising debris catch efficiency. (See Leaflet 705/1, para 10.1)

(ii) In gearboxes having potential failure modes involving the wear of non-ferrous materials, effective means shall be provided for the in-situ inspection of such wear material independent of the main filter. (See Leaflet 705/1, para 10.1)

(iii) Provision shall be made for the convenient removal of oil specimens for analysis. The location shall permit extraction of a true average sample. (See Leaflet 705/1, para 10.2)

4.2.4 Detection of Overtemperature in Splash Lubricated Gearboxes. Gearboxes without pressure lubrication systems shall incorporate means to provide an indication to the pilot when oil temperature exceeds a safe value. (See Leaflet 705/1, para 10.4).

4.2.5 Detection of Fracture. Where the probability of fractures of vital parts cannot be demonstrated to be less than Very Remote* provisions for the detection of such fractures at the earliest possible time shall be implemented in accordance with Chapter 727. (* for definition see Leaflet 705/1, para 2) (See Leaflet 705/1, para 10.3).

4.2.6 Gearbox Noise Reduction at Source.

(i) Consideration shall be given to the limitation of gearbox noise at source to aid the achievement of acceptable noise levels in the cockpit and cabin.

(ii) Before implementing measures to reduce gearbox weight, consideration shall be given to the magnitude of possible net weight penalties to the rotorcraft due to additional sound-proofing material required in the cockpit and cabin as a consequence.

4.3 LUBRICATION SYSTEMS

4.3.1 Functioning.

(i) Lubrication, free from leakage, shall be provided to all components subjected to rolling and/or sliding contact, and shall be effective over the range of temperatures, attitudes, and manoeuvres for which the rotorcraft is designed.

(ii) Consideration shall be given to the provision of redundancy in oil supplies to critical areas of the transmission systems to enhance survivability in emergency conditions. (See Leaflet 705/1, para 6.1)

(iii) All lubrication systems shall be tested in accordance with Leaflet 705/2, para 4.3-4.5, and 7.2.

4.3.2 Lubricant. Oil to an approved specification shall be used as the lubricating and cooling medium unless otherwise agreed by the Rotorcraft Project Director, and shall be tested in accordance with Leaflet 705/2, paras 7.2 and 8.4.

4.3.3 Oil Filtration and Avoidance of Blockage.

(i) Oil filters, sized to provide adequate filtration quality and capacity, shall be installed so that all oil passing through the gearbox pumps shall pass through the filters. The filter elements shall be readily removable without disturbing any other part of the oil system, and the filter housings so designed that entrapped contaminant and unfiltered oil within the housings are removed when the filter elements are removed. A by-pass valve shall be provided for each filter to allow oil to pass in the event of clogging of the filter element, and shall be located in such a position to prevent scouring of previously trapped debris into the oil feed lines when the valve is open. (See Leaflet 705/1, para 7.2). An indicator to show impending and actual operation of the by-pass shall be provided at each stage in such a position that is readily accessible to and can be easily observed by servicing personnel.

(ii) Lubrication channels shall be designed to ensure that blockage cannot be induced by sealants or other contaminants.

(iii) Strainers necessary to protect oil jets or metering orifices shall be designed to reduce the possibility of blockage to a reasonable minimum.

4.3.4 System Fault Warnings. Each independent pressure lubrication system affecting a component the failure of which would be catastrophic shall incorporate the means to provide:

(i) adequate warning of failure of the pressure lubrication system (see Chapter 107, para 14.2), including independent oil pressure indicating and low pressure warning systems with connections close to the main jets.

(ii) indication to the pilot when the oil temperature exceeds a safe value. (See Leaflet 705/1, para 10.4)

Note: See also Leaflet 726/3 para 6.5 for recommendations concerning lubrication system monitoring.

4.3.5 Cooling Fans. Where reliance is placed on cooling fans for continued safe operation, they shall possess integrity of the same order as that required of the rotor and transmission systems, having regard to all possible failure modes.

4.4 SHAFTS AND SUPPORT BEARINGS

4.4.1 Critical Speeds*. There shall be no critical speeds for any section of shafting within 15 per cent of any steady state operating speed up to 1.05 times maximum engine overspeed or maximum permissible rotorspeed whichever is greater. Where shaft critical speeds are encountered during transient operation, contractor shall demonstrate by test that the resulting stress amplification is within safe limits and the requirements of para 4.1.3 are met. (* for definition see Leaflet 705/1, para 2).

4.4.2 Supercritical Shafting*. Where supercritical shafting is used in the transmission, the design shall be such that failure of any one intermediate support does not jeopardize the safety of the rotorcraft. (* for definition see Leaflet 705/1, para 2).

4.4.3 Support Bearings. Adequate provision shall be made for flexibility of alignment in bearings supporting shafts which are subject to deflection or change of alignment with operation. Provision shall also be made for effective lubrication of shaft support bearings and for means of indicating unsafe wear, or impending failure.

4.4.4 Spline Lubrication. Where splined couplings or drives are used, effective lubrication shall be provided. (See Leaflet 705/1, para 11).

4.4.5 Battle Damage Tolerance. The transmission shaft(s) shall be designed taking into consideration such reduction in strength and balance as may be caused by the effects of a Defined or Specified Threat. (See Chapter 112)

4.5 CLUTCHES AND FREEWHEELS

4.5.1 Engagement Loads. The engagement of clutches and freewheels shall be smooth and free from snatch and shall not result in damage to any component or part. If slippage of freewheels resulting in impact loading cannot reliably be prevented throughout the service life of the rotorcraft then means shall be provided for monitoring sudden overload and freewheel failures.

4.6 ROTOR BRAKES

4.6.1. Fire Precautions. The design shall minimise the risk of fire initiated by the rotor brake. See para 3.2.10 for Rotor Brake control requirements.

4.7 ACCESSORY DRIVES - WEAK LINK

4.7.1 The design of the Transmission Systems shall be such that each accessory drive has provision for a weak link to be fitted which will prevent a malfunctioning accessory from applying a dangerously high torque to the system. All other parts of the Transmission System shall be capable of withstanding safely the torque required to fail the weak link.

4.7.2 The torque at which each weak link is designed to fail shall be indicated on the relevant drawings.

4.7.3 The weak link shall be designed so that its failure will not result in flailing or in loose parts causing damage to the rotorcraft.

5 TESTING (See also DEF STAN 05-123/1)

5.1 PROTOTYPE AND RIG TESTS

5.1.1 To show that an acceptable standard of design and development has been achieved, a programme of tests shall be conducted on the lines laid down in Leaflet 705/2.

5.2 PRODUCTION TESTS

5.2.1 To show that an acceptable standard of production has been achieved, a programme of tests shall be conducted, on each unit, on the lines laid down in Leaflet 705/2.

5.3 PRODUCTION QUALITY TESTS

5.3.1 To check that the standard of manufacture is maintained, a production verification test shall be made periodically on units that have already passed their production tests.

5.4 TESTING OF MODIFICATIONS TO HIGH INTEGRITY (VITAL) PARTS

5.4.1 To establish the effect of any changes in the design or manufacturing processes which could affect the integrity or reliability of the transmission systems, evaluation of such changes shall be made in accordance with the requirements of Para 5.1 unless otherwise agreed by the Rotorcraft Project Director.

5.5 TESTING OF MONITORING PROVISIONS

5.5.1 All health and usage monitoring provisions which are required to enable the transmission systems to achieve the airworthiness requirements of this chapter or other requirements of the Rotorcraft Specification shall be fully applied and evaluated in all of the tests identified above, and preliminary caution and rejection criteria shall be established.

5.5.2 It shall be demonstrated by tests acceptable to the Rotorcraft Project Director that those fault detection devices, which are intended to give an early warning of an impending failure which could lead to catastrophe, give adequate and reliable warning of the fault before it has progressed to a dangerous extent.

6 DEMONSTRATION OF COMPLIANCE

6.1 Compliance with the requirements of this Chapter shall be demonstrated by either Design Records (DR), Mock up (MU), Analysis (A), Component Test (CT), Ground Vehicle Test (GVT), or Flight Test (FT), Production Quality Tests (PQT), or a combination of these means in accordance with Table 1, unless otherwise indicated in the Rotorcraft Specification, or agreed by the Rotorcraft Project Director.
(NB alternatives are indicated by /).

Note: Where test data is required it will normally relate to production standard components of the Project Rotorcraft. However some requirements may be satisfied by data obtained from tests with other components subject to the agreement of the Rotorcraft Project Director.

7 CROSS REFERENCE TO OTHER CHAPTERS

7.1 A number of requirements directly related to transmission systems appear elsewhere in this publication. The most important of these are listed in Table 2 below.

TABLE 1

COMPLIANCE DEMONSTRATION REQUIREMENTS (see para 6.1)

Chapter 705 Paragraph	Method of Compliance Demonstration (See para 6.1 for explanation of codes)
1.4	DR and A and CT/GTV and FT
2.1.1	(containment - DR and CT/GTV (position - DR/MU)
2.1.2	FT
2.1.3	DR/MU
2.1.4	DR/MU/GTV and FT
2.1.5	DR/MU
2.1.6	DR/MU
2.1.7	DR/MU
2.2.1	DR and FT
2.1.8	DR and A/CT
2.2.2	A
2.2.3	DR/MU
2.3.1	GTV/FT
3.1.1	FT
3.1.2	GTV/FT
3.1.3	DR
3.1.4	DR and GTV/FT
3.1.5	DR
3.1.6	DR and GTV/FT
3.1.7	DR and GTV/FT
3.1.8	DR and FT
3.1.9	DR
3.1.10	DR and GTV/FT
3.2.1	DR and A and CT
3.2.2	DR and A and CT/GTV/FT
3.2.3	DR and CT/GTV
3.2.4	A and CT/GTV and FT
3.2.5	DR
3.2.6	A
3.2.7	DR
3.2.8	DR and CT/GTV and FT
3.2.9	DR and CT
3.2.10	DR and GTV/FT
3.2.11	DR
3.2.12	DR
3.3.1	CT/GTV/FT
3.3.2	FT
4.1.2	DR and A and CT
4.1.3	DR and A and CT
4.1.4	DR and A and GTV/FT

TABLE 1 (continued)

COMPLIANCE DEMONSTRATION REQUIREMENTS (see para 6.1)

Chapter 705 Paragraph	Method of Compliance Demonstration (See para 6.1 for explanation of codes)
4.1.5	DR and A and CT
4.1.6	DR and A
4.1.7	DR and CT
4.1.8	DR
4.1.9	DR
4.1.10	(i) DR and CT/GTV and PQT, (ii) DR and CT
4.1.11	DR
4.2.1	DR and CT and GTV/FT
4.2.2	DR and A
4.2.3	DR and CT and GTV/FT
4.2.4	DR and CT and GTV/FT
4.2.5	DR and CT and A
4.2.6	DR and A
4.3.1	(i) DR and CT and FT (ii) DR and CT (ii) See Leaflet 705/2
4.3.2	DR and See Leaflet 705/2
4.3.3	DR and CT/GTV
4.3.4	DR and CT/GTV
4.3.5	DR and A and CT/GTV
4.4.1	CT/GTV
4.4.2	DR and CT/GTV
4.4.3	DR and CT/GTV and FT
4.4.4	DR and CT/GTV
4.4.5	DR and A and CT/GTV
4.5.1	DR and CT and GTV/FT
4.6.1	DR and CT and GTV/FT
4.7.1	DR and CT
4.7.2	DR
4.7.3	DR and CT

TABLE 2

LIST OF OTHER IMPORTANT REQUIREMENTS (see para 7.1)

Chapter	Paragraph	Subject
100	1, 2, 6.2, 7, 8, 9, 14, 15, 17, 25, 26	General Requirements
101	1, 2, 6, 7	Operation in various climatic conditions
107	14	Warning, Cautionary, and Advisory signals
108	1.2, 4	Internal noise
112	1.3, 3	Reduction of vulnerability to battle damage
200	-	Static strength and deformation
201	-	Fatigue
307	4	Post crash fire hazard
400	-	General detail design
401	-	Design data for metallic materials
402	-	Processes and working of materials
403	-	Castings
404	-	Marking of rotorcraft parts
406	-	Stress corrosion cracking
407	-	Precautions against corrosion
409	2	Rubbers - grade A applications
501	-	Vibration and internal noise
600	3.4.1, 3.5.1	Normal operating and Failure States
605	3.2.3	Failure states (also Leaflet 605/1 para 3.2.3)
700	2.2, 2.6.1, 4.7	Propulsion system installations

TABLE 2 (continued)

LIST OF OTHER IMPORTANT REQUIREMENTS (see para 7.1)

Chapter	Paragraph	Subject
708	-	Bonding and screening
712	4.8	Fire precautions - transmission installations
722	-	Folding components
727	-	Health and usage monitoring system
800	2, 8	Design for maintenance/condition inspection
801	4.3	Slinging facilities - components
802	1.3, 1.5, 2.3	Routine servicing
803	1	Design for repair
804	3	Replacement of pipelines
805	2.7, 11.2	Interchangeability
806	4, 5	Marking and notices
900	7	Tests prior to flight tests
901	7.2.3	Rotor braking tests
905	6.1.1, 9.6, 9.7, 10.6	Demonstration of limits of flight
906	6.4, 7.3, 7.9, 7.14	Engine handling and rotor governing
1015	-	Structures
1016	-	Vibration and dynamic stability

LEAFLET 705/1

TRANSMISSION SYSTEMS

SAFETY CONSIDERATIONS

1 INTRODUCTION

1.1 This Leaflet contains recommendations on the methods to be used to satisfy many of the requirements relating to safety of the rotorcraft as influenced by the design of the Transmission Systems.

2 DEFINITIONS (See also Chapter 201, para 6)

2.1 FAILURE CONDITION (Chapter 705, para 3.2.1)
2.1.1 Failure condition here means technical defect or malfunction from any cause including aircrew or ground crew errors. See also Chapter 100, para 9, Note.

2.2 REMOTE (Chapter 705, para 4.7.4)
2.2.1 Remote here means an event unlikely to occur to each rotorcraft during its total operational life, but which may occur several times when considering the total operational life of a number of rotorcraft of the type. (See reference 1 for numerical interpretation).

2.3 VERY REMOTE (Chapter 705, para 3.2.1)
2.3.1 Very remote here means an event unlikely to occur when considering the total operational life of a number of rotorcraft of the type, but nevertheless has to be considered as being possible.
(See reference 1 for numerical interpretation).

2.4 EXTREMELY REMOTE (Chapter 705, para 3.1.7)
2.4.1 Extremely remote here means an event unlikely to occur when considering the total operational life of a fleet of rotorcraft of the type, but nevertheless has to be considered as being possible.
(See reference 1 for numerical interpretation).

2.5 ON-CONDITION REMOVAL (Chapter 705, para 3.2.8)
2.5.1 On-condition removal here means the removal of a component for rectification, overhaul, or retirement based on inspections, tests, or health or usage monitoring indications which determine that the failure resistance of the component has been reduced below an acceptable level. (See ref. 2)

2.6 PART (EXTENSION TO INCLUDE LUBRICANT(S)) (Chapter 705, para 3.2.9)
2.6.1 The term PART is here extended to include lubricant(s). The integrity of Transmission Systems is dependent upon the use of lubricant(s) substantiated in the Type Approval Tests of Chapter 705 para 5.1, and maintenance of specified volumes, and of satisfactory condition. Practical constraints prevent the classification of transmission lubricant(s) as VITAL PARTS.

2.7 OIL LOSS TOLERANCE (Chapter 705, para 3.2.9)
2.7.1 Loss of Oil. The term here means the reduction in the volume of oil from any self-contained oil system below the permitted minimum, for whatever reason.

2.7.2 Total Loss of Oil. The term here means the reduction in oil level below that necessary for recirculation by oil pumps, dipping gears, or other rotating parts with a gearbox, for whatever reason.

2.7.3 Flight Endurance Following Loss of Oil. The term here means the time interval following an indication from whatever source, of loss of oil, during which the rotorcraft can achieve a cruise flight followed by a power-on landing or autorotative landing.

2.7.4 Test Endurance Following Loss of Oil. The term here means the duration of test of the complete transmission or its components measured from the time whilst running when oil loss is apparent from normal instruments and indicators, following initiation of Total Loss of Oil from a gearbox or other lubricated component. The definition implies the simulation of rapid oil loss from the region of highest pressure in a pressure-recirculating system, or by drainage from the lowest practical point from a splash-lubricated gearbox.

2.8 SHAFT CRITICAL AND SUPERCRITICAL SPEEDS (Chapter 705, para 4.4)

2.8.1 The term shaft critical speeds here refers to shaft rotational speeds at which natural frequencies of deflection occur.

2.8.2 The term supercritical speeds here refers to shaft operating speeds which are above the lowest critical speed.

2.9 TYPE APPROVAL TESTS (LEAFLET 705/2, para 2.1)

2.9.1 The term here applies to the programme of tests on the Transmission Systems which are necessary to obtain CA release for a new rotorcraft type, or new or modified Transmission Systems.
(See also DEF STAN 05-123/1, Chapters 104, and 333)

3 SAFETY ASSESSMENT (Chapter 705, para 3.2.2)

3.1 A JAC Paper covering the requirements relating to rotorcraft and system safety analysis is envisaged following acceptance of the JAC paper relating to Aeroplanes.

3.2 The Design Criteria of Chapter 705 para 3.2.1, to which the Safety Assessment relates, may vary from one class of rotorcraft to another, both in respect of the rotorcraft mission, and in probable significance of the Transmission Systems to the safety of the rotorcraft. It is therefore necessary for the Design Criteria for the Transmission Systems, and the form of the Safety Assessment to be determined by the Rotorcraft Project Director in each case, unless otherwise defined in the rotorcraft requirement specification.

3.3 In general, evidence that the probability of a given failure condition is acceptable can be qualitative or quantitative. A statistical analysis may range from a simple report that interprets test results or makes comparison with similar Transmission Systems or components that have a well proven record, to detailed calculations using numerical probabilities. The approach and any numerical values used should be agreed with the Rotorcraft Project Director.

4 PROTECTION FROM OVERLOAD (Chapter 705, para 3.2.3 and 4.7)

4.1 The Rotorcraft Project Director may be prepared to consider the display of suitable markings on the torque indicators, or placards as a means of complying with the requirements if the inherent characteristics of the rotorcraft and instrumentation are such that the pilot can safely be left to provide the protection (See Chapter 900).

4.2 Automated monitoring of torque limit exceedances should be provided in order to quantify peak values, and hence aid maintenance decision criteria. (See Leaflet 727/3, para 6.4).

4.3 When the power absorbed by a transmission-drive accessory, e.g. electrical generator, is such as to create a hazardous condition in the event of mechanical failure of the accessory in spite of the presence of the weak link in the drive, means should be provided for the accessory to be disengaged from the transmission while the transmission is still running, unless it can be shown that the possibility of mechanical failure of the accessory is remote* and is unlikely to lead to a hazardous condition. (* for definition see Leaflet 705/1, para 2)

4.4 A weak link has to be designed to accommodate the highest peak torque and where the torque on the drive varies considerably with rpm, load, accelerations, etc., especially as in the case of electrical accessories, the weak link may not provide an effective safeguard at the normal operating torque. In such a case, other means of disconnect would have to be provided to permit disengagement of the equipment with the transmission running.

4.5 Where a means for disengagement is provided the conditions under which the accessory may be re-engaged should be established and if these provide for re-engagement in flight the declared technique to be employed should be demonstrated to prove safe operation.

5 CONTROL OF WEAR THROUGH COMPONENT OVERHAUL (Chapter 705, para 3.2.8)

5.1 The traditional method of controlling gearbox wear, corrosion, and age effects through overhaul at fixed intervals (TBO) is no longer universally considered to be entirely satisfactory (e.g. reference 2), and 'on-condition' removals based on health and usage monitoring indications are considered to offer a more reliable means of control.

5.2 Where fixed TBO procedures are to be employed for any part of the life of a rotorcraft fleet it is recommended that the initial TBO at entry to service be limited to a maximum value of 500 hours, and should be substantiated by the tests recommended in Leaflet 705/2, using a minimum of four different sets of components. Increments in TBO, to be agreed with the Rotorcraft Project Director should be limited to 250 hours, and should be based on satisfactory inspection results on a minimum of four samples of each component taken from representative service operation. Relaxations on the initial value of TBO, the maximum increment, and the sample size may be permitted by the Rotorcraft Project Director, particularly where effective Health and Usage Monitoring facilities are provided. (See also DEF STAN 05-123/1 Chapter 208)

6 OIL SYSTEM INTEGRITY (Chapter 705, para 3.2.9, and 4.3.1)

6.1 The importance of the lubricant should be recognised in terms of:-

(i) specification, brand, and product control.

(ii) adequacy and continuity of supply to lubricated components especially to relevant VITAL PARTS.

(iii) freedom from contamination and degradation.

To this end the term PART in connection with Transmission Systems is extended to include the lubricant. It should be noted that the loss of lubricant or failure of the lubricant either in respect of load capacity or cooling function could lead to the loss of control or of motive power. Where this is the case the lubricant containment and supply systems should be treated as GRADE A PARTS. In Transmission Systems components having only one reservoir the lubrication system cannot be considered to have redundancy, unless adequate tolerance to TOTAL LOSS OF OIL is obtained, and the components of the system should therefore be treated as VITAL PARTS. Only the practical difficulties relating to the traceability of lubricant charges and changes prevent this requirement being applied to the lubricant also, in respect of items (i) and (iii) above.

6.2 Experience with grease-lubricated gearboxes has demonstrated grossly inferior life and integrity to oil whether the oil is formulated for aero-engine or for gearbox use.

6.3 Care should be taken to prevent obscuration and staining of oil sight glasses, particularly with some synthetic based oils. As a minimum it should be possible to replace oil sight glasses without the need to drain the gearbox. Alternative means of oil contents gauging should not include optical elements unless they can be demonstrated to be free from obscuration and staining under all operating conditions, including those which cause staining of sight glasses.

7 ROLLING CONTACT FATIGUE (Chapter 705, para 4.1.10)

7.1 Rolling contact fatigue in bearings and in gears, otherwise known as micro-pitting in its early stages, and gross pitting or spalling in a more advanced stage, is characterised by the absence of an endurance limit typical of bending fatigue, and by a very wide scatter in performance of nominally identical components, in quality steels and finished to a high standard. Overhaul at fixed intervals would therefore not be expected to provide an adequate means of controlling the phenomenon and health monitoring is the preferred means, in particular wear debris monitoring. Magnetic plugs have proved effective for this role in rotorcraft gearboxes for many years, and more recent developments are mentioned in Leaflet 727/3, para 6.3.3. Rolling contact fatigue can also occur in shaft support bearings for which magnetic plugs or on-line variants may not provide practical monitoring solutions. Various forms of analysis of vibration or shock data from a single sensor, or alternatively bearing raceway temperature trends may prove to be more effective for these.

7.2 Rolling contact fatigue and other surface wear modes can be initiated at an early stage in the operation of a gearbox or shaft through hard contaminants in the oil (steel, silica, NDT inks, abrasives, etc.,). Filtration to 3 micro-inch absolute standard has demonstrated significant extension of rolling contact fatigue lives in gearboxes fitted with recirculatory oil systems (reference 3). To make an even bigger impact and provide this protection for such gearboxes from first turn-over, and for gearboxes not fitted with recirculatory oil systems it is necessary to achieve equivalent cleanliness levels both of the components and the oil at gearbox assembly - both new build and overhaul. The recommended cleanliness target is Class 800F (see DEF STAN 05-42/2), and an example of how this has been achieved in rotorcraft gearboxes is given in reference 4. There is some evidence (reference 3) to suggest that in gearboxes starting life with this level of cleanliness, and with oil to 3 micro-inch absolute initial cleanliness, wear particle generation rate will be very small and these cleanliness levels will be maintained for a considerable period in service provided that corrosion does not occur and that contaminants are not introduced. For fine filters another important characteristic is the position of the by-pass valve - it should be at inlet to the filter to prevent scouring of previously collected debris into the oil feed jets (reference 4)

8 PREVENTION OF INCORRECT ASSEMBLY OF PARTS (Chapter 705, para 4.1.11)

8.1 Gearboxes contain elements, the incorrect assembly of which can lead rapidly to catastrophe, angular contact and tapered roller thrust bearings being prime examples. If the incorrect assembly of such components cannot be prevented absolutely by detailed design then correct assembly should be verified by the testing under load of all new and rebuilt gearboxes.

9 GEARBOX CONFIGURATION (Chapter 705, para 4.2.2)

9.1 The integrity of Transmission Systems will benefit from the following gearbox configuration considerations:-

(i) Wide separation of input shafts to minimise the risk of power loss due to combat damage.

(ii) Minimise the number of gears and shafts in the combined power output stages.

(iii) Locate freewheels as far downstream in each engine branch of the gearbox as practicable i.e. as close as possible to the power combining stage.

(iv) Reduce to a minimum the number of parts which qualify for the classification of HIGH INTEGRITY (VITAL) PARTS.

(v) Provide adequate clearance for debris resulting from gear tooth failure to be ejected from the meshing area of the affected and neighbouring gears without further damage.
(See also reference 1 - Appendix)

10 DETECTION OF OVERTEMPERATURE, WEAR AND FRACTURE (Chapter 705, paras 4.2.3, 4.2.4, 4.2.5 and 4.3.4)

10.1 Facilities for monitoring the generation of wear particles in the size range 0.1 to 5 millimetres or greater leading dimensions are necessary for the control of rolling contact fatigue and other wear modes which can lead to relatively rapid failure. Simple magnetic plugs provide an effective means of performing this task, and certain on-line devices with electrical outputs also. To be effective the following factors should be included in design considerations:

(i) Probe type sensors inserted in gearbox sumps, upstream of any strainers, pipes, pumps, etc., provide a higher probability of detecting such large particles than systems placed downstream of such debris traps. Debris retention for inspection is important.

(ii) In the siting of such probes and catchment screens consideration should be given to the position of probable sources of wear debris, to oil flow effects, and to the possibility of gravitational drop-out and debris entrapment. Consideration should be given to the installation of lightweight debris catchment trays shaped to encourage debris wash out to an outlet point(s) where sensors can be conveniently fitted.

(iii) The probe or sensor should be fitted in a self-sealing housing with features which permit ready inspection of debris retained. Close attention should be given to the integrity of these arrangements, including:

(a) valve design and sealing arrangements to permit high capture efficiency with the probe in position, and effective sealing with the probe withdrawn.

(b) probe grip design to aid removal/installation and positive locking features.

(c) choice of materials for durability particularly in respect of possible handling/tool damage.

(iv) Non-ferrous wear particles may require inspection of screens or filters for identification of likely source and condition. Incorporation of filters designed specifically for this purpose is preferable to inspection of the main oil filter, both from integrity and diagnostic considerations. (See reference 4)

10.2 Wear debris obtainable from oil samples is generally in the sub-visible size range, the maximum particle size being dependent on oil viscosity and therefore the speed with which samples can be taken following shut down - 30 minutes maximum is recommended. Samples may be conveniently taken from removable wear detector probe ports but the adapters should be designed to ensure that the sampling tube penetrates beyond the level of accumulated debris and permits a true average sample of the recirculating oil. Oil samples thus extracted can be subjected to a range of at-aircraft or laboratory analysis techniques to identify the materials and quantities of each material, expressed as a proportion of the oil sample volume. Potential problems of a relatively long term nature such as corrosion, micro-pitting or bearing liner spinning may possibly be identifiable from oil analysis techniques, in addition to the determination of oil degradation or contamination by water or other fluids.

10.3 Fractures of gears, shafts, bearing raceways, etc., whilst relatively rare events are characterised by relatively rapid crack propagation with minimal debris manifestations. Vibration analysis techniques capable of responding quickly to both localised cracks and to those distributed around a gear coupling, etc., should be employed in an implementation compatible with shortest possible times to fracture - i.e. on-board data processing with cockpit indications where necessary. Component design and selection of materials should be such as to avoid inherently rapid crack propagation, wherever possible.

10.4 Detection of overtemperature in splash lubricated gearboxes be effectively implemented by incorporating a temperature monitoring device within an on-line probe-type wear sensor, where fitted. Detection of overtemperature in recirculating oil systems should be implemented at outlet from the gearbox upstream of the oil cooler. Integrity of the cooling system should be monitored by temperature drop and pressure drop measurements across the cooler.

11 LUBRICATION OF SPLINES (Chapter 705, para 4.4.4)

11.1 The spline teeth of couplings, shafts, and gear drives etc., should be lubricated or treated to prevent fretting and wear of mating surfaces. Oil is the preferred lubricant, grease tending to provide less effective lubrication and requiring greater control of build and maintenance procedures. Splines treated with low friction coatings designed for operation without a lubricant should have adequate life and wear inspection characteristics demonstrated by test, and should be subject to VITAL PARTS procedures where appropriate.

12 LESSONS FROM ACCIDENTS (Chapter 705, para 3.2.11)

12.1 Reports of accidents to civil transport rotorcraft collated by the Civil Aviation Authority have contained sufficient detail to permit a useful extraction of lessons for design. A study of accident data relating to transmission systems causes is published at reference 6, together with a review of recent advances in transmission technology. Details of transmission related occurrences and accidents to military rotorcraft are available from the Rotorcraft Project Director.

REFERENCES

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3	Stevenson B.C.J.	Lubricant Filtration. Engineering June 1983 pp466-469.
4	Astridge D.G.	Improving the Integrity, Reliability, and Life of Helicopter Gearboxes. Paper 5, I.Mech.E. Seminar Pushing Back the Frontiers of Failure in Aerospace Transmissions, London, December 1986.
5	Civil Aviation Authority	British Civil Airworthiness Requirements. Paper G778, 7th October 1985.
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LEAFLET 705/2

TRANSMISSION SYSTEMS

RECOMMENDED TEST PROGRAMME

1 INTRODUCTION

1.1 This Leaflet sets out a recommended test programme to meet the test requirements of Chapter 705. The tests listed below are stated in broad terms and represent the minimum in each case. Additional tests may be necessary for particular installations.

Note: The term 'part' used in this Leaflet also covers lubricants. (See Leaflet 705/1, Para 2.6).

2 GENERAL

2.1 The Type Approval* tests cover:

(i) rig tests,
(ii) preliminary ground tests,
(iii) preliminary flight tests,
(iv) fatigue tests, and
(v) ground and flight tests to demonstrate satisfactory functioning and mechanical reliability.

Production tests and production quality tests are also covered. (* See Leaflet 705/1, Para 2.9 for definition).

3 RIG TEST FACILITIES

3.1 A "rig" may be of any suitable form provided that the required conditions may be obtained in a controlled manner. In all rig tests it is essential to be able to set and record all test parameters (e.g. speed, torque) with precision, and to prevent damage to the unit under test due to imprecise control or undesirable rig characteristics (e.g. unrepresentative vibration). Traceability of test conditions is essential.

3.2 A "rig" must be capable of applying combinations of speed, torque, and external loads where appropriate in a manner representative of operation in service. It should permit the mounting and loading of driven accessories where appropriate.

3.3 A "rig" should be designed in such a manner that health monitoring techniques can distinguish between "rig" characteristics and those of the unit under test (e.g. vibration analysis-characteristic frequencies).

3.4 A "rig" should be capable of applying simultaneously all of the environmental conditions which can be expected to occur within the service life of the unit under test, including sand, dust, and salt sea spray where appropriate.

3.5 Test rigs should be designed to provide means of visual assessment of all external surfaces of the unit under test whilst running, and access for close inspection and partial disassembly where appropriate in the stationary condition. Means should also be provided for installing and readily changing instrumentation cabling between the test area and the control/equipment area, with attention of Electro Magnetic Compatibility and "low noise" power supplies.

3.6 The lubrication system of the unit under test should be completely separate from that of the test rig, and should reproduce as far as possible the characteristics of the service installation. Means should be provided to reduce to a minimum the risk of contamination of a unit under tests with oil-borne material accumulated in previous tests.

3.7 The external surfaces of the unit under test should be protected from oil mist or spray generated by the test rig, in order to facilitate continuous assessment of the unit for oil leakage.

3.8 All instrumentation used should be of an accuracy and reliability to be agreed with the Rotorcraft Project Director, and should be identifiable, and subject to regular calibration by approved methods.

3.9 It is preferred that means should be provided for ground testing the completely assembled transmission system for satisfactory functioning. These should represent the characteristics of the rotorcraft installation as far as is practical. In some cases it may be necessary to use the airframe as a rig. (See Para 8.2).

4 PRELIMINARY RIG TESTS

4.1 Preliminary rig tests are intended to prove the suitability and reliability of parts and sub-assemblies before complete installation and running in the airframe.

4.2 One prototype of each component should be tested for 5 hours under conditions equivalent to those occurring during operation representative of the most severe conditions expected during preliminary flight.

4.3 No test should be commenced without prior assurance that the lubrication system is functional and that all oil feed jets can pass design oil flows.

4.4 All components of lubrication systems (e.g. oil jets, pumps, filters, coolers) should be tested to establish their standards of performance. In all tests described in this Leaflet the effects of the most adverse combination of manufacturing tolerances should be represented. Those combinations which affect pressure or flow can be simulated by appropriate adjustment of oil pressure.

4.5 All parts of pressure lubrication systems should be pressure tested for leaks and distortion to pressures of 1.5 times the maximum operating pressure of the particular part, or 2.0 times the normal operating pressure, whichever is the greater. The effects of temperature associated with the most critical stressing case should be represented in these tests.

4.6 Where appropriate the minimum oil temperature that must be attained before increasing shaft speeds up to the maximum or the application of power shall also be determined.

4.7 Units in which centrifugal stresses are of major importance should be tested for 3 minutes non-stop at 20% above their maximum operating speeds, or, if a governor is incorporated, at 10% above the maximum transient speed permitted thereby.

4.8 Clutches and freewheel units should be tested to establish their engagement and disengagement characteristics including measurement of torque fluctuations, and peak values resulting from slip and sudden engagement where appropriate.

4.9 The components affected should be subjected to 50 cycles simulating starting, acceleration through clutch engagement to maximum operating conditions, rapid deceleration with rotor over-run, and stopping with application of the rotor brake or brakes.

Note: These tests may be made on the rotorcraft as part of the preliminary ground tests of Para 5.

4.10 In addition to tests required to demonstrate the effectiveness of fault detection and safety devices (See Chapter 705, Para 5.5.2), all such devices should be functional through the tests described in this Leaflet. (See Chapter 705, Para 5.5.1).

5 PRELIMINARY GROUND TESTS

5.1 When the rig tests have been completed satisfactorily the rotorcraft will normally be tested on the ground. The transmission should be tested as part of this programme.

Note: The components used in these tests need not necessarily be those used in the rig tests of Para 4, but should be closely to the same standard of design and manufacture.

5.2 Before commencing ground testing of the transmission in the rotorcraft, the following checks should be made:

(i) That the requirements of Chapter 705, Paras 3.1.5, 3.1.6 and 3.1.7 relating to interaction of rotors, "locking of dephasing devices, and rotor-structure clearance are satisfied".

(ii) That transmission system warning devices and monitoring systems are functional (See Chapter 705, Paras 3.2.5 and 4.3.4).

5.3 As early as possible in these tests and before beginning the tests of Para 5.4, vibration measurements of the transmission should be made, covering the widest possible range of conditions including clutch engagement when appropriate, to determine whether there are any conditions of resonance, whirling, high stress, gear tooth meshing, irregularities, or accessory vibration.

5.4 If the vibration measurement results are satisfactory, a test of 25 hours duration should be carried out at conditions covering those expected during preliminary flight.

5.5 At the end of these tests the transmission systems should be inspected for signs of oil leakage and external signs of distress, particularly at interfaces with other systems, and then dismantled and inspected. Particular attention should be paid to gear teeth, bearings, splines, seals, mounting surfaces, and joint interfaces for evidence of wear, surface distress, lack of lubrication, and for incorrect alignment, tracking, or tooth meshing.

6 PRELIMINARY FLIGHT TESTS

6.1 After satisfactory completion of the preliminary ground tests the rotorcraft will normally carry out restricted flying, with periodic inspections. (See also DEF STAN 05-123/1 Chapter 202).

6.2 Normally additional flying will then be carried out within a restricted envelope during which strain gauge tests of the transmission over the range of flight conditions should be made. The tests should include measurements, at suitable points, of both steady and fluctuating torque (or stress) and stresses at other critical areas of the transmission (e.g. hub bending stresses). These should be related to predicted stress levels at those points, and the predicted levels at points of maximum stress reconsidered in the light of any differences found.

6.3 As the restrictions on the flight envelope are progressively removed during further flying, strain gauge tests of the transmission should be continued until the complete range of conditions, including manoeuvres and restarting of an engine in the air, has been covered. To provide information on the most severe fluctuating loads or stresses likely to occur, the effects of rotor vibration, wear and other relevant factors should be investigated as far as possible, with all health monitoring facilities functioning. Sufficient data should be obtained to establish the likely vibration effects imposed by the transmission systems on other systems (e.g. the engines), and the extent of relative movement between components within the transmission systems, and between those and other systems. (See Chapter 705, Paras 2.1.2, 2.1.3, 2.2.1 and 2.2.2).

6.4 After the flight tests the transmission should be inspected for oil leaks, contact with other parts, and external defects, dismantled and inspected as per Para 5.5 and with attention to the possibilities of structural distortion.

7 FATIGUE TESTS

7.1 Those parts of the Transmission Systems which are shown to require an established Safe Fatigue Life are to be subjected to fatigue tests conducted with the representative gearbox mounted in a suitable test rig as specified in Chapter 201. Where more than one fatigue test is performed the specimens tested should be selected such that they do not have common batch origins, or heat treatments, and are not tested in the same gearbox casing.

7.2 Parts and assemblies used in the fatigue substantiation tests should be fully representative of the design, materials, and manufacturing process of those proposed for CA Release. In tests aimed at determining the tooth root fatigue strength of gears, care should be exercised when selecting an alternative lubricant to those specified. Lubricants can influence root bending strength through differences in their reaction to the environment (e.g. water absorption) and in their effect on crack propagation. Use of a non-specified lubricant should therefore be subject to the agreement of the Rotorcraft Project Director. Results in which failure initiated at working surfaces (e.g. gear tooth flanks) may not be acceptable if a non specified lubricant was used.

7.3 The test environment should be as representative as possible of service conditions, as determined in agreement with the Rotorcraft Project Director.

7.4 The fracture monitoring and fatigue usage monitoring elements of the Health and Usage Monitoring provisions proposed for the Rotorcraft Transmission Systems should be applied to all fatigue tests.

8 GROUND AND FLIGHT TESTS TO DEMONSTRATE SATISFACTORY FUNCTIONING AND MECHANICAL RELIABILITY

8.1 Before the tests begin, the transmission systems should be inspected in detail and a record made of the condition of the components and of the clearances between parts in relative motion, including dimensions of individual parts where appropriate. A record should also be made of the performance of any transmission accessory (e.g. oil pump), the engagement characteristics of the clutch and similar information concerning the functioning of the equipment. The degree of imbalance in the rotors should be recorded also.

8.2 Endurance tests will normally be carried out as part of the testing of the complete rotorcraft. The ground testing may however be carried out in conjunction with the appropriate engine(s) by using a rig which reproduces the loadings which occur in the rotorcraft (Ground Test Vehicle, GTV). For a single engined rotorcraft the tests should include 150 hours ground testing and 100 hours flight testing. For a twin engined rotorcraft these times should be increased by 50 hours in each case and periods of running of each engine alone should be included. During these tests representative movements of the controls should be undertaken.

8.3 All environmental service conditions which are agreed by the Rotorcraft Project Director to be significant should be reproduced in the programme of tests, with durations to be agreed. (See Chapter 705, Paras 1.4 and 4.3.1).

8.4 It is not safe to assume that all lubricants to the same Joint Service Designation (DEF STAN 01-5/5), US MIL-SPEC, or other type classification systems will result in similar wear performance of rotorcraft transmission systems. It is desirable therefore that each lubricant of different sources and manufacturers identification that is proposed for service use should be included in the transmission endurance substantiation programme. Representative samples should be analysed for degradation at the end of the endurance tests.

8.5 It should be demonstrated that no dangerous torsional or flexural vibrations occur at any permissible torque and at any rotational speed up to the maximum engine overspeed or the maximum permissible rotor speed, whichever is the greater.

8.6 Additional tests may be required to demonstrate that requirements have been met in respect of:-

		Chapter 705 Para No.
(i)	oil loss tolerance	3.2.9
(ii)	containment of fractured parts	2.1.1,4.2.2
(iii)	vibration	2.1.2,4.1.3
(iv)	noise	2.3.1,4.2.6
(v)	contaminant shielding	2.1.7
(vi)	internal cleanliness	4.1.10,4.3.3
(vii)	oil sight glass transparency	3.2.9
(viii)	impact loads	3.2.3,4.1.3
(ix)	assessment of safety devices*	3.2.6,4.1.10 4.2.3,4.2.4,4.2.5
(x)	rotor brakes	3.2.10,4.6.1
(xi)	cooling fans	4.3.5
(xii)	rotor drive shafts	4.4
(xiii)	torque limiting device	3.2.3
(xiv)	clutches and freewheels	4.5.1
(xv)	accessory drives	4.7
(xvi)	other potential failure modes	3.2.11

*Note: In the assessment of safety devices relating to wear or damage, monitored tests in which degradation takes place gradually from the undamaged condition tend to be more useful than "seeded fault" tests which require disruption of the assembled unit to introduce the fault, and therefore lack the continuity in degradation from the "normal" state of components.

8.7 Critical Loads. Any components which can be subjected to critical loading conditions not adequately covered by the standard test conditions, e.g. bearings which can be off-loaded by gear reaction or preloading, should be subjected to fatigue and endurance tests or additional tests acceptable to the Rotorcraft Project Director.

8.8 Inspection of the transmission systems on completion of ground tests (before flight) should involve external examinations and a minimum of dismantling which should not materially affect the flight test results. Full analysis of health and usage monitoring data should be undertaken before commencement of flight tests.

8.9 On completion of the flight tests the functional checks, and detailed inspection (see Para 8.1) should be repeated so that deterioration and wear can be assessed and compared with monitoring system indications and data. Representative samples of lubricant from each lubrication system should be analysed for deterioration and determination of useful service life.

8.10 Determination of the initial time in service before strip inspection (TBO) should be made on the basis of the condition of components examined from all of the above tests.

9 TESTS OF MODIFICATIONS

9.1 Before a modification is introduced, tests of the modified parts may be necessary. (See Chapter 705, Para 5.4).

9.2 When the modification proposal is agreed, the programme of tests required should be agreed with the Rotorcraft Project Director.

10 PRODUCTION TESTS

10.1 GENERAL

10.1.1 Each unit needs to be submitted to a production test to comply with the requirements of Chapter 705, Para 5.2.1. The following indicate the lines on which the tests will be based.

10.1.2 Each component or assembly which is subjected to hydraulic or pneumatic pressure during operation should be tested to the appropriate requirements of Chapter 703 and 704.

10.1.3 An oil pump or similar transmission systems accessory should be tested to show that, when temperatures and affected clearances have become stable under the most adverse operating conditions, functioning is satisfactory. It should then be dismantled for detail inspection and, after reassembly, tested to show that performance (e.g. delivery rates and pressures) is satisfactory and that any other operational of functional requirements are fulfilled.

10.1.4 Production tests should not be undertaken without the assurance that the lubrication and cooling system is functional and that all oil jets are passing oil flows in accordance with the design.

10.1.5 Production quality assessment monitoring provisions, designed to assess build cleanliness, gear tooth meshing conditions, and lubrication system performance, should be functional throughout the production tests, and should be indicating satisfactory condition of the transmission system before release to service.

10.1.6 A clutch should be tested for a total of 2 hours including 1 minute at the speed corresponding to the maximum engine overspeed or the maximum permissible rotor speed, whichever is the greater, 10 accelerations from idling speed through engagement to maximum speed and power and periods at operational conditions including maximum power and speed and maximum torque. It should then be dismantled for detailed inspection. After re-assembly it should have a final test of 30 minutes in periods at operational conditions with 5 accelerations as above, and the engagement and disengagement characteristics should be checked.

10.1.7 A gearbox should be tested for 2 hours including 1 minute at the speed corresponding to the maximum engine overspeed or the maximum permissible rotor speed, whichever is the greater, and periods at operational conditions including maximum power and speed and torque. After dismantling, or partial dismantling, for detailed inspection and re-assembly, it should be tested for 30 minutes at operational conditions, and finally inspected for leaks.

10.1.8 The test of the over-running clutch (or freewheel) should be the same as that of the gearbox (See Para 10.1.17) but in addition there should be 10 cycles, before the inspection, of engagement, acceleration to maximum power and speed and rapid deceleration to give over-run operation and 5 similar cycles during the final test. The unit should be inspected for potential slip.

10.1.9 Rotor brakes should complete 10 operations in accordance with recommended practice followed by inspection and 5 operations after re-assembly. The unit should be inspected for leakage.

10.1.10 Should a part fail during the test or should inspection show any part to be unsatisfactory, the test (or part thereof) and inspection, should be repeated. Data from the health monitoring and quality assurance monitoring provisions should also be examined.

10.2 RELAXATIONS

10.2.1 When a satisfactory production standard has been established by results from a sufficient number of a particular unit, relaxations of testing and dismantling for inspection may be permitted at the discretion of the Rotorcraft Design Authority. Relaxations may be influenced by the quality assurance monitoring provisions implemented.

10.3 PRODUCTION QUALITY TESTS (See also DEF STAN 05-123/1, Chapter 203 Annex B)

10.3.1 The production quality test should be approximately one quarter of the tests of Para 8 or equivalent thereto, and should have all the quality assurance monitoring provisions applied.

10.3.2 On completion of the test, the unit should be dismantled for detailed inspection and, after re-assembly, should complete the final part of the acceptance test.

CHAPTER 706

ELECTRICAL INSTALLATIONS

1 INTRODUCTION

1.1 The requirements for this chapter apply to all electrical systems in aircraft. An electrical system comprises those electrical units and components which generate, distribute and control the supply of ac and/or dc electrical power for other systems.

1.2 Characteristics of electrical generating and distributing and consumer units are given in BS 3G100 or superseding Defence Standards.

NOTE: BS 3G100 is gradually being incorporated into appropriate Defence Standards.

1.3 Leaflet 706/1 makes general recommendations concerning the design of electrical installations, and Leaflet 706/2 gives further information on the standard power supplies and their application to the installations. Leaflet 706/3 lists British Standards and Defence Standards which contain advice on accessories and components.

2 GENERAL REQUIREMENTS

2.1 The operating requirements will be stated in the Aircraft Specification.

2.2 All installations and systems shall function correctly under all conditions, on the ground, in flight and at the altitudes for which they are required to operate. In pressure cabin aircraft, any electrical equipment necessary to enable the aircraft to return to base safely in all weather conditions shall continue to function satisfactorily in the event of cabin pressure failure occurring at maximum distance from base. The maximum distance from base should be related to the maximum altitude at which the aircraft can operate in the unpressurized condition.

2.3 The aircraft designer shall categorize all electrical services as either *Essential, Standby, or Emergency, as appropriate to the aircraft and the operational role specified. The Aircraft Project Director/specification will specify the reliability required in terms of failure probability for each of these categories. The designer shall prepare a safety assessment of the electrical system in accordance with the requirements of Chapter 117 (to be issued) using the reliability targets quoted in the specification/Leaflet 117/2 or as agreed with the Aircraft Project Director.

*NOTE:

a) Essential Services are those required for continued safe flight.

b) Standby Services are those providing secondary alternatives to a primary system.

c) Emergency Services are defined in Chapter 105, para 11.3.

2.4 No failure condition of the electrical installation resulting from a single failure, a second failure or unrevealed dormant fault, or a combination thereof shall jeopardise:

a) the safety of the aircraft, or its occupants in flight, taxying, take-off, landing, or on the ground

b) the ability of the crew to escape from the aircraft

c) the ability of the aircraft to return safely from a mission subsequent to such a failure.

2.5 All electrical equipment, including wires and cables, shall be so installed as to:

a) operate satisfactorily in the particular local environment having due regard to the possibility of that environment becoming more adverse as a result of a failure or battle damage (see Chapters 101 and 112)

b) minimize the secondary effects of primary failure (however caused)

c) remain unaffected by moisture and liquids liable to come in contact with it

d) be easily accessible for inspection and servicing

e) have minimum vulnerability, to battle damage (see Chapter 112)

f) minimize the risk of fire or explosion from inflammable liquids and gases and from electrically initiated explosive devices, both in flight and during ground operations (see Chapter 712)

g) minimize the risk of electrical shock or injury to personnel, e.g., from exposed live conductive parts, fire, toxic fumes etc. Exposed live parts of systems and equipment shall be mechanically protected so that the probability of short circuits and earth faults is remote

h) minimize the risk of accidental damage by the crew and ground personnel

j) be marked so as to make identification possible under all conditions of servicing (see Chapter 806, para 8).

2.6 The changeover from external to internal power supply shall be automatic when the aircraft generator(s) come "on line", except where the two supplies can be safely operated in parallel.

2.7 Where load shedding occurs under conditions of engine or generator failure, provision shall be made for retaining those services, including the essential services defined in Leaflet 712/1 para 2.7, necessary for the time stated in the aircraft specification. (see para 3.3, Chapter 600 para 5 and Chapter 100 para 9.2).

2.7.1 After Total Generator Failure. Power supplies for the following services shall be available to enable continued controlled flight and emergency landing to operate for the period defined in the aircraft specification:

a) primary flight controls (when electrically actuated)

b) power plant controls (when electrically actuated)

c) stability augmentation systems (when these are critical to flight safety)

d) one indicator each displaying aircraft attitude, airspeed and altitude, including operation of the relevant pitot-static de-ice and instrument lights

e) one radio communication equipment and minimum internal communication equipment

f) one navigation aid

g) re-excitation of an inoperative generator

h) re-start of an inoperative engine

j) jettison of stores (see Chapter 100, para 18)

k) escape system(s)

l) minimum instrumentation displayed, and continued fuel supply

m) environmental control system.

2.7.2 After Single Generator Failure of a Multiple Generator System. Power supplies for the following services shall be available in addition to those listed in para 2.7.1 to allow completion of the aircraft sortie:

a) fuel management

b) where navigation, communication or instrument systems are duplicated, only one of each system may become inoperative

c) any service essential for the completion of the operational role (see para 3.3.2).

2.7.3 Continuous Supply. Power supply for the following shall be available at all times independent of any aircraft master switch or load shed arrangements other than battery contactors:

a) fire extinguishers and detection

b) *emergency lighting

c) *systems having volatile memory (when electrically activated)

d) primary flight controls

e) crew intercommunication (when appropriate)

f) escape means

g) ADR and CVR (see Chapter 100, paras 21 and 22)

*NOTE: If these systems are fitted then their continuous power supply must be provided by additional batteries charged from the aircraft main bus bars.

See also Chapter 100 para 9.2.

2.8 Emergency escape illumination shall be provided independent of normal power sources (see Chapter 102, para 3.1 (d)).

2.9 Means of isolating electrical supplies shall comply with para 8.

3 POWER SUPPLIES

3.1 GENERAL

3.1.1 The power supplies shall be selected from those listed in Table 1 and shall meet the requirements of BS 3G100 Part 3[1]. The nominal voltage and nominal frequency at the bus bars shall be as specified in Table 1. All consumer equipment shall be suitable for operating on a power supply conforming to the requirements of BS 3G100 or superseding Defence Standard.

NOTE : Where there is a requirement for a different power supply to those listed in Table 1 and Table 2 - approval must be obtained from the Aircraft Design Authority and agreed with the MOD Project Director.

3.1.2 For installed aircraft electrical systems, transient voltage spikes shall not exceed the limits for equipment exported spikes given in DEF STAN 59-41, Part 3, test method DCE03.

3.1.3 All generators shall be self exciting.

3.1.4 The electrical installation shall be capable of being functioned from a ground supply whose characteristics conform to those specified in BS G219

3.1.5 Means shall be provided to give immediate warning (see Chapter 107, para 12.1) of the failure, or deterioration beyond trip limits of voltage or frequency (see Leaflet 706/2), of power supplies to the bus bars from each source of electrical power. The warning shall identify the particular source affected and shall not itself be rendered inoperative by any single failure of the power system.

3.2 SYSTEM CAPACITY (NORMAL)

3.2.1 The generating capacity shall be adequate to meet the largest transient peak loads which can occur with the aircraft fulfilling its operational roles, including taxying, take-off and landing and shall meet the failure conditions considered in para 2. Generating capacity shall not rely on assistance from storage batteries, except that peak demands of short duration may be proposed by the Aircraft Design Authority for consideration by the MOD Project Director.

3.2.2 The generating capacity shall be at least 50 per cent greater than the maximum continuous demand (see also Leaflet 706/2, para 3). This estimate shall be based on the fully equipped aircraft as at the date of the Final Conference or earlier as agreed with Aircraft Project Director.

3.2.3 With throttles set at approach r.p.m. there shall be sufficient generated power to provide full operational facilities when descending at minimum speed from maximum attainable altitude to ground level. When a constant speed drive is fitted, there shall be sufficient generated power to provide full operational facilities if the engine power is set at 'flight idle'.

3.2.4 With the necessary engines running at a speed approximately 20% above minimum ground idling speed, it shall be possible, without discharging batteries, to exercise and check on the ground all electrical equipment installed in the aircraft.

3.3 SYSTEM CAPACITY (EMERGENCY) (see also Chapter 100, para 9)

3.3.1 Sufficient power shall be available to retain those electrical services necessary, for operating the aircraft for the endurance period stated in the Aircraft Specification.

3.3.2 The Aircraft Specification will state whether the aircraft is required to complete the mission without the loss of services after the failure of a particular supply channel.

3.3.3 In the event of main generator failure alternative supplies shall be provided from either:

a) the main battery

b) a standby battery carried solely for the purposes of emergency power

c) a standby generator powered either by an emergency ram air turbine, or an auxiliary power unit capable of starting and operating in flight.

3.3.4 Where batteries are installed the duration of operation when supplying emergency loads shall be determined and shall be declared for inclusion in Air Publications (see also paras 2.4 and 7.2.1).

4 GENERATOR SYSTEMS

4.1 GENERAL

4.1.1 Each generator system shall provide an output conforming to the supply characteristics required by BS3G100 or superseding Defence Standard (see Table 1) and shall be capable of withstanding the transient condition specified in BS3G100 Part 3.

4.1.2 Self exciting generators are required, but additional instantaneous excitation shall be provided to overcome a possible emergency situation.

4.1.3 Provision shall be made to limit the input torque to a generator unless means to disconnect the drive is provided.

4.2 INSTRUMENTATION

4.2.1 Means shall be provided to monitor the voltage and load provided by each generator, and in addition a frequency meter shall be provided for ac systems.

4.2.2 Warning lights shall be provided to indicate whenever a generator output is disconnected from the distribution system. This warning shall be repeated in a central or master warning system (where fitted).

4.2.3 Warning and control systems shall not be reliant on the power supply they monitor.

4.3 LOAD SHARING

4.3.1 Provided regulation circuits ensure the total load is evenly divided:

a) similar dc generators may be operated in parallel

b) similar ac constant frequency generators may be operated in parallel.

4.3.2 Ac frequency wild generators shall not be operated in parallel unless their output is rectified.

4.3.3 The capacities of generators operating in parallel shall be selected in accordance with para 3.

4.4 GENERATOR CONTROL AND PROTECTION

4.4.1 Each generator installation shall have its own segregated control and indication system. Separate or switch selected instruments may be installed.

4.4.2 Control shall provide for interruption of the generator output and separate interruption of the generator excitation.

4.4.3 Each generator shall be protected from overload and fault conditions. Ac generators shall be protected from excessive imbalance of the load. The response of a particular system shall be adequate to safeguard the equipment but shall not be so sensitive as to give nuisance disconnections.

4.5 GROUND CURRENT EFFECTS

4.5.1 The metallic airframe is used normally as the ground conductor for:

a) the negative or return in a 28V dc system

b) the neutral in a 115V ac single phase system

c) the star point in a 200/115V ac 3 phase system.

Where a metallic airframe is not used special bonding conductors shall be installed (see Chapter 708).

4.5.2 Main connections to ground shall be adequate for the conduction of any current, including possible fault currents, which it may be necessary to carry.

4.5.3 Main connections to ground shall be located such that high circulating currents do not exist in the vicinity of compass detectors or directionally sensitive radio aerials.

4.5.4 Where special ground bonding conductors are used these are considered in a similar manner to cables carrying high currents. Electromagnetically sensitive equipments shall be located remote from these installations.

4.6 COOLING

4.6.1 Adequate ventilation and cooling shall be provided to meet all possible operating conditions.

5 CIRCUIT

5.1 BASIC REQUIREMENT

5.1.1 The aim shall be to ensure that the installation of cables, bus bars, electrical components and their supports and insulation will be such that they are capable of withstanding throughout the life of the aircraft the worst effects of vibration (in particular the effects of vibration and environmental conditions such as gun firing effects, severe wind and moisture prone areas, high temperatures, condensation, fuel contamination, hydraulic oil contamination and de-icing fluids). All conditions likely to be encountered on the ground and up to the design flight envelope shall be taken into account.

5.1.2 Consideration as required by DEF STAN 00-40 shall be given to reliability and maintainability. The guidance on reliability and maintainability procedures and practice given in DEF STAN 00-41 shall be followed where practicable.

5.2 CIRCUIT CONTROL

5.2.1 All switching of electrical circuits shall be as shown in Table 1.

5.3 CIRCUIT PROTECTION

5.3.1 Each electrical circuit and distribution feeder shall be automatically protected against short circuits (see also Table 1). When safety considerations so dictate, similar protection shall also be made against overload. Electronic Circuit Protection devices that can be reset whilst in flight manually or automatically should be limited to 3 or less circuit resets for essential loads. The circuit protection device should be inhibited from attempting to reset if short circuit or overload conditions persist.

NOTE: This requirement does not apply to heavy duty circuits for direct electrical starting of engines.

5.3.2 When circuit breakers are used they shall be of such design that although the external operating mechanism may be held closed, the tripping device when carrying overload current will open the circuit.

5.3.3 Where protective devices are used in series, their characteristics shall be compatible (see also Leaflet 706/1, para 3).

5.3.4 The characteristics of devices used to limit current, voltage or frequency in any one circuit shall be compatible with each other and their effective operation shall not be prevented by any power supply deviations which may occur.

5.3.5 Protective devices shall be disposed in groups, physically protected to decrease vulnerability, and shall be installed as near to the bus bar or appropriate distribution point as possible. Such devices and their controls shall be accessible and easily identified. A statement of the nominal rating of each fuse shall be provided in the vicinity of the fuse. Where neon lamps are used as fuse indicators, they shall adjoin the fuse monitored.

5.3.6 In certain circuits, as agreed with the Aircraft Project Director it will be necessary for a circuit breaker or fuse to be under the control of a particular crew member. In such circuits the full circuit protection of para 5.3.1 shall also be provided. Circuit breakers shall not normally be used as ON/OFF switches.

5.3.7 Provision shall be made for carrying as spares at least 5 per cent and a minimum of 1 of each type of fuse of the ratings installed. They shall be located at the appropriate station in the aircraft. Fuse stowages shall be provided with means for identifying the fuse rating. Unused fuse holders shall be fitted with dummy fuses.

5.4 SPARE CIRCUITS

5.4.1 For new aircraft, allowance shall be made for a minimum of ten percent increase in the number of main cable runs, connectors, circuit breakers and fuse holders above those in use at the time of the Final Conference, or earlier as agreed with the Aircraft Project Director, and sufficient space be left in junction boxes and control panels for such increases (for updates to existing aircraft a five percent increase is acceptable).

5.5 GROUNDING OF CIRCUITS

5.5.1 When single pole wiring is used, the airframe shall be capable of carrying the electrical load of grounded equipment between the local ground points and the main ground points with a voltage drop not exceeding 0.5 volt. This is of particular importance for the airframe path between the ground point of each power source, i.e., Generator, TRU, or battery, and that of its associated voltage regulator.

5.5.2 Grounding attachments shall be so secured to the airframe that during their operational use there will be no risk of the attachment loosening.

5.5.3 The grounding attachments to the airframe shall be protected to prevent corrosion at the joint.

5.5.4 Failure of a grounding cable or attachment shall not adversely affect more than one circuit or more than one part of a multiple circuit, nor shall it cause the inadvertent operation of any circuit (see Leaflet 706/1, para 2.1.3). No more than four wires shall be connected to a single ground stud provided the effect of failure of the stud is no more severe than the associated supply protection or control device going open circuit.

5.5.5 When terminal block connections are used for grounding groups of cables the common ground shall be connected to the airframe by at least two cables and two separate grounding points, each capable of carrying the full ground current of all equipments connected to the block with a voltage drop not exceeding 0.1 volt. (see also para 5.5.8).

NOTE : Modern earthing modules have built in earthing feet which dispenses the need for earthing cables.

5.5.6 Ground points shall be clearly identified and readily accessible to servicing personnel (see Chapter 806).

5.5.7 The grounding of all electrical panels, chassis equipment and wiring components shall be such that under any fault condition the protection device shall operate and the sustained voltage developed between exposed metal and the aircraft structure shall not exceed a maximum of 50 V ac and 120 V dc (including ripple).

5.5.8 Grounding terminations of ac and dc systems or systems having different voltage supplies shall not have a point of common mode failure and shall not be connected to a common stud, bolt or other connection.

6 INSTALLATION

6.1 GENERAL

6.1.1 All electrical equipment shall comply with BS3G100 or superseding Def Standard.

6.1.2 All electrical equipment shall be adequately attached to airframe, sub assembly or power plant unit. Neither the items of equipment nor their immediate casings shall be subject to, or assist in the carriage of structural loads existing in the assembly to which it is attached.

6.1.3 All installations shall comply with the requirements for Electromagnetic Compatibility contained in DEF STAN 59-41.

6.1.4 Interconnecting electrical wiring shall be installed using components, (e.g., terminal posts, terminal blocks, terminal junction modules or electrical connectors) and methods of support (e.g., cable ducts, conduit, open looms in conjunction with cable ties and cleats) that meet with specifications approved by the Aircraft Design Authority and agreed with the MOD Project Director.

6.1.5 Wiring of a type appropriate to the electrical load, duty and environment shall be used. (See also Chapter 706 para 6.6, Chapter 712 para 6.6.4 and DEF STAN 61-12)[2].

6.1.6 The choice of termination shall be determined by the duty and environment envisaged. Components featuring crimped cable ends are preferred (see DEF STAN 59-71 (Part 1)). Where necessary cables, connections and terminations shall be provided with additional protection.

6.1.7 The cable and type of termination shall be selected and approved by the Aircraft Design Authority and agreed with the MOD Project Director.

6.2 TERMINAL POSTS (see DEF STAN 59-3)

6.2.1 Terminal posts shall be adequately supported in an insulator suitable for the environment. Terminal posts shall be sized appropriately for the circuit rating. The terminal post manufacturer's recommended electrical and physical stud loading shall not be exceeded.

6.3 TERMINAL BLOCKS

6.3.1 Only approved assemblies of appropriate rating shall be installed. The manufacturers' recommended configurations shall be followed. Terminal blocks shall be selected appropriate to the circuit rating.

6.4 ELECTRICAL CONNECTORS

6.4.1 Circular electrical connectors shall be selected from the types listed in DEF STAN 59-35 (Part 0), Section B, Table C-1 which are approved for airframe-fit and avionics applications. Limitations of use shall be noted, e.g., unsuitability for blind mating etc. The type of connector selected shall be agreed with the Aircraft Project Director.

6.4.2 Cable-clamping accessories shall be used to provide proper restraint for wire and cables. Straight and angled outlets are available for each type of airframe connector.

6.4.3 The applications of mated pairs of connectors shall be such that in the event of separation, only female (socket) contacts shall be connected to the live supply. Unmated connectors shall be fitted with protective covers.

6.5 INTERCONNECT WIRING

6.5.1 When installed, all electrical wires shall be adequately restrained with approved devices appropriately applied. Electrical wires shall be held clear of moving mechanical parts.

6.5.2 All electrical wires shall be clearly identified either by code, legend, or colour or combination thereof. Each wire of a multicore cable shall be identified. (see Chapter 806).

6.5.3 All wires and cables shall be installed so as to minimise the probability of inadvertent operation or malfunction of equipment resulting from:

a) resistive voltage components and/or fast flux coupling generated by the application of the lightning test waveforms defined in the UK AEA Culham Laboratories Report CLM-R-163 (see Leaflet 708/3 and DEF STAN 59-31)

b) coupling with electromagnetic fields to the levels defined in Naval Weapons Specification (NWS) 1006 Annex A and BS 3G100 or superseding Defence Standard.

6.5.4 All wires and cables associated with fire warning and fire extinguishing equipment shall be so routed that the probability of damage resulting from the break up of a high energy rotor, the rupture of a high energy storage container, or a crash landing is reduced to a minimum. (see Chapter 712 and Chapter 100, para 24).

6.5.5 At distribution boxes (panels) the sections carrying voltages exceeding a nominal 28 volt shall be separated from sections carrying voltages not exceeding 28 volt. High voltage sections shall be clearly marked with the highest effective voltage value found in that section. Lids (covers) of the distribution boxes (panels) shall also be clearly marked with the voltage values and shall be so designed that they cannot be replaced on the wrong box.

6.5.6 All external terminals of electrical equipment shall be locked by methods to be agreed with the Aircraft Project Director (see also Leaflet 706/1, para 5.3).

6.5.7 In-line splices shall not normally be used; when their use in general airframe wiring is unavoidable, the agreement of the Aircraft Project Director shall be obtained. The location of these splices shall be clearly defined in the aircraft drawings and the Air Publication. (Order 4901 of AP 100B-01).

6.5.8 The outer conductor of a screened cable shall be terminated with a suitable RFI backfitting. Should this not be possible, cable screens may be terminated by a short flying lead of adequate cross section provided it is not grounded within any equipment case.

6.5.9 Where conduit is installed adequate drainage shall be provided.

6.5.10 Wiring between terminal blocks or connectors and an airframe grounding stud shall be clearly marked as a 'Ground' connection.

6.5.11 Wiring harness power density criteria shall be as specified in US MIL-W-5088L para 3.8.8-1.1.

6.5.12 Compatibility between circuit breaker time/current characteristics, cable ampere rating characteristics and load characteristics shall be as specified in US MIL-W-5088L, para 6.7.

6.6 SELECTION OF INTERCONNECT WIRING

6.6.1 For national programmes, general purpose aircraft wires and cables are to be qualified to DEF STAN 61-12 Part 33 or an equivalent Specification demonstrated to fulfill these requirements as a minimum. On international projects, the technical basis of national Specifications should be applied. See also Leaflet 706/1 para 4.1.14

6.6.2 Wires and cables shall have a demonstrated resistance to arc tracking as defined in Defence Standard 61-12 Part 33.

6.6.3 Wires and cables which employ an outer sheath, commonly known as a topcoat, in their construction are to have proven topcoat durability, by compliance with test procedures defined in DEF STAN 61-12 Part 33 or an equivalent Procedure demonstrated to fulfill these requirements as a minimum. It should be noted that wires and cables compliant with these requirements may contain polyimide (commonly referred to under the trade name Kapton) in the construction of their insulation. See also leaflet 706/1 para 4.1.14.

7 BATTERIES

7.1 CONTROL (see also Chapter 107, Table 6, Item 2)

7.1.1 Each battery shall be capable of isolation from the aircraft electrical system except those services required in an emergency and specified in the aircraft specification.

7.1.2 The battery shall be designed to withstand thermal runaway without failure of the case structure or leakage of electrolyte. The battery installation shall be designed to minimise damage to the airframe from battery electrolyte in the event of a cell or battery case failure from any cause.

7.1.3 A warning should be provided in the Pilot's Notes, that in the event of a total loss of generated power, the battery must be connected to its appropriate bus bar regardless of an overheat condition being indicated.

7.2 CAPACITY

7.2.1 The capacities of all batteries installed in the aircraft and their duration appropriate to their maximum loads shall be agreed with the Aircraft Project Director (see also para 3.3). It is Service policy to use batteries until they give 80 per cent of specified capacity.

7.3 INSTALLATION

7.3.1 Batteries shall be so installed that they are adequately protected from extremes of heat and cold during flight (see also para 9.1).

7.3.2 Unless, of the sealed type, batteries shall be enclosed in compartments vented to atmosphere and, for batteries requiring assisted ventilation, suitable venting provision made. The installation shall be such as to minimise the risk of explosion or of corrosion of the aircraft or its equipment. The ventilation shall be adequate during all conditions of flight, whilst taxying and during servicing and shall allow for all conditions of battery malfunction (see also Chapter 407, para 22).

7.3.3 The design of the battery stowages shall permit rapid removal and replacement of batteries without hazard and the need for special ground equipment under all climatic conditions.

7.3.4 Stowage shall be provided for the battery connectors when they are of such type that when free they might cause a short circuit to the airframe.

7.3.5 When a separate emergency battery is provided for the operation of certain vital services, the installation shall be engineered to the same standard as the main electrical installation.

7.3.6 Metal-cased batteries shall be electrically insulated from airframe structure. Where bonding to earth of the case is necessary, the connection shall be made through a fusible element. An appropriate dormant failure check shall be devised.

7.4 CHARGING

7.4.1 The need for a charge control system and the provision of a warning that the battery is not being charged shall be considered.

7.4.2 Provision shall be made for the use of suitable equipment for cooling the battery and/or charge control system, and ventilating the battery and/or battery compartment.

7.4.3 Means shall be provided to minimise the risk of overcharging or overheating of batteries.

7.4.4. Nickel-Cadmium batteries used for engine or Auxiliary Power Unit (APU) and/or as part of the main power supply shall have:

a) an automatic charge control system to meet the requirements of para 7.1.2 and/or

b) an overheat warning system at the flight crew station together with a means of disconnecting the battery from its charging source in the event of an over temperature condition.

8 FIRE PRECAUTIONS (see also Chapter 712)

8.1 Means shall be provided to disconnect electrical power supplies not essential to post crash conditions. If automatic disconnection of power supplies and automatic activation of the fire extinguisher systems are required, these will be stated in the Aircraft Specification and shall be discussed with the Aircraft Project Director.

8.2 The systems which are required during or after a crash shall include:

a) means of escape

b) fuel Shut Off valves

c) fire extinguishing systems

d) communication

(e) emergency escape/evacuation illumination (See Chapter 102 para 3).

8.3 These circuits shall be left connected to a battery and shall be protected so that the risk of their causing a fire under these conditions is a minimum (see para 8-1).

8.4 Electrical cables, terminals and equipment in designated fire zones, that are used during emergency procedures, shall be at least fire resistant .

8.5 Main power cables (including generator cables) in the fuselage shall be designed to allow a reasonable degree of deformation and stretching without failure and must:

a) be isolated from flammable fluid lines, or

b) be shrouded by means of electrically insulated flexible conduit, or equivalent, which is in addition to the normal cable insulation.

8.6 Insulated electrical wire and cable installed in any region of the aircraft shall comply with the flammability requirements in BS G212, para 2.28A Flammability Test Method 3.

9 TEMPERATURE LIMITATIONS

9.1 The design of the electrical system shall be such that the specified temperature limits of the component parts are not exceeded. The possibility of the components' environment becoming overheated due to the failure of another adjacent system shall be considered at the design stage (see Chapter 712, para 10-6).

10 GROUND CHECKING AND SERVICING

10.1 The electrical system shall include a means for connection to the external power supply system for either servicing, pre-engine start drills, or engine starting (see also Chapter 802, para 1-7). Ac and/or dc ground power receptacles shall be fitted to comply with BS G173. They shall be mounted at a suitable angle to reduce to a minimum the effect of the weight of the ground trailer cables and shall be located so that ground crew may safely connect or disconnect ground supply, when engines are running, flight control surfaces are being moved and/or wings are spread or folded. When two or more connectors are fitted, the spacing between each shall be of such as to prevent the possibility of fouling by the mating connectors or their associated cables.[3]

10.2 Facilities shall be provided for checking the performance of primary and secondary power supply circuits, including dormant circuits, by means of built-in or external test equipment, without disturbing the internal or external wiring of, the equipment (see also Leaflet 706/1, para 10).

10.3 Provision shall be made in the aircraft installation for power supply socket outlets. The number, type and location of such socket outlets shall be agreed with the Aircraft Project Director.

10.4 Servicing ground bolts in accordance with SBAC Drawings RS 682 or equivalent shall be provided.[4]

11 TESTS

11.1 The electrical system shall be subjected to functional tests on the working rig required by DEF STAN 05-123 Chapter 230. The electrical rig shall simulate every significant aspect of the total electrical system. It shall be constructed at the design stage, and tests shall be conducted to demonstrate that the system proposed satisfies the design requirements of this chapter and the appropriate requirements of DEF STAN 05-123 Chapter 130.

11.2 The functioning of the system in the aircraft shall be demonstrated to the satisfaction of the Aircraft Project Director (see Chapter 1003).

11.3 Where an electrical power supply is software controlled the flight safety critical aspects of the software must be verified/validated to the satisfaction of the Aircraft Project Director.

12 CROSS REFERENCE

12.1 A number of requirements directly related to the electrical installation appear elsewhere in this publication and these are listed in the Alphabetical Index. In addition, certain general requirements apply when electrical actuation is chosen for special tasks and it is the designer's responsibility to ensure compliance with all such requirements.

13 ACCESSORIES AND COMPONENTS

13.1 See Leaflet 706/3.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	25/20	3456
2	-	3317
3	25/18	3302
4	25/19	3303
5	25/25	3632

TABLE 1

PRIMARY POWER SUPPLIES

(Characteristics shall be in accordance with BS 3G100 Part 3)

<table>
<tr><th>NOMINAL VOLTAGE AT BUS BAR</th><th>NOMINAL FREQUENCY (Hz)</th><th>TRANSMISSION SYSTEM</th><th>SWITCHING DISTRIBUTION CIRCUITS</th><th>PROTECTION OF DISTRIBUTION CIRCUITS</th></tr>
<tr><td>28V dc</td><td></td><td>SINGLE LINE AND NEGATIVE GROUND</td><td>IN POSITIVE LINE (SEE ALSO NOTE 3)</td><td>IN POSITIVE LINE</td></tr>
<tr><td>115/200V
3 PHASE ac</td><td>500
(CONSTANT)</td><td rowspan="2">THREE LINES - NEUTRAL POINTS

GROUNDED AT GENERATORS</td><td rowspan="2">IN ALL THREE LINES</td><td rowspan="2">IN ALL THREE LINES</td></tr>
<tr><td>115/200V
3 PHASE ac</td><td>400
(VARIABLE)</td></tr>
</table>

NOTES:

1 The only ground supplies available will be 28 V dc and 115/200 V ac 400 Hz.

2 In certain circuits it may be desirable to use negative switching e.g., use of solid state switches. A fail-safe circuit shall be achieved.

3 The 115 volt supply is angle phase derived from the primary 3 phase supply and connected between line and neutral; switching and protection to be in the line.

4 For turbo-propeller aircraft with electrical propeller anti-icing, the neutral for this circuit should be grounded through a high impedance.

5 When it is essential to use existing equipment requiring alternating current of different voltages and frequencies from those given in Table 1, the approval of the Aircraft Project Director is required.

6 Power conversion devices having an input supply as specified in Table 1 may be used to provide the supplies specified in Table 2 and shall be approved by the Aircraft Project Director.

TABLE 2

SECONDARY POWER SUPPLIES

AC	115V SINGLE PHASE 400 Hz 115V THREE PHASE 400 Hz 26V SINGLE PHASE 400 Hz 5V SINGLE PHASE 400 Hz
DC	270V dc 28V dc 14V dc 5V dc

LEAFLET 706/0

ELECTRICAL INSTALLATIONS

REFERENCE PAGE

Defence Standards

00-35	Environmental Handbook for Defence Materiel
00-40	Reliability and maintainability
00-41	Reliability and maintainability, MOD guide to practices and procedures
00-971	General specification for aircraft gas turbine engines
58-95	Electronic assemblies
59-3	Terminals electrical
59-4	Plugs and sockets, electrical
59-7	Relays, armature and relays, thermal
59-27	Precision instrument, rotating, servo-components
59-30	Resistors, fixed, of assessed quality
59-35	Part 0: Connectors electrical for dc and low frequency applications
59-36	Selection electrical and electronic components for use in defence equipment
59-40	Lugs, terminals, and terminal strips
59-41	Electromagnetic compatibility
59-42	Heat shrink solder sheaths
59-44	Capacitors of assessed quality, selection and procurement
59-45	Filter networks
59-51	Relays, electrical, of assessed quality
59-56	Plugs and sockets, electrical
59-61	Semiconductor devices
59-71	Crimped electrical connections for copper conductors
59-75	Switches of assessed quality
59-76	Transformers and inductors
59-96	Fuse links, electrical
59-97	Heat shrinkable insulation sleeving
59-100	use holders, carriers and bases electrical fuse (block and extractor post types)
61-3	Electrical wire and power distribution equipment
61-5 to 61-7	
61-9 and 61-10 ..	
61-12	
62-9	Lamp holders; lights, indicators, and lenses, indicator light, for use in equipment
62-10	Lamps, filament
66-7	Instruments; electrical indicating (sealed)
66-26	General requirements for aircraft instruments and displays

NOTE: BS 3G100 is gradually being incorporated into appropriate Defence Standards

British Standards

2G 100	..	..	General requirements for electrical equipment and indicating instruments for aircraft
3G 100	..	..	General requirements for equipment for use in aircraft
G 102 ..	..	..	General requirements for rotating electrical machinery
2G 124	..	..	Ac generators for aircraft
2G 127	..	..	Power and current transformers for use in aircraft electrical power supply systems
2G 132	..	..	Electromagnetically operated contractors
2G 134	..	..	Dc generators
G 142 ..	..	..	Heavy duty, electromagnetically operated, single-pole circuit breakers for extra low voltage dc systems in aircraft
2G 143	..	..	Ac and dc rotary and linear actuators for aircraft
2G 146	..	..	Dc motors for aircraft
2G 147	..	..	Ac motors for aircraft
4G 173	..	..	Connectors for ground electrical supplies for aircraft
G 174 ..	..	..	Invertors for secondary electrical supplies for aircraft
G 176 ..	..	..	Cartridge fuses for aircraft
5G 17 ..	..	..	Crimped joints for aircraft electrical cables and wires
G 179 ..	..	..	Hand operated circuit breakers for aircraft
G 180 ..	..	..	Permanent splicing of aircraft electrical cables
G 184 ..	..	..	Aluminium terminal ends and in-line connectors for hexagonal crimping to aircraft aluminium electric cables
G 194 ..	..	..	Lever-operated manual switches for aircraft
G 195 ..	..	..	Minyvin type electric cables for aircraft
2G 196	..	..	Static invertors for aircraft
3G 197	..	..	Stud-type terminal blocks for terminations on aircraft electric cables
3G 198	..	..	Sleeves for aircraft electric cables and equipment wires
2G 202	..	..	Specification for design and performance requirements for airframe - fit electrical connectors for dc and low frequency ac applications
G 203 ..	..	..	Unified screws with captive facing and locking washers
G 204 ..	..	..	Copper terminal ends for crimping to electric cables with copper conductors
5G 205	..	..	Secondary batteries for aircraft
G 206 ..	..	..	Fepsil type electrical cable with copper conductors
G 207 ..	..	..	Aircraft electrical circuit diagrams
G 208 ..	..	..	General requirements for electrical relays up to 10A rating
G 209 ..	..	..	Transformer rectifier units
2G 210	..	..	PTFE insulated equipment wires (with silver plated copper conductors)
G 212 ..	..	..	General requirements for aircraft electric cables
G 213 ..	..	..	Mechanically operated switches

G 216 ..	..	..	Hand operated, thermally compensated miniature, single and triple pole circuit breakers
G 217 ..	..	..	General requirements for cockpit lighting controllers
G 218 ..	..	..	Proximity switches for air-craft
2G 219	..	..	General requirements for ground support electrical supplies
2G 221	..	..	Minyvin type electric cables (metric units)
G 222 ..	..	..	Efglass type electric cables (metric units)
2G 223	..	..	Screen termination devices for aircraft electric cables
G 244 ..	..	..	Lever operated manual switched for aircraft
G 225 ..	..	..	Performance of environment-resistant terminal junction modules with removable crimp-type contacts
G 227 ..	..	..	Tersil type electric cables (metric units)
G 228 ..	..	..	Clearance and fixing dimensions for 2A and 3A, two and four pole sealed electromagnetic relays for aircraft
2G 229	..	..	Schedule for environment conditions and test procedures for airborne equipment
G 230 ..	..	..	General requirements for aircraft electrical cables (second series)
G 231 ..	..	..	Conditions for general purpose aircraft electrical cables and aerospace applications
G 232 ..	..	..	General requirements for general airframe or equipment interconnect use (135°C) wrapped insulation
G 233 ..	..	..	General requirements for general airframe or equipment interconnect use (135°C) extruded insulation
J 12	..	..	Pressure-sensitive adhesive identification
M 43 ..	..	..	Methods of zoning aircraft and referencing access doors and panels
M 44	..	..	Identification of aircraft servicing maintenance, general handling and safety - hazard points
4999	..	..	General requirements for rotating electrical machines
5000	..	..	Rotating electrical machines of particular types or for particular applications
9522	..	..	Rules for the preparation of detail specifications for circular electrical connectors of assessed quality for dc and low frequency ac applications

US Military Specifications

MIL-W-5088L .. Wiring, aerospace vehicle

Naval Weapons Specification

NWS 1006 Annex A

UK AEA Culham Laboratories Report

CLM-R-163 .. .. Recommended practice for lightning simulation and testing techniques for aircraft

LEAFLET 706/1

ELECTRICAL INSTALLATIONS

GENERAL RECOMMENDATIONS

1 INTRODUCTION

1.1 This Leaflet makes general recommendations concerning the design of electrical installations in aircraft.

2 SAFETY IN THE EVENT OF FAILURE

2.1 GENERAL

2.1.1 The requirements of Chapter 706, para 2.2 are aimed at ensuring that, in the event of cabin pressure failure, the most adverse combination of failures likely to occur in the electrical system will not prevent continued safe flight and landing with a probability of occurrence not greater than extremely unlikely. The circuits, components and installation of the electrical systems covered by these requirements shall be assessed accordingly and a fault analysis prepared in accordance with Chapter 117 (to be issued) requirements.

2.1.2 Certain circuits are clearly vital to the safety of the aircraft (e.g. powered flying controls) but this may not be so obvious with other circuits. Examples of the latter are given below but it is emphasised that the list is not comprehensive and the designers should review the concept of a total system function in identifying potentially critical aspects of a particular electrical circuit:

a) flying control trimmer systems (see Chapter 604)

b) undercarriages

c) explosive bolts

d) auto-stabilizer

e) wing folding

f) fuel systems (see Chapter 702, para 9.2.4)

g) engine control

h) power supplies to essential navigation equipment and its displays

j) power supplies to terrain following guidance systems.

2.1.3 Electrical circuits which provide the primary initiation of vital systems should be wired in such a manner that no single failure of the wiring system can cause inadvertent operation of any circuit. Particular attention should be paid to ground connections in this respect. Multiple primary initiation circuits should be separately grounded direct to airframe.

2.1.4 Particular attention should be given to avoid the possibility of common mode faults occurring. All circuits should be analysed to identify the possibility of dormant faults occurring and the design or operating drills should provide for these accordingly. (An unnatural operating drill would not be acceptable).

2.2 WARNING OF FAILURE

2.2.1 Only warnings of failure for which there is an automatic or direct corrective action should be provided. Warnings may be cancelled automatically when corrective action is taken, but with regard to critical systems it is advantageous to annunciate a warning as a reminder that an abnormal condition exists.

2.2.2 Care should be taken to ensure that warning lights, indicator lights or annunciators show clearly and precisely the particular function that has failed. Whilst in general, systems will be designed to "fail-safe", the possibility of false indications should not occur.

2.2.3 Lamps, filament used as indicators should be replaceable without removing supports, panels or other components. Provision of individual press-to test or general lamp test circuits is recommended.

2.3 ALL ENGINE FAILURE CASE

2.3.1 The requirement of Chapter 100, para 9.2 calls for the ability to make a descent and emergency landing with an extended glide in the event of all engines failing. Even where powered flying controls do not depend on electrical operation, the designer should consider what other electrical services may be required during the descent and ensure that adequate stored or auxiliary electrical power is available beyond that from windmilling engines.

2.3.2 Where the ability of the aircraft to meet this requirement depends on stored energy, the capacity of the battery must be such that it can meet standby requirements and still retain sufficient power to operate vital services with a reasonable safety margin (Chapter 706, para 7.1.3).

3 CIRCUITS

3.1 BONDING AND GROUND STRAPS

3.1.1 Bonding straps fitted in accordance with Chapter 708 may be used to meet the requirements of Chapter 706, para 5.5.1 provided they are of sufficient cross sectional area to carry the full electrical load likely to be imposed on them.

3.1.2 The bonding, straps should be configured so as to minimise the risk of additional spark discharges.

3.2 REDUCTION OF VULNERABILITY (see also Chapter 112)

3.2.1 When the installation necessitates all, or the majority of, the protective devices being located in one compartment, they should be split into at least two separate groups, with the maximum, available space being provided between the groups. Groups of protective devices on each side of the fuselage are preferred, each group having its own feeder cable. The added protection afforded by armour plate may be considered. Components of duplicated systems should not be located on a common radial path emanating from potential damage centres e.g., high energy rotors, potential weapon impact points, (see Chapter 112) and should not be supplied from a common distribution bus bar.

3.2.2 Wiring should be arranged so that cables are in two or more groups with a reasonable space between them or preferably are divided on each side of the fuselage. Cables should leave panels in at least two groups spaced apart.

3.3 PROTECTION

3.3.1 When circuit breakers or fuses are used in series it is important to ensure that their characteristics are such that a fault in a sub-circuit will not cause the main feeder protection to operate. Supply and distribution cables should be located so that damage is unlikely to short circuit the protective device, e.g., fire in a cable loom.

4 WIRES CABLES AND CONNECTORS

4.1 WIRE AND CABLE INSTALLATION

4.1.1 When additional protection is not required open wire and cable looms may be installed. Otherwise protection of wires and cables may be provided by running them in ducts which should be:

a) large enough to allow additions to the number of wires and cables accommodated and to allow any single wire or cable to be withdrawn (see Chapter 706, para 5.4.1),

b) adequately drained,

c) adequately flared or bushed with insulating material at the points of entry and exit,

d) such that the risk of damage to wires and cables being drawn into them is minimized,

e) fitted, when necessary, with easily detachable or hinged covers.

4.1.2 Wire and cable runs should be routed so as to avoid areas where heat is likely to be encountered. If it is unavoidable wire and cable assemblies likely to be subjected to high temperatures, i.e., those passing through bays adjacent to fire zones, those adjacent to hot air ducts or adjacent to the aircraft skin should be assembled from components having suitable high temperature duty rating. Where there is a danger of direct air impingement due to leakage protective guards should be considered.

4.1.3 Wire and cable assemblies should be adequately supported or protected to prevent chafing against adjacent objects under all conditions of vibration and loading of the aircraft. Wire and cable insulation should not be allowed to touch the aircraft structure.

4.1.4 Wires and cables should be securely clamped to relieve terminals and connections of mechanical load. Clamps should not affect the mechanical or electrical properties of the wire and/or cable being secured.

4.1.5 On aircraft with folding components, the wires and cables should be suitably protected from damage where they pass over hinged joints. The components should be able to fold without the disconnection of wires and cables being necessary.

4.1.6 To avoid replacement difficulties, wires and cables should not be twisted together, except where it is essential to minimise interference, e.g., 3 phase ac cables, armament feeders, Fly-By-Wire(FBW) and Full Authority Digital Engine Control (FADEC).

4.1.7 Wire and cable looms should be positioned so that they are unlikely to be used as hand or foot holds.

4.1.8 Where positive location of wire and cable terminations is not provided within an item of equipment, adjacent wires and cables should be secured together to prevent movement.

4.1.9 To allow remaking of connections, wire and cable lengths should allow for two remakes. If this is impractical in congested areas the advice of the Aircraft Project Director should be sought.

4.1.10 Sufficient slack should be left at the ends of the wires and cables to allow displacement of components to which the wires and cables are attached.

4.1.11 Adequate wire and cable lengths should be provided to allow full movement of all moving electrical apparatus and the wires and cables should be suitably run and supported to prevent damage.

4.1.12 For armament electrical wires and cables see Chapter 710.

4.1.13 Bend radii of installed wiring should not be less than 10 times the outside diameter of the largest single wire or 6 times the overall diameter of the cable loom whichever is the larger.

4.1.14 Use of Wire and Cable: The use of wire and cable containing polyimide in various parts of an Aircraft is to be as follows:

a) General Purpose Wire and Cable: Electrical interconnection wire and cable with a specified minimum temperature rating of -65°C to +135°C is to be used in general purpose areas. Hybrid wires that utilize polyimide in their construction are acceptable for use in these areas. There are no restrictions on the use of wires or cables compliant with DEF STAN 61-12 Part 33 or an equivalent Specification demonstrated to fulfill these requirements as a minimum, noting that these wires and cables may contain an element of polyimide. See also Chapter 706 para 6.6.

b) High Temperature Wire and Cable: Electrical interconnection wire and cable with a specified minimum temperature rating of -65° to +260°C is to be used in high temperature areas. Hybrid wires that utilize polyimide in their construction are most suitable for use in these areas. There are no restrictions on the use of wires or cables compliant with DEF STAN 61-12 Part 33 or an equivalent Specification demonstrated to fulfill these requirements as a minimum, noting that these wires and cables may contain an element of polyimide. See also Chapter 706 para 6.6.

c) Severe Wind and Moisture Prone (SWAMP) Areas: Typical SWAMP areas include undercarriage bays, flap, slat, airbrake and wing fold areas. This environment is such that if a wire utilizing polyimide in its construction is selected for use, then greater justification for its use will be required by the MOD Project Director.

d) Pylons and Launchers: Pylons and launchers are areas of high levels of maintenance activity and demanding environmental conditions, including buffeting. Wire and cable used in these areas require a high degree of flexibility and external durability. Therefore, wire and cable used in these areas should only utlilize a top coat of known durability in their construction. Wires containing polyimide in their construction are **NOT** to be used in these areas.

4.2 CONNECTORS

4.2.1 MULTIWAY CONNECTORS

a) General guidance on the selection of a connector for a particular application with regard to environmental and operating requirements is given in DEF STAN 59-35 (Part 0), para 5.

b) Connectors to be reached through an access panel should be positioned to allow easy disengagement and removal. To facilitate servicing, sufficient lengths of wires or cable should be provided to allow the free connector to reach the panel opening.

c) Crimped terminations using approved tooling should be used except where hermetically sealed connectors with solder terminations are required.

4.2.2 TERMINAL POST

a) At a terminal post where two or more terminations are used they should normally be mounted in pairs back to back and where more than one pair of terminations is fitted to one side of a terminal, spacing washers should be inserted between the pairs.

b) In the case of a fully utilized single row double entry terminal the terminal ends should be interleaved from each side of each terminal to obviate the need for spacing washers.

c) Terminals reached through an access panel should face outward to facilitate manipulation.

d) Crimped terminations using approved tooling should be used.

4.3 LOCKING

4.3.1 Wherever possible stud terminals and nuts with lock washers or metallic stiffnuts should be used. The use of screws with loose washers is not acceptable.

4.3.2 Where tapped inserts or clinch nuts are used, they should be of an approved type with an integral locking.

4.4 INTERFERENCE PREVENTION (see also Chapter 708)

4.4.1 Guidelines for Electromagnetic Compatibility are given in AvP 118 and should be adopted wherever possible in all cable and aerial installations. Problem areas should be resolved with the Aircraft Project Director.

4.4.2 Composite structures do not provide the inherent screening characteristics of metal fabrications.

5 BATTERIES

5.1 Batteries should not be fitted in engine nacelles.

5.2 When calculating the capacity of the batteries, account should be taken of the voltage to which the batteries will be charged in flight.

6 PROTECTION FROM MOISTURE

6.1 Cables connecting to equipment should, wherever possible, be arranged to run downwards from the item of equipment in drip loops. The ends of any spare cables or cores should be sealed to prevent the ingress of moisture.

6.2 Where apparatus is exposed and water can impinge upon it under any conditions, the apparatus should be suitably weatherproofed and the compartment provided with adequate drains.

6.3 Particular attention should be given to the type of protection provided on naval and other aircraft operating over the sea to prevent deterioration due to driven sea spray. (see DEF STAN 00-35 and BS 3G100 or superseding Defence Standard).

6.4 Junction boxes, distribution boxes, fuse boxes, etc., should be protected against deterioration due to ingress of moisture. Adequate draining and ventilation should be provided.

7 POWER INTERRUPTION

7.1 Where equipment is transferred from one power source to another the functioning of the equipment should not be adversely affected during the transfer period. The transfer period is defined as the time of departure from a steady state voltage and frequency to the time of recovery to a similar steady state voltage and frequency.

7.2 Where a 'no break' supply is required the transfer should be designed with a minimum interrupt.

8 ELECTRICAL SPECIFICATION

8.1 Attention is drawn to the requirements of DEF STAN 05-123 which require the Aircraft Contractor to agree with his sub-contractors for each particular item of electrical equipment a specification to the standard laid down in DEF STAN 05-123.

9 TEST FACILITIES

9.1 The purpose of the test facilities required by Chapter 706, para 11 is to:

a) check the voltage and frequency

b) check the protection circuits which do not function under normal system operation

c) locate defective constituent parts of the electrical system, the repair of which is by replacement at First Line Servicing, e.g., generators, alternators, contractors, etc.

d) check the correct function of the system after rectification of defects.

LEAFLET 706/2

ELECTRICAL INSTALLATIONS

POWER SUPPLIES

1 INTRODUCTION

1.1 This Leaflet gives further information on the standard power supplies and their applications to the installation.

2 CHOICE OF SUPPLY

2.1 BS 3G100, Part 3 details characteristics of three forms of electrical power supply for aircraft, namely constant frequency alternating current, variable frequency alternating current and direct current. The supply most suitable for the consumer item being fitted should be installed.

NOTE: BS 3G100 is gradually being incorporated into appropriate Defence Standards.

2.2 It is possible that more than one power system deriving power from engine driven generators may be fitted. It is possible that distribution to services may be provided at differing voltages derived through conversion devices, e.g., instrument supplies, emergency lighting supplies.

2.3 Choice of systems will depend, therefore, on :-

a) total power requirements

b) the nature of loads

c) the power to be provided by conversion devices

d) the alternative standby power required

e) the stored power required

2.4 Separate systems should not be interconnected, but circuits may be arranged to allow for one system to provide a standby power source for another by using a suitable conversion device.

2.5 The choice of power supply, i.e., nature of supply, capacity and means of distribution, should be approved by the Aircraft Project Director.

3 DESIGN OF MAIN SYSTEM

3.1 During the initial design stages of an aircraft the maximum continuous electrical load is not definitely known. For instance, there may be new equipment developed before the final acceptance date and this may be added as a requirement.

3.2 In addition, Chapter 706, para 3.2.2 required a reserve of power at the time of the Final Conference in order to allow for further requirement of the aircraft during its service life.

3.3 In order to cover these two points it has been found by experience that 100% spare power should be allowed at the initial design stage based on the maximum continuous demand under the worst conditions of flight. The maximum continuous demand should not include high current peaks of short duration such as starting current for heavy motors. In calculating spare power, certain large loads, e.g., de-icing, need not be factored. When application of the requirement to the overall electrical power supply could lead to serious penalty the designer should consult the Aircraft Project Director.

3.4 In the case of an aircraft having four generators, the load at the time of the Final Conference plus the reserve of power required by Chapter 706, para 3.2.2 should be met by any three generators. It should be possible to operate the aircraft after the loss of any two generators.

3.5 The main system should also be capable of accepting the maximum peak load and still maintain the required regulation. If this should mean larger capacity than arrived at above, then the larger capacity should befitted .

3.6 As some items of equipment may not provide their specified performance within the trip bands of voltage and frequency, deterioration of performance should be indicated by the equipment.

3.7 Where desirable, provision should be made for running alternative generator(s) while the aircraft is on the ground without resorting to the use of the aircraft engines. The same system may be used as a standby power system when the aircraft is airborne.

3.8 Where invertors are used as the main source of power for certain services, suitable trip devices should be provided in order to protect consumer items of equipment against damage if the characteristics of the output exceed the permitted limits.

3.9 Protection devices should be used wherever possible to localise possible faults that could occur within a main generator and distribution system.

3.10 Generator controls end indications should be logically grouped together at the cockpit or other crew station and should be so positioned that the controls and indications applicable to a particular power source are readily discernible.

3.11 The equipment and protection devices associated with separate power sources should be separately housed. Where possible feeder cables from separate systems should be segregated to reduce to a minimum the possibility of a common failure incapacitating more than one power source.

3.12 Distribution systems associated with a particular power source should not be installed in an assembly common to distribution systems associated with another power source.

3.13 Where more than one storage battery is required to achieve the desired total storage capacity these should be engineered as separate power sources and should not be located immediately adjacent to one another.

LEAFLET 706/3

ELECTRICAL INSTALLATIONS

ACCESSORIES AND COMPONENTS

1 INTRODUCTION

1.1 This Leaflet makes general recommendations concerning the manner in which electrical accessories are applied and installed in various systems.

1.2 All accessories and components shall comply with BS 3G100.

2 DATA

2.1 General advice regarding the application of accessories and components is given in the documents listed in Table 1.

TABLE 1

ITEM	BS	DEF STAN	OTHER
Actuators		61-14	
Alternators	G124		
Batteries	G205	61-9	
Capacitors		59-44	
Circuit Breakers	(G142 (G179 (G216		
Connectors	(G713 (G180 (G184 (G202, 204 (G220 (G223 (G225	59-35 59-42 59-71	
Contactors	G172		
Cable Accessories	G198		
Electronic Components		58-95	
Fuses	G176	(59-96 (59-100	
Filters		59-45	
Generators	G134		
Heaters			
Indicators and Displays		66-26	
Indicators	(G133 (G191		
Invertors	(G174 (G196		
Lamp Units	G193	62-9	

TABLE 1 (continued)

ITEM	BS	DEF STAN	OTHER
Microswitches	9562	59-75	EL2123
Meters	(G146 (G147	61-1	
Proximity Devices	G218		
Relays	G208 G228	(59-7 (59-51	
Resistors		(59-30	
Rectifiers	G209		
Semi Conductors		(59-61 (59-62	
Systems	G183	59-27	
Switches	(G194 (G213 (G224	59-75	
Terminals	G197	(59-3 (59-40	
Transformers	(G127 (G207	59-76	
Wires and cables	(G212 (G230	61-12	

3 APPLICATIONS

3.1 Component and Accessories manufacturers' recommendations should be followed.

3.2 All mounting points of an accessory or component should be used to achieve a secure attachment to the airframe unless the unit is configured to permit alternative mountings or alternative orientation.

3.3 Components should be located so that electrical connections may be completed after installation. This particularly applies to ease of access to the terminal connections.

3.4 After completing the electrical connections exposed electrical conductors should be at the underside of the component; otherwise means should be provided to protect the exposed parts from falling debris etc.

3.5 Accessories and components should be purposefully orientated to gain the best advantages with regard to function, access, protection and (if appropriate) operation and indication.

3.6 Ensure that adequate grounding is provided when required. (see also Chapter 708).

3.7 Electrical cables to a component should be adequately supported and should be installed in such a way that interchange of connections is not readily possible when the unit is removed for maintenance.

CHAPTER 707

RADIO AND RADAR INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this Chapter apply unless specified otherwise, to all radio and radar installations forming part of the operational equipment of a rotorcraft.

1.2 A RADIO INSTALLATION comprises all those items of equipment necessary to communicate or receive information (e.g., speech, navigational data etc)., via the medium of electromagnetically radiated waves to/from a similar system with which it has no direct physical contact.

1.3 A RADAR INSTALLATION comprises all those items of equipment necessary to radiate electromagnetic waves, and then to utilise the reflected, or automatically re-transmitted waves, to gain information concerning distant objects (e.g., range and relative position, topographical features, meteorological conditions etc).

1.4 Installations shall be so planned that when utilising the appropriate rotorcraft power supplies and (even when subjected to the most severe rotorcraft environmental conditions envisaged), they will achieve optimum specified performance when used alone or when integrated into a larger system of installations.

1.5 Installations shall have a proof factor of not less than 1.0 and an ultimate factor of not less than 1.33 on the loads arising in all conditions of flight specified for the rotorcraft as a whole. (For the crash case see Chapter 307).

2 AERIAL DESIGN

2.1 Aerials shall be designed having due regard to:

2.1.1 Electrical Performance.

(i) Signal polarisation.

(ii) Optimised radiation efficiency.

(iii) The required radiation pattern coverage.

(iv) Optimum energy transfer (i.e. , electrical matching) to associated rf feeders and equipment assemblies.

(v) The effect on the above features, of any fitment required to affix the item to the airframe.

(vi) The effect, on the electrical performance, of any additional protective dielectric cover or housing.

(vii) Where individual aerials are used in pairs for 'homing' purposes they shall be accurately electrically matched, to close tolerances, over the whole of the frequency band in which they are to be used. Likewise, their connecting rf feeders are to be accurately matched before assembly on to the airframe. Consideration shall be given to out of band attenuation and cross polarisation performance.

2.1.2 Mechanical Performance With Respect to Rotorcraft Permitted Aerodynamic Limitations.

(i) Minimised mass, and optimised mechanical integrity consistent with the intended application i.e.,

(a) to withstand maximum aerodynamic loads envisaged at the proposed airframe location

(b) to have freedom from flutter and to withstand loads arising from the oscillatory behaviour of fixed or trailing wire elements. For whip aerials consideration shall be given to the need for flexible mountings.

(c) to withstand damage due to vibration.

(d) to suffer minimal damage due to impact e.g., through birdstrike, (see Chapter 206), rough terrain etc.

(e) the physical balance of moving components e.g., scanner, under the influence of rotorcraft manoeuvre.

(ii) Environmental effects including:

(a) ice accretion

(b) erosion through rain, hail, rotorcraft fluids.

3 AERIAL LOCATION ON AN AIRFRAME

3.1 Aerials shall be located having due regard to:

3.1.1 Electrical Performance.

(i) Signal polarisation, (of special note where an aerial is mounted on a movable surface e.g., nosewheel door, cargo ramp etc).

(ii) Required radiation pattern coverage, i.e., a site with a 'clear field of view' from which the radiated signal is not:

(a) Obstructed by intervening airframe structure, or projections therefrom of significant electrical size, i.e., producing 'holes' in the patterns.

(b) Distorted by electromagnetic wave re-radiation/ reflection from 'nearby' airframe components or projecting objects e.g., rotors, other aerials.

(c) Obstructed/distorted by the movement of airframe components e.g., undercarriage, cargo ramp etc.

(d) Obstructed/distorted by carriage and/or disposal of external stores e.g., weapons, fuel tanks etc.

(iii) Electromagnetic interference (EMI) from other electronic sources including rf feeder cables, and power supplies within the rotorcraft or the radiation/induction of EMI onto other systems.

(iv) Where communications with other aircraft are envisaged, minimisation of any deleterious effects due to multipath signal reflections from terrestrial sources.

3.1.2 Mechanical Constraints

(i) Mechanically deployed aerials (e.g., trailing wires, long 'whips' under the fuselage, and those in retractable housings etc)., shall be 'fail-safe' in order that no damage to the rotorcraft shall occur during any phase of deployment or retraction, or when the rotorcraft is taking off or landing. (see also paras 4.1.7 and 4.1.8).

(ii) No damage shall arise from the movement of adjacent airframe components.

(iii) Where the aerial is fixed to a movable airframe component, its associated rf feeder, and power supply leads where supplied, shall not be subject to undue stress during such movement.

(iv) Vulnerability to battle damage shall be minimised (see Chapter 112).

(v) In-flight detachment of the aerial through mechanical failure shall not cause significant damage to the rotorcraft, particularly the rotors.

(vi) In-flight separation of accreted ice shall not cause damage to engines, or significant damage to the airframe or rotors.

(vii) Freedom from contamination by aircraft fluids and water.

(viii) Freedom from excessive heating (e.g., engine efflux).

(ix) No hazard shall be presented to maintenance or ground handling personnel. Where such a possibility exists, adequate warnings shall be provided.

(x) 'Hands/Feet Off' warnings shall be prominently displayed wherever an aerial is likely to, present a hand or foothold to aircrew and maintenance or ground handling personnel.

(xi) Access for maintenance purposes (i.e, fitting or removal of the aerial, attention to rf and power connections) shall be as simple as possible, to minimise rotorcraft 'down' times.

4 AERIAL INSTALLATION ON AN AIRFRAME

4.1 The following requirements shall apply when installing aerials on to an airframe:

4.1.1 A homogeneous conductive path, of dc resistance less than 5 milliohms, shall be maintained between the aerial mounting flange and that part of the airframe which forms an extension of the 'earth plane'. Where there is a mechanical discrepancy between the aerial mounting face and the rotorcraft structure, the aerial shall be assembled on to the airframe using a profiled packing piece interposed between aerial mounting and airframe. Such packing shall have a minimal effect upon the required radiation performance of the aerial (see para 3.1.1). The packing piece shall be of material in compliance with BS 3G100 Part 2 Section 3 Sub Section 3.2 Table 2, and shall not increase the dc resistance between aerial and airframe earth to more than 5 milliohms.

4.1.2 Dissimilar metals in contact can cause corrosion due to galvanic action The requirements of Chapter 407 are to be observed. See Leaflet 407/3 for additional information on galvanic action.

4.1.3 Where possible, 'wet assembly' techniques shall be used, including release agents where necessary. This is essential to prevent moisture or fluid ingress from both inside and outside the rotorcraft from causing contamination of the aerial-to-structure interface, and to facilitate aerial removal (see para 3.1.2(xi)). However where the aerial requires a large earth plane, mating surfaces of metal skin joints may be treated with a suitable conductive protective treatment. These joints shall receive suitable treatment after assembly.

4.1.4 An approved sealant shall be used around the outside periphery of the aerial junction to the airframe and, in the case of lower fuselage aerials, to the inside junction also. The seal may be achieved with a gasket complying with the requirements of para 4.1.1.

4.1.5 Fixed wire aerials shall have a device to maintain a constant tension and a weak link to ensure failure can be anticipated at a known point. The design shall consider the effect of having a free wire aerial as defined by the position of this weak link (see also para 3.1.2 (i)).

4.1.6 Adequate drainage shall be provided in the vicinity of a belly mounted aerial so that accumulation of water or other liquids may be reduced to a minimum. Such aerials shall have liquid-proofed feeder cable mating connectors to prevent ingress of moisture.

4.1.7 To achieve necessary clearances or reduce drag some aerial configurations may, according to application, be installed on a retractable mounting. Such a mounting shall be provided with the necessary interlocks and indications to ensure the assembly may only be extended, operated and retracted at the correct phase of flight.

4.1.8 No single failure shall prevent the operation, retraction or extension of an aerial system where such failure could affect the safe handling of the rotorcraft.

4.1.9 Where an aerial is shared by two or more transmitters, appropriate interlocks shall be provided to prevent inadvertent simultaneous transmissions.

5 RADOMES AND AERIALS FAIRINGS

5.1 MECHANICAL REQUIREMENTS

5.1.1 Radome shape will be constrained by the aerial application, and ideally will offer minimum aerodynamic drag. Where a radome alters the original airframe profile, the significance of its aerodynamic effects shall be assessed through re-test of the rotorcraft's handling and performance characteristics.

5.1.2 The radome shall be structurally sound, capable of withstanding the aerodynamic loads and be resistant to erosion by rain, hail, and to damage from debris when landing on 'rough' terrain.

5.1.3 Where a radome fairing is located at the extremity of the airframe e.g., nose, tail, sponsons, it shall be fitted with lightning diverter strips either on the inside or outside of the radome. These must be electrically bonded to airframe earth (dc resistance of not more than 50 milliohms). The configuration, arrangement and material for the lightning diverter strips shall be adequate to provide the necessary protection against a direct lightning attachment and to conduct the ensuing lightning current to the airframe without damage to the radome.

5.2 ELECTRICAL REQUIREMENTS

5.2.1 The electrical performance of the radome contributes to the performance of a total installed system. Hence the radome shall be designed to provide maximum transmissivity, over the frequency range concerned, of radiation from the enclosed aerial.

5.2.2 In addition to structural integrity the material used in the construction and geometry of the radome shall provide a clear undistorted aerial view, with minimum reflection over the required arcs of radio wave transmission.

5.2.3 Where a mechanically scanning aerial causes high energy transmissions either as main beam or sidelobe to be directed inboard at the airframe or other equipment then radar absorbent material, appropriate to the frequency band in question, shall be used to protect those parts and to minimise reflection.

6 RADIO AND RADAR EQUIPMENTS

6.1 LOCATION

6.1.1 Each unit shall be suitable for the environment in which it is located and shall be capable of normal operation to the maximum altitude to which the rotorcraft is designed.

6.1.2 Additional cooling or heating shall be provided where the installation or the radio manufacturer's specification dictates.

6.1.3 Units generating electromagnetic interference shall be located so as to minimise the effect on other systems. Similarly, interconnections between units that may generate electromagnetic interference shall be installed so as to minimise the effect. (see DEF STAN 59-412).

6.1.4 All equipment and cables shall be positioned where practical to avoid damage by acid, water, oil, fuel or other fluid or by the air crew or ground crew in the normal course of their duties.

6.1.5 Cable connections to units shall not run adjacent to the rotorcraft's main power distribution feeders or other likely sources of interference. Where systems are duplicated the cable connections of the separate systems shall be kept well apart.

6.1.6 Control panels, indicators and selectors shall be located at crew stations to comply with the provisions and requirements of Chapter 107.

6.1.7 Units shall be located to minimise the effects of power supply voltage reduction and rf losses through excessively long cables/feeders.

6.2 MOUNTING

6.2.1 Where the vibration limits applicable to a radio or radar unit would not be satisfied by direct attachment to the airframe means of isolating the vibration shall be employed. However adequate electrical bonding, resistance not greater than 5 milliohms, shall be provided between the equipment and airframe earth.

6.2.2 Equipment mounted in quick release trays shall not be so oriented that the equipments, if not properly secured, would be likely to impact an occupant or other equipments in the event of a crash landing. (see Chapter 307).

6.2.3 Units, connectors, complete with their interconnections, feeders, etc., shall be installed as far as practicable so that they can be quickly and easily replaced when the rotorcraft is on the ground (see also Chapter 802) and also without the need to first remove other equipment.

6.2.4 Units shall not be mounted on a rotorcraft component which is itself subject to a different periodic removal unless that component forms part of the same radio or radar installation.

6.2.5 The removal of one component shall not loosen another component.

6.3 POWER SUPPLIES

6.3.1 Power supplies for rotorcraft installed radio and radar equipment shall conform to those specified in Chapter 706. Details of distribution, capacity, alternative or standby systems and special voltages (if applicable) shall be agreed with the Rotorcraft Project Director.

6.3.2 The distribution system for radio and radar power supplies and for equipment controlling radio and radar outputs shall be designed to minimise interruption or loss of service, and loss of facility, (where duplicated equipments are fitted) in the event of a failure occurring at a power source or in the distribution system itself.

6.3.3 Equipments requiring a continuous power supply shall be provided with a standby battery source, and a purpose converter if necessary.

6.3.4 Equipments sensitive to phase alignment shall have a suitable power supply.

6.3.5 Where alternative or standby power supplies are desirable the supply shall have the same integrity as the main system. The capacity of an alternative or standby power supply shall be adequate for the intended operation of the rotorcraft.

6.4 INTERCONNECTION

6.4.1 The installation of interconnecting electrical wires and cables shall be in accordance with Chapter 706.

6.4.2 Where necessary, high voltage cables shall be provided with protection, in addition to the basic covering, to guard against mechanical damage. Where practicable such cables shall be routed to provide adequate clearance from sharp angles or corners of equipment and structure.

6.4.3 Cable connections to units, plug boards and bulkhead plugs shall be accessible, clearly identified and not capable of cross connection.

6.4.4 As a design aim cables shall be routed to avoid areas of elevated temperatures but where such routing is unavoidable, e.g., in engine bays and locations close to exhaust pipes, cables and connectors shall be constructed from components and materials having a suitable high temperature duty rating.

6.4.5 Transmitter control circuits shall be interlocked to ensure that incompatible transmissions cannot operate simultaneously. Also where transmissions may adversely affect the operation of a working receiver, the operating crew shall be provided with a label or warning indication that the system is temporarily degraded during the period of transmission.

6.4.6 Coaxial feeder cables require special attention to ensure that the dielectric is not deformed or crushed by the installation of retaining devices, e.g., clips, straps etc., and that the cable is not subjected in the installation or during installation to bends having a radius less than that specified as the minimum bend radius.

6.5 HIGH TENSION VOLTAGES

6.5.1 Installations operating at high voltages shall be adequately insulated.

6.5.2 Where high voltage insulation is achieved by virtue of crew compartment or equipment pressurisation, means to isolate the power in the event of loss of pressure shall be provided. Equipment requiring such protection shall not be installed at the flight deck.

6.6 ELECTROMAGNETIC INTERFERENCE

6.6.1 The level of possible electromagnetic interference from a unit or its associated wiring shall be compatible with other equipments fitted and those likely to be fitted and shall conform to the radiated emission limits specified in BS 3G100, or as agreed with the Rotorcraft Project Director (see also DEF STAN 59-41).

6.6.2 An electromagnetic compatibility test schedule shall be prepared and agreed with the Rotorcraft Project Director.

6.7 TEMPEST

6.7.1 Requirements for Tempest clearance shall be considered in conjunction with the Rotorcraft Project Director.

7 RADIO AND RADAR SYSTEM CONTROL

7.1 Crew members shall be provided with the necessary control of equipments and systems under their charge. The status of this control shall be positively and clearly confirmed to the crew members concerned in an unambiguous manner.

7.2 Conflicting systems and operations shall be provided with interlocks to the satisfaction of the Rotorcraft Project Director.

7.3 Elementary means of aircrew intercommunications, radio communications and radio or radar navigation shall be maintained in all situations, including a total generated power supply failure.

8 STATIC ELECTRICAL CHARGES

8.1 Adequate means for the dispersal of static electrical charges shall be provided.

9 TESTING

9.1 Flight and ground testing shall be in accordance with the Rotorcraft Specification or as agreed with the Rotorcraft Project Director.

10 SERVICING

10.1 Within any limitations which may be imposed by technical consideration and operational requirements of the equipment, radio and radar units shall be so installed that the servicing requirements of Part 8 are met.

10.2 When an electrical and/or air supply is required for ground testing a radio or radar installation, the connection for this shall be readily accessible and shall not necessarily require the ground testing equipment to be taken into the rotorcraft.

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RADIO AND RADAR INSTALLATIONS

THE INSTALLATION OF AIRBORNE SPEECH COMMUNICATIONS SYSTEMS

1 INTRODUCTION

1.1 This Code of Practice sets out the principles recommended for application in installing airborne speech communications systems into military rotorcraft.

1.2 It is intended for use by the various Service agencies and rotorcraft contractors responsible for planning and designing the initial installation and any subsequent modifications, with the object of ensuring that each complete system operates as nearly as possible at its full capability.

1.3 It is strongly recommended that the provisions of the Code should be adopted by the installation designer wherever applicable; where a contractor proposes a departure from the recommended practice, the appropriate Rotorcraft Project Director should be consulted on alternative ways of meeting the objectives.

1.4 This document supplements but does not supersede the current series of publications relating to the installation of airborne electronic equipment and all mandatory requirements appearing in those publications must be met.

2 GENERAL

2.1 Speech communications systems are essential to the operation of military rotorcraft yet the rotorcraft environment is hostile to the efficient working of such systems. A detailed assessment during 1975 of a number of airborne speech communications installations showed that performance often fell far short of that achieved under the ideal conditions of a laboratory test bench. In most cases investigated, poor performance under flying conditions was found to be directly attributable to insufficient care in the planning and layout of the installation and to unsatisfactory integration of the individual equipment items.

2.2 The main problems are the degradation of speech signals by distortion and crosstalk and high levels of electrical noise which not only cause nuisance to the listener when endured for long periods but also affect the intelligibility of speech signals.

2.3 Distortion and cross talk frequently result from the presence of excessive signal levels at the inputs to the CCS amplifiers, which often have inadequate dynamic range. (CCS: Communications Control System - That section of the installation responsible for distributing and routing all baseband signals to the various crew stations and providing crew members with the facilities required to select, control and operate the available intercom and communications services).

2.4 The effects of the high level of acoustic noise existing in some rotorcraft environments can, at present, be reduced only by the use of noise cancelling microphones and by the provision of microphone switches which enable the system to operate with as few live microphones as possible; the reduction achieved by these means is limited.

2.5 Noise of electrical origin enters the installation at vulnerable points such as long and inadequately screened sensitive audio wiring, ground loops and microphone transformers located in areas affected by high level electro-magnetic fields.

2.6 No common standard of design and performance has yet been adopted for individual units of communications equipment and input/output signal levels and impedances vary widely, particularly between units obtained from different manufacturers.

3 CRITERIA

3.1 GENERAL

3.1.1 The standard of engineering must be such that the process of installing the system in a rotorcraft does not result in an unacceptable degradation of its performance when compared with that obtained on a test bench under ideal conditions. The installation is considered acceptable if the following requirements are met:

(i) Coupling of all types between the audio frequency signal lines in the inter-unit wiring, as installed in the rotorcraft, should be sufficiently low to reduce line-to-line crosstalk to better than -120 dB. (This figure assumes that all lines are correctly terminated and does not include any crosstalk occurring within the equipment items).

(ii) The frequency response of each complete audio channel should be within ± 1 dB of the amplitude/frequency characteristic specified for the basic system between 50 Hz and 7 kHz.

(iii) The attenuation of the transmitted signal due to losses in the cables and connectors for each complete signal path should not exceed 1 dB.

(iv) The total noise of electrical origin produced at each headset earpiece, under all operating conditions and with all gain controls set to maximum, should not exceed a sound pressure level of 60 dB above 20 micropascals. This level does not include any electrical noise inherent in the audio signals at the output terminals of the various radio receivers of local noise transmitted acoustically through the headgear.

3.2 SOUND PRESSURE LEVELS

3.2.1 The electrical level corresponding to a particular sound pressure level, in terms of millivolts rms developed across the telephone terminals, depends upon the characteristics of the particular type of headset used. The relevant factors are:

(i) the sensitivity of the telephone insert(s). This is usually quoted in dB sound pressure level above 20 micropascals per milliwatt of applied power, eg., 85 dB/mW,

(ii) the impedance of the telephone insert(s),

(iii) if two inserts are used, whether they are series or parallel connected,

(iv) if a single insert is used, the attenuation caused by the acoustic coupling tubes,

(v) the attenuation due to any built-in attenuator network.

3.2.2 The relationship between sound pressure level and the corresponding electrical levels for particular headsets is demonstrated in the following examples:

(i) Example 1

A Mark 2/3 flying helmet, using a single telephone insert coupled to the ear pads by acoustic tubes which attenuate the sound pressure level by 10 dB.

Insert impedance	:	300 ohms nominal at 1 kHz
Insert sensitivity	:	85 dB/mV

The maximum permitted sound pressure level will be reached when the applied noise power at the telephone insert is:

(85-10-60) dB down on 1 mW

In terms of voltage this is:

$$15\text{dB down on} \sqrt{\frac{1 \text{x} 300}{1000}} \ \text{V}$$

i.e. 15 dB down on 550 mV, which is approximately
100 mV rms.

(ii) Example 2

A headset using two 300-ohm telephone inserts wired in parallel:

Insert impedance	:	300 ohms nominal at 1 kHz
Insert sensitivity	:	(say) 100 dB/mW

Because the inserts are wired in parallel, the maximum permitted sound pressure level at either earpiece will be reached when the applied noise power across each insert is:

(110-60) dB down on 1 mW

In terms of voltage this is:

$$15\text{ dB down on}\sqrt{\frac{1\text{x}300}{1000}}\ \text{V}$$

i.e. 50 dB down on 550 mV, which is approximately 1.7 mV rms.

In this example, the maximum permitted electrical noise level as fed to the crew member's headset is 1.7 mV rms across 150 ohms.

(iii) Example 3

A headset using two 300-ohm high-sensitivity inserts wired in parallel and fed via a single 13 dB attenuator pad.

Insert impedance	:	300 ohms
Insert sensitivity	:	123 dB/mW
Built in attenuator	:	13 dB

In this example, the effective sensitivity of the telephone inserts, as measured at the headset connector terminals is 123 - 13dB, = 110 dB and the maximum permitted noise voltage is, therefore, as in the previous example, 1.7 mV as measured across the headset telephone input impedance.

4 ELECTRICAL DESIGN

4.1 GENERAL

4.1.1 It is essential that the techniques used to meet the design requirements do not render the final installation vulnerable to degradation of performance in the rotorcraft environment. Among the aspects which are of particular concern in this respect are:

(i) the number of mic-tel connectors and recorders permanently wired into the flight-crew intercom circuit should be as small as possible,

(ii) the practice of connecting a number of ground-crew, auxiliary flight crew or passenger mic-tel jack points directly in parallel with flight crew jack points should be avoided,

(iii) the ground crew network should be switched completely out of circuit during flight,

(iv) in complex installations, consideration should be given to providing a separate intercom network for the flight deck crew stations, thus, reducing the number of sources of electrical noise feeding into the crew members' telephones.

4.2 PRELIMINARIES

4.2.1 The design of the installation should begin with an examination of each item of equipment to determine the extent of any incompatibility with other items in the system. In particular, input/output signal levels and impedances should be matched to ensure that the various amplifiers are not overloaded and that each sub-system operates at its optimum level. It should be noted that:

(i) published data sheets are written in terms of sinewave test signals, which bear little resemblance to speech,

(ii) speech signals can be defined and measured in a number of ways, some of which conceal the fact that significant peaks in the waveform commonly rise to levels five to eight times greater than the long term rms value,

(iii) quoted impedances are often recommended matching impedances, or even merely nominal or conventional load impedances, rather than the actual measured values. This can be significant when designing matching devices and also when considering the effects of variation of impedance which may result from parallel loading,

(iv) input and output impedances are sometimes dependent on frequency and can vary considerably over the audio frequency band.

4.2.2 The information required includes:

(i) input levels and load impedances and output levels and source impedances of the CCS equipment,

(ii) dynamic range of output signal levels obtained from the specified microphone when operated in the actual rotorcraft environment, taking into account the tolerances quoted in the microphone production test specification,

(iii) impedance, sensitivity and power requirement of the headset telephone insert when installed in the crew member's helmet; the variation of impedance with frequency should be measured in order to determine the need for a shaping network.

(iv) for each type of radio receiver:

(a) output impedance at 1000 Hz and its variation with frequency,

(b) recommended load,

(c) output signal level when the rf test signal applied to the aerial terminals is sinewave modulated to a depth of 80%, open-circuit and with the recommended load.

(d) audio frequency response curve and bandwidth of the audio output signal at the -3 dB points,

(v) the output signal level and source impedance over the audio frequency band and recommended load for all other signal sources feeding into the CCS.

Note: When defining tone or pulse signals, the amplitude must be taken as the peak rms value, whatever the waveshape,

(vi) for each type of radio transmitter:

(a) modulator input impedance at 1000 Hz and variation with frequency,

(b) input level/modulation depth characteristic at 1000 Hz. (If there is a choice between alternative high and low level modulator input circuits, the high level is to be preferred),

(c) modulation depth/sidetone audio output level characteristic at 1000 Hz, open-circuit and with the recommended load. Also, recommended load, output impedance at 1000 Hz and variation of impedance with frequency.

4.2.3 Preamplifiers or matching devices should be provided at all sub -system interfaces where the signal levels and impedance requirements are found to be incompatible and no other means exists within the sub-system for eliminating the incompatibility.

4.3 MICROPHONE PREAMPLIFIERS

4.3.1 Microphone preamplifiers may be used to raise the level of the microphone output signal or to isolate microphones operated in parallel input circuits. In the design of such amplifiers the following points should be noted:

(i) The input circuit should be matched to the impedance of the microphone in use.

(ii) The preamplifier should be able to handle the dynamic range of the microphone speech signal with less than 1% total harmonic distortion.

(iii) The output circuit should be designed to operate into the load impedance presented by the CCS (taking into account the possible variation caused by the connection of further preamplifiers in parallel), providing an output signal at the level specified for the system. If necessary, a preset gain control may be incorporated to permit accurate adjustment.

(iv) The frequency response at the -3 dB points should be 200 Hz to 6 kHz, with a sharp rate of cut-off outside this band.

(v) Microphone input transformers should not be used unless they are effectively screened.

(vi) The complete unit should be effectively screened against electro-magnetic radiation.

4.4 MATCHING DEVICES

4.4.1 In order to reduce distortion and to ensure that the system operates under optimum conditions, all equipment that feeds into the CCS should be matched into that system to provide the correct signal level at the recommended impedance. It is necessary also to ensure that, when operating in the transmit mode, the CCS microphone output circuit is correctly matched into each transmitter modulator. These requirements may necessitate the design and construction of special matching devices.

4.4.2 In its simplest form, for an unbalanced circuit, the matching device can consist of a single series resistor, but it is more usually an inverted L network or other suitable attenuator. The importance of maintaining symmetry in a balanced circuit should be noted.

4.4.3 Normally, the matching device should be designed so that it does not change the frequency response of the signal channel by more than ± 1 dB over the specified frequency band. However, when designing a matching device for a radio receiver having an audio bandwidth greater than 6 kHz, it is desirable to incorporate a bandpass filter to restrict the bandwidth to 200 Hz to 6 kHz at the -3 dB points. This, in conjunction with the following CCS amplifiers, will reduce the overall receiver channel bandwidth, thus limiting the higher frequency electrical noise and distortion products in the output signal.

4.4.4 It is important to describe each matching device fully in the installation documentation and to give any necessary adjustment and servicing instructions.

4.5 REDUNDANT CONTROLS

4.5.1 Some radio receivers are supplied with remote volume controls or controller panels. When the receiver audio output signal is fed into the CCS for distribution, these items are redundant because the system is already provided with selector switches and potentiometers specifically for the control of all incoming audio signals.

4.5.2 To avoid the possibility of operating controls in opposition and to eliminate unnecessary wiring with its attendant hazard of increased electrical noise and crosstalk, all redundant controls should be excluded from the installation.

4.6 SNEAK CIRCUIT PROTECTION

4.6.1 Sneak circuits are unintended electrical paths which permit the unwanted operation of control relays or diode switching circuits through apparently unassociated switches. Often this occurs when two relays are inadvertently connected in series through a common connection at the service selector switch and they are then sufficiently sensitive to operate from the power supply before the switch is closed to energize the selected relay.

4.6.2 If the CCS does not incorporate automatic protection against the possibility of sneak circuits, consideration should be given to the need for providing isolating diodes in series with the lines to all control relays and diode switching circuits, particularly to those in the radio transmitter-receivers.

4.7 DC SUPPLIES

4.7.1 In order to reduce electrical noise, it is essential to exclude raw (unsmoothed) dc from the system. All dc supplies to the installation, including those to amplifiers, relays, transmitter control lines and panel lamps, should be derived from adequate low-impedance filter networks preferably provided with effective transient suppressor circuits. Suitable suppressors should be fitted as necessary to eliminate high voltage switching spikes on transmitter control lines.

5 MECHANICAL DESIGN

5.1 PHYSICAL LAYOUT

5.1.1 In addition to normal consideration of convenience of operation and ready access for routine servicing, careful thought should be given to the physical location of units within the rotorcraft to ensure that:

(i) no unit (or its cabling) is placed in an area affected by strong electro-magnetic fields,

(ii) microphone preamplifiers are located very close to the microphones with which they are used. If they cannot be contained in the microphone housing, or mounted on the crew member's headgear, they should be located at the rotorcraft-side member of the personal equipment connector, or within the jack box,

(iii) all matching devices are located close to the associated radio transmitter, receiver, or audio signal source, so that the signal is maintained at its optimum level in the main connecting cables,

(iv) the specified compass safe distance is maintained.

5.2 LAYOUT DIAGRAMS

5.2.1 In order to take full account of all significant factors and to ensure satisfactory cable design, efficient screening and short cable runs, a layout diagram should be prepared showing in detail the complete electrical interconnections between all units in the installation. This diagram should show each cable screen and include all relevant items such as matching devices, terminal blocks, bulkhead connectors, junction boxes and plugs and sockets. It is an advantage also to indicate the positions of possible sources of electrical noise and interference.

5.2.2 The diagram will be particularly valuable in determining the correct grounding point for each screen and will also help avoid multiple ground connections and the resulting creation of ground loops or common impedance circuits which can lead to the introduction of electrical noise. In pursuing this requirement, it may be necessary to modify the grounding arrangement within certain equipment items.

5.3 CABLE ROUTEING

5.3.1 To reduce the pick-up of electrical noise and interference, the routeing of cables should be planned with care, taking into account the following points:

(i) No cable loom containing audio signal lines should carry ac power supply lines of any description.

(ii) All signal cables should be segregated from ac power supply cables and navigation lamp wiring, with a separation of at least 150 mm. Where it is impracticable to maintain this separation, areas of proximity should be restricted to the absolute minimum and cables which cross should do so at an angle of 90 degrees.

(iii) Cables should not pass through regions affected by strong electro-magnetic fields. Where conditions make adequate isolation impossible, suitable metal screening or a cable conduit should be provided.

(Note: At frequencies below about 1 kHz, copper and aluminium screens are not effective against magnetic fields. The only protection against magnetically induced interference is a ferro-magnetic conduit having a wall thickness equivalent to at least 0.05 mm of mumetal).

(iv) In areas close to rf transmitters and aerial systems, full advantage should be taken of the partial screening provided by the rotorcraft structure, which provides a useful degree of attenuation except where discontinuities occur at doors, windows and behind non-metallic structures.

(v) Consistent with all other requirements, cables should be kept as short as possible. The total length of cabling between each crew member's headset and the local CCS control unit should not exceed 5 metres, including the length of the actual headset connector.

5.4 WIRING

5.4.1 In addition to meeting the general requirements for all aerospace electrical wiring specified in DEF STAN 00-10, all wiring should conform to the following requirements:

(i) Where possible, specially designed multiway cables should be used in preference to made-up cable looms (see para 5.5). Each cable should be paired, twisted and provided with an outer tinned copper braided screen to reduce the exportation of interference to other rotorcraft systems and should then be insulated with an outer sheath.

(ii) Plugs, sockets and terminal blocks should be used as sparingly as possible; terminal blocks should be used in preference to plugs and sockets where space permits and quick disconnection is not essential.

(iii) Cable plugs, sockets and terminal blocks should be selected from the types listed in para 5.6.

(iv) All microphone signal leads and balanced audio lines, whether run independently, included in a multiway cable or incorporated in the wiring of a matching device or preamplifier, should consist of a pair of twisted leads, with a pitch not exceeding 10 mm, enclosed in a screen; the twisting should be maintained up to the connecting points.

(v) In multiway cables, all microphone lines should be grouped at the centre with the remaining wires arranged concentrically around them and signal lines should be alternated with non-signal lines, such as ground returns, dc power and control lines.

(vi) All telephone and unbalanced audio lines should consist of a single screened lead with a separate, unscreened return lead. If necessary, to avoid an excessive number of wires in multiway cables, several related unbalanced screened audio signal lines carrying signals at the same basic level can share a common unscreened signal return line. Telephone lines and receiver service lines should not share the same signal return line.

(vii) Each screened lead should be individually enclosed in an insulating sheath to isolate all screens throughout the length of the cable.

(viii) The rotorcraft structure should never be relied upon to provide the ground return path for any audio frequency signal line; a separate return lead should be provided for the unbalanced signal lines.

5.5 CABLE SPECIFICATIONS

5.5.1 Standards. All cables should be constructed to conform with the design requirements and wire specifications detailed in:

AP 113D17001 :	Cables, general use, aircraft and ground equipment
DEF STAN 61-11 (Pt 1) :	Cables, radio frequency
DEF STAN 61-12 (Pt 4) and (Pt 5) :	Interconnecting cables
DEF STAN 61-12 (Pt 6) and (Pt 8) :	Equipment wires
DEF STAN 61-12 (Pt 9) :	Cables, radio frequency

5.5.2 Materials. Multiway cables should be constructed using the following materials:

Wire	:	tinned annealed copper
Wire size	:	16/0.2 mm for cables with up to 6 cores 7/0.2 mm for cables with more than 6 cores
		(Note: Power lines can be duplicated as necessary for increased capacity).
Wire insulation	:	pvc or ptfe
Screening	:	tinned copper, braided with a minimum fill factor of 64%
Outer sheath	:	Glass fibre, braided and lacquered

5.5.3 Colour Coding

(i) The following basic colour coding should be used:

Microphone (+ve)	:	yellow
Microphone (-ve)	:	violet
Telephone (+ve)	:	blue
Telephone earth	:	brown
Audio service lines	:	white
Signal earth	:	Green
Intercom mic lines	:	orange/green and orange/black
Combined intercom line	:	orange
Override audio	:	pink
DC supply lines (+ve)	:	red
Panel lamps (dimmer)	:	red/green
DC return (0V)	:	black
R/T control lines	:	grey
Override control	:	pink/red
I/C mode switch	:	pink/green
R/T mode switch	:	pink/blue
Mute control	:	pink/brown

(ii) Where several lines in the same category are included in one cable, they should be distinguished by a coloured tracer added to the basic function colour. Tracers should be used in the following order:

1 Red
2 Orange
3 Green
4 Blue
5 Brown
6 Black

(iii) In accordance with AvP 118, the outer sheath of all cables carrying audio signal lines should be colour coded yellow to denote that they are category X; i.e., susceptible to interference.

5.5.4 Connector Pin Allocation

(i) The allocation of line functions to specific connector pins as established at the system LRUs, particularly within the CCS, should be preserved so far as possible throughout the installation.

Note: LRU - Line Replacement Unit. A separately referenced, individual unit of equipment scheduled as replaceable during first line servicing procedures on board the rotorcraft.

(ii) For connectors at which the established connection does not apply, the greatest practicable separation should be maintained between high and low level signal lines; such as microphone, telephone lines and audio service lines should be separated, with intervening pins used for ground, screening, dc supply and switch connections to provide some isolation.

(iii) All plug and socket shells should be connected to the dc ground return line and not the signal return line.

5.6 RECOMMENDED CONNECTORS

5.6.1 All cables connecting to LRUs will be fitted with the appropriate mating connector. Wherever possible, signal-line connectors specified for use elsewhere in the installation should be selected from the following recommended types:

(i) Terminal Blocks

(a) In accordance with BS 2G 197, but limited to sizes 6.32 and 4.40, colour black, fig. 1 and fig. 3 only.

(ii) Mic-tel Jacks and Sockets

(a) LEMO Type 2314 (Plessey Assembly Drawing No.508/1/17516/007).

(Note: The Pattern 107 NATO Jack plug and socket, 10H/9466652 and 10H/18574 respectively are not recommended because no contacts are available for the separate connection of the microphone and telephone line screens).

(iii) Multiway Plugs and Sockets

(a) DEF STAN 59-56 (Pt 1) Plugs and sockets electrical Pattern 602.

(b) Pattern 100 to DEF STAN 59-35 Part 1. Range available: 9, 15, 25, 37 and 50-way, normal or floating-bush mounting types.

(c) CANNON DPX rack mounting range.

5.7 SCREENING

5.7.1 The effectiveness of audio frequency cable screening can easily be impaired and it is, therefore, important to observe the following requirements:

(i) Cable screens should not be used to provide a signal return path; a separate lead should be used.

(ii) Each screen should be grounded at one point only. In general, this point should be at the cable termination nearest, electrically, to the main system distribution box.

(iii) Apart from the single grounding point, each cable screen should be insulated from all other screens and from all possible ground connections.

(iv) Apart from the outer cable screen and the headset connector, no screen should be connected directly to the metal shell or shield of a plug, socket or terminal block and each should be connected to a separate pin.

(v) The outer cable screen should be connected to the plug or socket shell only at one end of the cable.

(vi) The screen grounding point for all signal lines should be either a common signal (no dc power) ground busbar having a single connection to the airframe structure, or the body of an LRU bonded directly to the airframe.

(vii) At every plug, socket and terminal block, each cable screen should be allocated a separate pin to provide an individual through-connection, isolated from any local ground connection.

(viii) The outer cable screen should be connected to the shell of the plug or socket at one end of the cable only - the end nearest, electrically, to the main system distribution box. The plug/socket pin allocated to the outer screen should be wired to the chassis ground within each LRU.

(ix) On each headset connector, the plug/socket shell and internal shield should be connected to the microphone lead screen only.

(x) At each termination, cable screens should be kept intact as close as possible to the pin connection; the gap in the screening of each individual core should not exceed 20 mm.

(xi) Terminal blocks carrying audio frequency signal lines should be screened by means of a close fitting shield, grounded locally.

5.8 BONDING

5.8.1 It is essential that all permanently mounted items of equipment are efficiently bonded to the rotorcraft structure, as detailed in Chapter 708.

6 INSTALLATION TESTING

6.1 GENERAL

6.1.1 A detailed and comprehensive test schedule should be prepared to prove that the installation meets the criteria defined in para 3. The schedule should include the tests listed in the following paragraphs.

6.2 CABLING

6.2.1 Insulation Resistance. Measure the insulation resistance between all cable cores and between each individual core and earth. This test is to be carried out on all installed cables while they are totally disconnected from power supplies, LRUs, matching devices and other electronic units. The insulation resistance should be not less than 100 megohms.

6.2.2 Continuity. Verify the electrical continuity and grounding of all cable screens, shields and conduits. The measured dc resistance should not exceed 10 milliohms.

6.2.3 Transmission Losses. With each signal path terminated in appropriate resistive loads, confirm that transmission losses at 1000 Hz do not exceed 1 dB.

6.2.4 Crosstalk. Confirm that, at 1000 Hz, the coupling between all signal lines when disconnected from all LRUs but terminated in appropriate resistive loads, does not give rise to a level of crosstalk exceeding -120 dB.

6.3 LRUs

6.3.1 Confirm that all LRUs are effectively bonded to the mounting structure, with a measured dc resistance not exceeding 10 milliohms.

6.4 COMPLETE SYSTEM

6.4.1 GENERAL

(i) Confirm that the necessary power supplies to the system are connected and are at the correct voltage levels.

(ii) Confirm that the complete system is fully operational and that the performance of each item of equipment meets the appropriate specification, in respect of audio output levels, distortion, crosstalk and power supply ripple rejection.

(iii) Confirm that the overall frequency response of each signal channel is within ± 1 dB of the specified amplitude/frequency characteristic from 50 Hz to 7 kHz (taking into account any filters incorporated in the matching devices).

6.4.2 Electromagnetic Compatibility

(i) Confirm that, with all other electrical installations in the rotorcraft operative, any unwanted signal appearing in any section of the system is at least 50 dB below the normal level of the wanted signal.

6.4.3 Electrical Noise (static test)

(i) Confirm that, with the rotorcraft ac generators switched on and all ac and dc power circuits operative, the induced electrical noise appearing on the microphone and audio service lines is at least 5O dB below the normal level of the wanted signal.

6.4.4 Electrical Noise (dynamic test)

(i) Under flight test conditions, with engines running and all electrical services operational, all dummy microphones switched on, and the intercom facility selected in isolation but with all volume controls set to maximum, confirm that the level of noise of electrical origin produced in the crew member's headset earpieces is at a sound pressure level not greater than 60 dB above 20 micropascals. This can be related to the electrical noise level developed across any particular type of telephone insert, as shown in para 3.2.

(Note: For the purpose of this test it is necessary to exclude electrical noise of acoustic origin. This can be achieved by the use of dummy microphones, manufactured by drilling a small hole in the diaphragm of a normal microphone and filling the unit with Araldite. The dummy microphone is thus acoustically dead but it is still susceptible to electrical pick-up).

LEAFLET 707/2

RADIO AND RADAR INSTALLATIONS

SERVICING

1 INTRODUCTION

1.1 This Leaflet gives recommendations on the installation of radio equipment to aid servicing.

2 GENERAL

2.1 Items of associated equipment of a particular system should be grouped together as far as is practicable. (Chapter 112).

2.2 Where components are bolted to the rotorcraft structure, either quick release nuts or anchor nuts should be used, to ensure easy and quick removal. The use of loose nuts is not generally acceptable.

2.3 Units not provided with purpose built removing trays should be attached to airframe with quick release fasteners.

3 AERIAL SYSTEMS

3.1 As far as possible, and compatible with the structural requirements, aerials should be easy to replace in case of breakage or other failure.

4 AERIAL FEEDER AND ELECTRICAL CABLE INSTALLATION

4.1 Where a cable may be flexed, adequate slack should be provided.

4.2 The cleats securing radio cable assemblies to the airframe or to installed structure should be such that the connector can be removed from or placed in the cleat without the use of tools, except where bonding or special connector installations require a more positive form of cleat. Recommended forms of cleating include:

(i) Strapping for separating connectors of one installation from another to facilitate identification, except where a single connector is concerned,

(ii) Ducting and other runs with quick release attachments,

(iii) Spring clips,

(iv) Straps with button or buckle attachment,

(v) P. Clips

4.3 The above recommendations should be read in conjunction with the requirements of Chapter 707, para 6.4.

LEAFLET 707/3

RADIO AND RADAR INSTALLATIONS

AERIALS, ANTENNAE AND SCANNERS

1 INTRODUCTION

1.1 This Leaflet makes general recommendations regarding the manner in which aerials are located on airframes.

1.2 An aerial is a means of coupling electromagnetic radiations external to the rotorcraft with the equipment carried within the rotorcraft.

1.3 Transmitting aerials may handle considerable power. The requirements in the chapter are to ensure that an efficient aerial installation is produced and that energy is not wasted in coupling losses. The manufacturers' recommendations regarding matching of feeder cables should be followed.

1.4 Each aerial system will have a characteristic polar plot which may be modified by its installation in an airframe or by adjacent aerial systems. The Rotorcraft Project Director will determine if the polar performance of an aerial system needs to be demonstrated. Acceptable demonstration may involve ground tests, airborne tests or tests with models.

2 FIXED AERIALS

2.1 OMNI-DIRECTIONAL AERIALS

2.1.1 In general, communication aerials are omni-directional and may take the form of wire, probe, whip, blade or notch according to operating frequency.

2.1.2 Various manufacturers group stages of radio equipment in differing packages, but in all cases it is necessary to establish a high quality RF bond between the ground plane (usually airframe) and the RF output stage in the case of a transmitter, or the preamplifier of a receiver.

2.1.3 With MF and HF systems, separate aerial coupler units are usually fitted. These should be located as close as possible to the active aerial element. Long feeders within the rotorcraft are not desirable and provide the possibility of coupled interference with other rotorcraft systems. Special tuning units are required in connection with notch aerials involving close liaison between the airframe and radio equipment manufacturers.

2.1.4 Aerials mounted in a prominent position susceptible to lightning strike should have a low d.c. resistance path to airframe so as to protect the radio or radar equipment.

2.1.5 Where the low resistance path is not inherent in the aerial design (such as folded dipole) then means to provide lightning strike diversion (e.g., a glass enclosed spark gap) should be provided. A high resistance short to dissipate static is also necessary.

2.2 DIRECTIONAL AERIALS

2.2.1 Directional aerials may be either a purpose built assembly installed as a unit, or a combination of fixed aerials to give a directional effect.

2.2.2 Directional aerials should be located and orientated so as to look away from the airframe and not across it e.g., radio altimeter aerials will look vertically downwards from the underside.

2.2.3 Directional aerials intended to identify azimuth bearing of a signal relative to airframe should be located as near to the electrical centre as practical. Such aerials will exhibit quadrantal error characteristics. The limits of quadrantal error will be set by the Rotorcraft Project Director according to the application.

3 SCANNING AERIALS

3.1 There may be a need to scan highly directional aerials in order to find a signal. This will apply to aerials which are most sensitive to received signals within a small solid angle subtended at the aerial. Scanning may be accomplished electrically or by physically moving the aerial system.

3.2 ELECTRICAL SCANNING

3.2.1 Electrical scanning is usually achieved within a packaged electronic unit. The active aerial face is normally protected with a dielectric panel suitable for mounting external to or flush with the rotorcraft's skin. The 'viewing' angles of the device must be ascertained from the manufacturer and the unit installed where these arcs are not obstructed.

3.3 MECHANICAL SCANNING

3.3.1 Depending on the application aerials may move in one, two or three axes. Each axis may have full or restricted movement.

3.3.2 Control of the movement may be simple on/off selection of scan or servo controlled alignment permitting a crew member or other system on board to direct the aerial precisely.

3.3.3 By virtue of their configuration mechanically scanning aerials are externally mounted and normally housed within a weatherproof radome specially constructed to be transparent to the radio frequency being used. Close liaison is required between the equipment manufacturer, the radome manufacturer and the airframe manufacturer to ensure adequate electrical, aerodynamic and structural performance is achieved.

4 GENERAL PERFORMANCE

4.1 Calibrated ranging systems should not be susceptible to changes in rotorcraft configuration resulting from movement of flying controls or landing gear and/or carriage and discharge of external stores.

4.2 Aerials should be located to minimise mutual interference. A definition of the total aerial system should be agreed by the Rotorcraft Project Director.

4.3 Where aerials provide their associated system with bearing data relative to airframe their alignment must be achieved within the system manufacturers' prescribed limits. Jigs and dowel pins may be used to assist primary location and subsequent replacement.

CHAPTER 708

BONDING AND SCREENING

1 GENERAL

1.1 The requirements of this chapter are applicable to rotorcraft, excluding the engine ignition systems. The bonding and screening requirements applicable to engine ignition systems are given in Specification D.Eng.R.D.2015.

1.2 The purpose of screening is to prevent the radiation of radio-interference fields from and to electrical equipment and the associated wiring.

1.3 The purpose of bonding is to reduce the risk of fire due to lightning discharges and the accumulation of electrostatic charges, and to increase the efficiency of the radio installation.

1.4 For the purpose of the requirements of this chapter, the main ground system is defined as the rotorcraft structure for metal rotorcraft and as the interconnected main bonding system in other cases.

2 SCREENING AND INTERFERENCE SUPPRESSION

2.1 CIRCUITS

2.1.1 All circuits which are liable to induce radio frequency disturbances into radio circuits shall be totally enclosed by a grounded metallic sheathing or screen, and the screen shall be bonded to the main ground system at two points at least; these points shall be within 457 mm (18 in.) of each end. Circuits associated with electrical fuzes, explosive caps and other circuits sensitive to induced effects are dealt with in Chapter 706, para 5).

2.1.2 Circuits which cause interference but which operate for a short time only and at infrequent intervals usually need not be screened.

2.2 RADIO INTERFERENCE SUPPRESSORS

2.2.1 Radio interference suppressor components shall, where possible, be incorporated within the equipment.

2.2.2 When external suppressor units are used, they shall be located as near as possible to the source of interference and screened cable shall be used between the source of interference and the suppressor.

2.3 SUPPRESSION OF IGNITION INTERFERENCE

2.3.1 Ignition interference need be suppressed during normal engine running only.

2.3.2 L.T. booster coils shall be fitted with a suppressor in the supply leads from the battery, and all the wiring from the suppressor to the coil and the magnetos shall be fully screened.

2.3.3 H.T booster coils shall have a spark gap inserted in the H.T. starting lead. The cable from the magneto to this isolating spark gap shall be screened. The cable from the isolating spark gap to the booster coil shall be unscreened. A suppressor in the battery supply leads is not necessary.

3 BONDING[1]

3.1 GENERAL

3.1.1 The purpose of electrical bonding of the rotorcraft structure, components and equipment is:

(i) to prevent electro-static potential differences between adjacent parts (Class S bonding),

(ii) to minimise the possibility of electric shock from the electrical supply and distribution system (Class H bonding),

(iii) to provide an adequate path for electrical fault currents on those systems which use the airframe as a conducting path (Class C bonding),

(iv) to prevent electrical interference with the functioning of the systems (Class A bonding, antenna installations, and Class R bonding, RF potential),

(v) to provide an adequate, homogeneous ground plane for each aerial,

(vi) to facilitate the passage of lightning currents across the surface of the rotorcraft.

3.1.2 All metallic parts of the structure and skin shall be connected together to form an electrically continuous system of low and substantially constant impedance and resistance (see AvP118, Chapter 6 para 5). Precautions shall be taken to prevent variable, intermittent or vibratory contact between metal parts.

3.1.3 Due regard shall be given to the prevention of corrosion at all contacting surfaces as laid down in Chapter 407.

3.1.4 Consideration shall be given to the bonding of non-metallic structure or components, in particular those made from carbon fibre reinforced composite materials, taking into account:

(i) the electrical properties of the material,

(ii) the ability to make satisfactory connections.

3.1.5 Bonding straps shall have minimum inductance. They shall be of highly conducting material, preferably solid but which may be flexible, with a width/thickness ratio of at least 5:1 and length/width ratio not normally greater than 5:1.

3.1.6 All bonding straps shall be kept as short and direct as possible. The number of straps to be installed shall be kept to a minimum by careful design. Straps in series are not permitted. When flexible bonding leads are used they should be in accordance with DEF STAN 61-12 (Pt.20).

3.1.7 Recommended maximum values of the direct current resistance of electrical bonds are given in AvP118 Chapter 7 Table 3. Where it is found impossible to achieve a value lower than the appropriate recommended maximum, the problem shall be discussed with the Rotorcraft Project Director.

3.1.8 All control and distribution panels of metal construction shall be bonded to the airframe.

3.1.9 The engine shall be electrically connected to the Main Ground System by at least two removable Primary Conductors as in para 4.1, one on each side of the engine.

3.2 CLASS A BONDING (ANTENNA INSTALLATION)

3.2.1 The specific bonding requirements of antennae will be dictated by the type of antenna used, and decided in consultation with the Rotorcraft Project Director. Generally, antenna systems shall be bonded so that current flow from the attachment surface of the antenna to the mating surface of the rotorcraft shall have maximum possible contact area with minimum impedance.

3.2.2 Hatches in the vicinity of, or forming part of, an antenna ground plane shall be continuously bonded to the rotorcraft skin.

3.2.3 Provisions shall be made for circumferential RF continuity between outer conductors of coaxial antenna transmission lines and ground planes of antennae.

3.3 CLASS C BONDING (CURRENT RETURN PATH)

3.3.1 The bond between equipment and rotorcraft structure shall be adequate to carry the maximum fault current.

3.3.2 The total impedance of wires, cables, ground return paths and their interconnections shall be such that the voltage drop between the point of regulation and the load does not exceed the limits stated in BS 3G100. For current return leads the bonding connection shall be made to a terminal end of sufficient size properly attached to the Grounding Point.

3.3.3 Magnesium alloy structure shall not be used as a current path return.

3.3.4 To prevent ignition in the event of power faults within an equipment, bonding shall be provided in areas where hazardous conditions exist, due to the presence of explosive fuels and gases.

3.4 CLASS H BONDING (SHOCK HAZARD)

3.4.1 In no case shall a path, which carries a voltage greater than 30 volts RMS A.C. or 50 volts D.C. and which may be touched by personnel, be exposed except during servicing.[2]

3.5 CLASS R BONDING (RF POTENTIAL)

3.5.1 All electrical and electronic equipment which emits electro-magnetic energy shall be installed to provide a continuous low-impedance path from the equipment enclosure to the metal rotorcraft structure. Bonding straps shall be provided across vibration isolators with a minimum of two straps per unit of equipment.

3.5.2 All conducting items having any linear dimension of 300 mm (11.811 in) or more installed within 300 mm (11.811 in) of unshielded transmitting antenna lead-ins shall have a bond to the airframe.

Note: Metal-to-metal contact is preferred but, if a strap is used, it should be as short as possible.

3.5.3 The rotorcraft shall be designed with inherent RF bonding and a uniform low impedance conducting path produced during construction, through the skin and between all structural components. Hatches, access doors, etc, shall be either:

(i) continuously bonded to the rotorcraft skin, or

(ii) permanently insulated from the rotorcraft skin and the conducting path maintained by bonding straps (see also para 3.2.2).

3.5.4 All metallic equipment mountings shall be bonded to the airframe.

3.5.5 Where equipment mounting trays carry fixed back-plug and socket assemblies, all cable screens and connector shells shall be RF bonded to the tray.

3.6 CLASS S BONDING (STATIC CHARGE)

3.6.1 All isolated conducting items (except antennae) having any linear dimension greater than 100 mm which are external to the rotorcraft, carry fluids in motion or otherwise are subject to frictional charging shall have a mechanically secure connection to the rotorcraft structure. The resistance of the connection shall be less than 1 ohm when dry.

3.6.2 All metal parts, pipes, tubes, or hoses that carry fuel or other fluids shall be bonded to the Main Ground System with a mechanically secure connection having a resistance of 1 ohm or less. The pipe, tube or hose installation shall not be designed to be a path of primary electrical power. Non-metallic plumbing installations shall be designed so that the static charges have an energy level less than the value needed for ignition of an air/fuel mixture. Metallic unions in non-metallic pipes shall be bonded to the Main Ground System.

Note: Flexible tubing connecting economisers to oxygen masks need not be bonded.

4 LIGHTNING STRIKE PROTECTION[1]

4.1 PROTECTION - GENERAL REQUIREMENTS (CLASS L BONDING)

4.1.1 Throughout the design of the rotorcraft, consideration shall be given to the possible effects of a lightning strike and the incorporation of protected measures to minimise the effects.

4.1.2 At an early stage in the design process a Study of lightning protection shall be carried out to check that due attention has been given to lightning hazards and to decide whether any additional protection need be provided. In doubtful cases the Study shall recommend a programme of simulated lightning tests. The Study shall pay particular attention to those external parts of the rotorcraft which may be attachment points for lightning and to the possible lightning current paths through the rotorcraft skin and structure.

4.1.3 Primary conductors (i.e., straps in the Main Ground System and for lightning protection), when made from copper, shall have a cross-sectional area of not less than 18 mm^2, (and a thickness between 0.2 mm and 1 mm) where a single conductor is likely to carry the whole discharge. Where two or more conductors are likely to share the discharge the cross-sectional area of each shall not be less than 9 mm^2 with the same thickness requirements. Aluminium Primary conductors shall have a cross-sectional area giving an equivalent carrying capacity for a lightning stroke i.e., a single pulse of current: this means that the above figures for cross-sectional area are to be multiplied by a factor of 1.5.

4.1.4 Primary conductors shall be used for:

(i) connecting together the main Grounds of separable major components which may carry lightning discharges,

(ii) connecting engines to the main Ground,

(iii) connecting to the main Ground all metal parts presenting a surface on or outside of the external rotorcraft surface.

4.1.5 Individual bonding straps shall be as short as possible and have a cross-sectional area not less than 18 mm^2 for tinned stranded copper wire where a single strap is likely to be subjected to the full lightning discharge, and proportionately less where two or more straps are likely to share the discharge. For aluminium straps the cross-section area shall be 1.5 times that for copper.

4.1.6 Soldered connections shall not be used on straps required to carry lightning currents. The electrical and mechanical adequacy of the attachment to terminals shall be verified by test.

4.2 PROTECTION OF THE STRUCTURE

4.2.1 Rotorcraft of conventional metallic construction are considered to be adequately protected structurally, provided that they comply with the bonding requirements of para 3 of this chapter.

4.2.2 Where non-metallic or composite materials are used in construction, consideration shall be given to their protection by strike plates or other means. Strike plates shall extend round the nose and tail of the fuselage, round the tip of each wing and the extremities of the tail unit. The strike plates shall be provided on the exterior of the structure except where existing metallic structure serves the same purpose.

4.2.3 Unless otherwise agreed with the Rotorcraft Project Director, each strike plate shall consist of a strip equivalent to at least 0.45 mm thick (26 swg) copper, 25 mm in width, which shall be bonded to the main metallic structure.

4.2.4 Where a non-metallic or reinforced composite structure is used to house electronic equipment then precautions shall be taken to guard against possible structural damage due to lightning strikes.

4.2.5 Where transparent components such as windows, windscreens, etc, contain electrically heated films or elements, precautions shall be taken to prevent a lightning strike or an electro-static discharge puncturing the transparency system.

4.2.6 Precautions shall be taken to ensure that induced voltages from lightning strikes and electro-static discharges do not produce dangerous permanent or transient effects on the rotorcraft's electrical systems.

4.3 PROTECTION OF CONTROL SURFACES AND CONTROL SYSTEMS

4.3.1 Control surfaces, and any other moving parts shall have bonding straps across each hinge except for installations having a single hinge in which case a minimum of two straps is required.

4.3.2 Where necessary, additional straps shall be used between the control surface and structure to protect the control system so that the length of the discharge path through the system is at least 10 times the length of the path through the bonding strap (or straps).

Note: For lightning protection, a piano type hinge may be considered as self bonded provided the resistance across the hinge is less than 0.01 ohms.

4.4 PROTECTION OF PROTRUSIONS AND EXTERNAL PARTS

4.4.1 All external electrically isolated conducting objects which protrude outside the rotorcraft surface, excluding antenna radiating elements, shall be bonded to the rotorcraft skin or structure.

4.4.2 Antenna systems shall be so designed or protected that a lightning discharge will cause only local damage and will not endanger the rotorcraft or its occupants.

4.4.3 Consideration shall be given to the possibility of damage to the rotorcraft electrical system due to voltage spikes caused by strikes to protrusions or other external parts which have connections into the electrical system (e.g., pitot booms).

4.4.4 Consideration shall be given to the protection of large non-conducting projections such as canopies, astrodomes and radomes taking account of their operational function and performance. Protection shall be by a conducting cage bonded to the airframe and the geometric design of the cage shall be such as to prevent puncture of the non-conducting projection and to protect enclosed personnel or equipment.

Note: To confirm the geometrical design of the protection system it may be necessary to carry out 'puncture/flashover' tests at a High Voltage laboratory to check that strikes go to the protection system and do not puncture the dielectric wall.

4.4.5 The conducting cage shall be made up of conductors of current carrying capacity and mechanical robustness (including the means of fixing to the dielectric wall) sufficient to carry the full lightning discharge current and shall comply with the requirement for Primary Conductors (see para 4.1.3).

4.4.6 During consideration of protrusions and external parts, attention shall be paid to all flight configurations, including the various external store options and the landing gear in the extended position.

4.5 PROTECTION OF THE FUEL SYSTEM

4.5.1 The outlets of fuel venting and jettisoning systems shall be so located and designed as to minimise the risk of their being struck by lightning. They shall not, under the atmospheric conditions for which the rotorcraft is designed, produce corona discharges of such magnitudes as will ignite any fuel/air mixtures of the ratios likely to be present and they shall not permit fuel and its vapours in flammable concentrations to pass close to parts of the rotorcraft capable of producing corona discharges which could ignite fuel/air mixtures.

4.5.2 The fuel system of the rotorcraft shall be so designed in relation to the Main Ground System that the passage of lightning discharges through the Main Ground System will not produce, by conduction or induction, such potential differences as could cause electrical sparking in areas where there may be flammable vapours.

4.5.3 Externally mounted metallic fuel tanks need not be provided with special protection other than the provision of normal bonding, providing precautions have been taken to prevent damage to the rotorcraft.

4.5.4 All metal parts on or inside non-metallic fuel tanks shall be bonded together and to the Main Ground System. Externally mounted non-metallic fuel tanks shall be provided with a Ground System consisting of not less than three strips equivalent to at least 0.45 mm thick (26 swg) copper of 25 mm width extending the whole length of the tank and evenly disposed about the circumference.

These strips shall be bonded together at the ends and shall be electrically connected to the Main Ground System.

Note: Copper or copper alloys shall not be used for bonding in fuel tanks unless suitably protected (tinned or plated) so that contact with usable fuel is not possible. Plating medium shall not be soluble in fuel nor have any effect on fuel properties.

4.6 LIGHTNING PROTECTION TESTS

4.6.1 When the Study of lightning protection has recommended high current pulse tests on parts of the rotorcraft in order to verify the adequacy of design, the tests shall conform to the recommendation of Leaflet 708/2, except as agreed with the Rotorcraft Project Director.

4.6.2 The electrical resistance of a complete rotorcraft of metal construction, as measured during these tests between any of the metal extremities, shall not exceed the values agreed with the Rotorcraft Project Director (see para 3.1.6).

REFERENCES

Reference	ASCC Air Standard	STANAG	DEF STAN
1	-	3659	-
2	-	-	61-15

LEAFLET 708/0

BONDING AND SCREENING

REFERENCE PAGE

<u>MOD Specifications</u>

D Eng PD 2015	.. ..	High-tension ignition harnesses for piston type aero-engines

<u>RAE Reports</u>

EL.1489	.. .. ..	Lightning and aircraft, with particular reference to incidents involving Royal Air Force aircraft during the period 1948 to 1957

<u>British Standards Specifications</u>

G 175	.. .. ..	Aircraft fuel nozzle ground plugs and sockets

<u>SBAC Drawings</u>

AS 6322	.. .. ..	Assembly of bonding socket

<u>Defence Standards</u>

61-12 (Pt.20)	.. ..	Wires, cords, and cables, Electrical - Metric Units Pt 20 Braids, wire
61-15	.. .. ..	Precaution against electric shock in design, installations and use of servicing workshops and training laboratories for electrical, electronic and weapons equipment

LEAFLET 708/1

BONDING AND SCREENING

SCREENING - GENERAL RECOMMENDATIONS

1 In general, commutators, sliprings and moving contacts, should always be treated as possible sources of radio interference, and appropriate screening and suppression action should be taken.

2 The continuity of screening, should be maintained by the use of approved cable glands and the use of metal enclosures for junction terminals and equipment.

LEAFLET 708/2

BONDING AND SCREENING

RECOMMENDED LIGHTNING TESTS

1 ELECTRICAL PARAMETERS OF LIGHTNING

1.1 When a rotorcraft is struck by lightning it forms part of the discharge path between two charge centres or between a charge centre and earth, so that there is at least one entry point and one exit point for the current that flows; because it is in practice impossible to distinguish between entry and exit points they are more generally called "attachment points". There are three main types of lightning flash, as follows:

(i) Intra-cloud (between charge centres of opposite sign in the same cloud).

(ii) Inter-cloud (between charge centres of opposite sign in different clouds).

(iii) Earth flashes (between a charge centre and earth). These may be sub-divided into positive flashes and negative flashes depending on the sign of the charge centre. Most earth flashes are of the negative type.

1.2 It is generally considered that the earth flash is most damaging and therefore simulated lightning tests are based on this type. The positive flash usually consists of a single stroke (that is, a single pulse of current) typically of the shape shown in Fig.1(a). A negative flash, shown diagrammatically in Fig.1(b), typically consists of several strokes of high peak amplitude with a low amplitude continuing current between some of the peaks; in addition, the peaks are sometimes extended by currents of intermediate magnitude (about 2 kA) flowing for a few milliseconds.

1.3 The most important parameters of the current waveform are peak current, rate of rise, total duration, charge transferred (coulombs) and action integral $\int i^2\,dt$ (A^2s). The last is proportional to the energy dissipated in a given resistance and may alternatively be expressed in units of joules per ohm (the units are equivalent, so that the numerical value is the same in either case). Because of the wide range of numerical values of the parameters of natural lightning it is necessary for purposes of test to specify a simplified waveform incorporating the essential features of both positive and negative discharges to earth, and to select the level of severity to be applied. A waveform having four components and with numerical values of the parameters as given in Table 1 below is recommended for test purposes; this is designed to give effects equivalent to those produced by both positive and negative flashes of severity towards the high end of the range. The waveform is sketched in Fig.2. It should be noted that Component C (continuous current) should drop to near zero before the restrike (Component D) commences. For induced voltage applications, Component D should have an average rate of rise during the rise time of 100 kA/µS + 10%. It is not always necessary to apply all four components in a given test; selection of the components to be supplied is dealt with in para 4 below. It may also sometimes be allowable to apply the components in separate tests instead of combining them in one composite waveform.

Note: Further background information on lightning testing may be obtained from Refs 1 to 5.

2 GROUPING OF TESTS ACCORDING TO EFFECTS PRODUCED

2.1 The effects of lightning which should be simulated by laboratory testing can be divided into two groups, referred to as Group 1 and Group 2. The need for such a division is primarily brought about by limitations of laboratory test facilities. Group 1 covers those effects where the rise time of the pulse is relatively unimportant such as burning, eroding, and structural deformation, as well as the high pressure shock waves and magnetic forces produced by the associated high currents. Group 2 effects are predominantly those due to induced or injected voltages deriving from the flow of the lightning current and are essentially associated with a high peak current and a high rate of change. Table 2 indicates the effects and the electrical parameters most closely associated with them.

3 DIVISION OF ROTORCRAFT INTO ZONES ACCORDING TO LIGHTNING ATTACHMENT CHARACTERISTICS

3.1 Rotorcraft surfaces can be divided into three zones, with each zone having different lightning attachment and/or transfer characteristics. These are defined as follows:

(i) Zone 1 Surface of the vehicle for which there is a high probability of initial lightning flash attachment (entry or exit).

(ii) Zone 2 Surfaces of the vehicle across which there is a high probability of a lightning flash being swept from a Zone 1 point of initial flash attachment (see para 4 below for a discussion of swept strokes).

(iii) Zone 3 Zone 3 includes all of the surface area other than that covered by Zones 1 and 2. In Zone 3 there is a low probability of an attachment of the direct lightning flash. However Zone 3 areas may carry substantial lightning currents by direct conduction between two attachment points.

3.2 Zones 1 and 2 may be further sub-divided into A and B regions depending on the probability that the flash will hang on for a protracted period of time. An A region is one in which there is low probability that the arc will remain attached and a B region is one in which there is a high probability that the arc will remain attached. Some examples of zones are:

(i) Zone 1A Initial attachment point with low probability of flash hang on, such as a leading edge.

(ii) Zone 1B Initial attachment point with high probability of flash hang on, such as trailing edge.

(iii) Zone 2A A swept stroke zone with low probability of flash hang on, such as wing mid-chord.

(iv) Zone 2B A swept stroke zone with high probability of flash hang on, such as a wing inboard trailing edge.

4 APPLICATION OF CURRENT TEST WAVEFORMS

4.1 An important factor affecting lightning testing is the phenomenon of "swept strokes". A swept stroke results from the fact that the lightning channel remains substantially stationary in space as the rotorcraft moves forward. Thus relative to the rotorcraft the point of attachment sweeps backwards; it may either move smoothly or else it may dwell at various surface locations for differing periods of time, thus resulting in a skipping action which produces a series of discrete attachment points along the swept path. Thus for a part of a rotorcraft in a swept-stroke zone (Zones 1 and 2), an important problem in deciding the level of severity to which it should be tested is to determine the duration of lightning attachment (the dwell time) to one point. The character of the sweeping action and the length of dwell time are complex functions of the waveform, the local geometry, the nature of the surface, the speed of the rotorcraft and the nature of the airflow at the place considered. On a metallic surface it is a major factor in determining if sufficient heating can occur at one point to burn a hole or form a hot spot capable of igniting combustible mixtures or causing other damage. Experiments in swept-stroke simulators suggest that the dwell time is about 1 to 20 ms for rotorcraft of conventional shape; it is also generally agreed that a restrike will cause a re-attachment and that the interval between restrikes (or subsequent strokes) is about 50 ms. Therefore when the dwell time for a part in Zone 2A is being decided it is necessary either to conduct swept-stroke tests (see Ref 5) or to make the pessimistic assumption that the dwell time is 50 ms.

4.2 It is important to note that in natural lightning, the intermediate currents can flow at the end of a subsequent stroke (or restrike), so that in principle each attachment point could have intermediate currents flowing into it.

4.3 Tables 3 and 4 give the recommended application of current test waveforms for testing for Group 1 and Group 2 effects in the various zones. Tests should be applied in the order given. Since current flows in Zone 3 only by conduction from the other zones and there is therefore no arc attachment point, current for Zone 3 tests should in all cases be applied through a solid connection, not an arc.

5 REFERENCES

5.1 For further information on simulated lightning testing reference may be made to the following publications:

1	T E James and J Phillpott	Simulation of lightning strikes to aircraft UKAEA Culham Laboratory Report CLM-R111 May 1971.
2	R H Golde A A Hudson J D Ibbott and E L White	An aircraft lightning strike test facility - a study of requirements. Electrical Research Association, Leatherhead, Report ERA 71-167 December 1971.
3	J Phillpott	Simulation of lightning currents in relation to measured parameters of natural lightning. Proceedings of the 1975 Conference on Lightning and Static Electricity at the Culham Laboratory April 1975. Published by the Royal Aeronautical Society.
4	A W Hanson	Techniques of stroke tests on structures, components and materials. Proceedings of the 1975 Conference, as Ref 3.
5	UKAEA Culham Laboratory	Recommended practice for lightning testing of aircraft. UKAEA Culham Laboratory Report.

TABLE 1
TEST CURRENT WAVEFORM

	Parameter	Value	Tolerance
High Current Component A	Peak Current	200 kA	± 10%
	Action integral	$2 \times 10^6\ A^2s$	± 20%
	Pulse length	500 μS	
	Rise time	25 μS	
Intermediate current Component B	Average amplitude	2 kA	± 10%
	Charge transfer	10 C	
Continuing Current Component C	Amplitude	200-800 A	
	Charge transfer	200 C	± 20%
Restrike Component D	Peak amplitude	100 kA	± 10%
	Action integral	$0.25 \times 10^6\ A^2s$	± 20%

TABLE 2
CLASSIFICATION OF LIGHTNING EFFECTS

BASIC EFFECT	IMPORTANT PARAMETERS
GROUP 1 Metal skin puncture Hot spot formation Mechanical damage Magnetic forces Damage to composite structures Damage to lightning arresters Sparking Fuel ignition	Charge (coulombs) $(\text{Peak current})^2$ Action integral $\int i^2\ dt$
GROUP 2 Induced or injected voltages in wiring or equipment. Voltage flash-over Sparking Fuel ignition	Peak current Rate of rise current

TABLE 3

APPLICATION OF TEST CURRENT WAVEFORMS (GROUP 1 EFFECTS)

Test Zone	Current Component A	B	C	D
1A	✓	✓	✓ Note 1	
1B	✓	✓	✓	✓ Note 2
2A		✓	✓ Note 1	✓ Note 2
2B		✓	✓	✓ Note 2
3	✓ Note 3		✓ Note 3	

Note 1 Assume a current duration of 50 ms for continuing current components unless swept stroke testing shows otherwise.

Note 2 The continuing current should drop to near zero before the restrike commences.

Note 3 Current to be applied through a solid connection, not an arc.

TABLE 4

APPLICATION OF TEST CURRENT WAVEFORMS (GROUP 2 EFFECTS) (NOTE 3)

Test Zone	Current Component A	D
1A	✓	
1B	✓	✓Note 4
2		✓Note 4
3	✓	✓Note 4

Note 4 Component D to have an average rate of rise during the rise time of 100 kA/µS ± 10%.

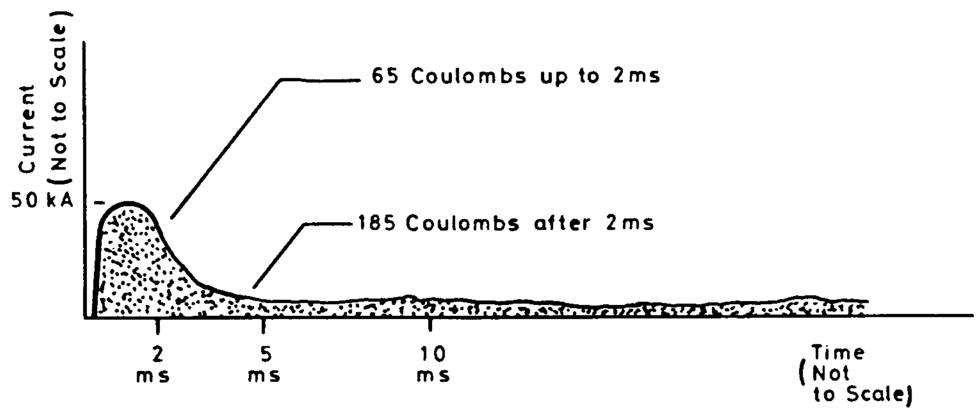


Fig 1a Current Waveform of Severe Positive Earth Flash

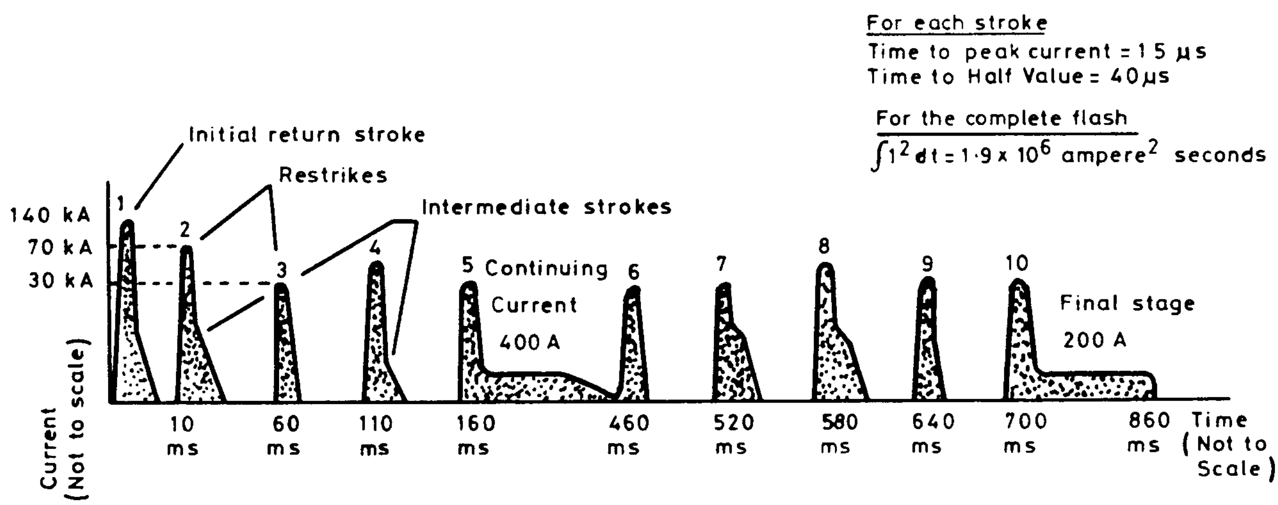


Fig.1b Current Waveform of Severe Negative Earth Flash

FIG. 1 - CURRENT WAVEFORMS OF SEVERE EARTH FLASHES

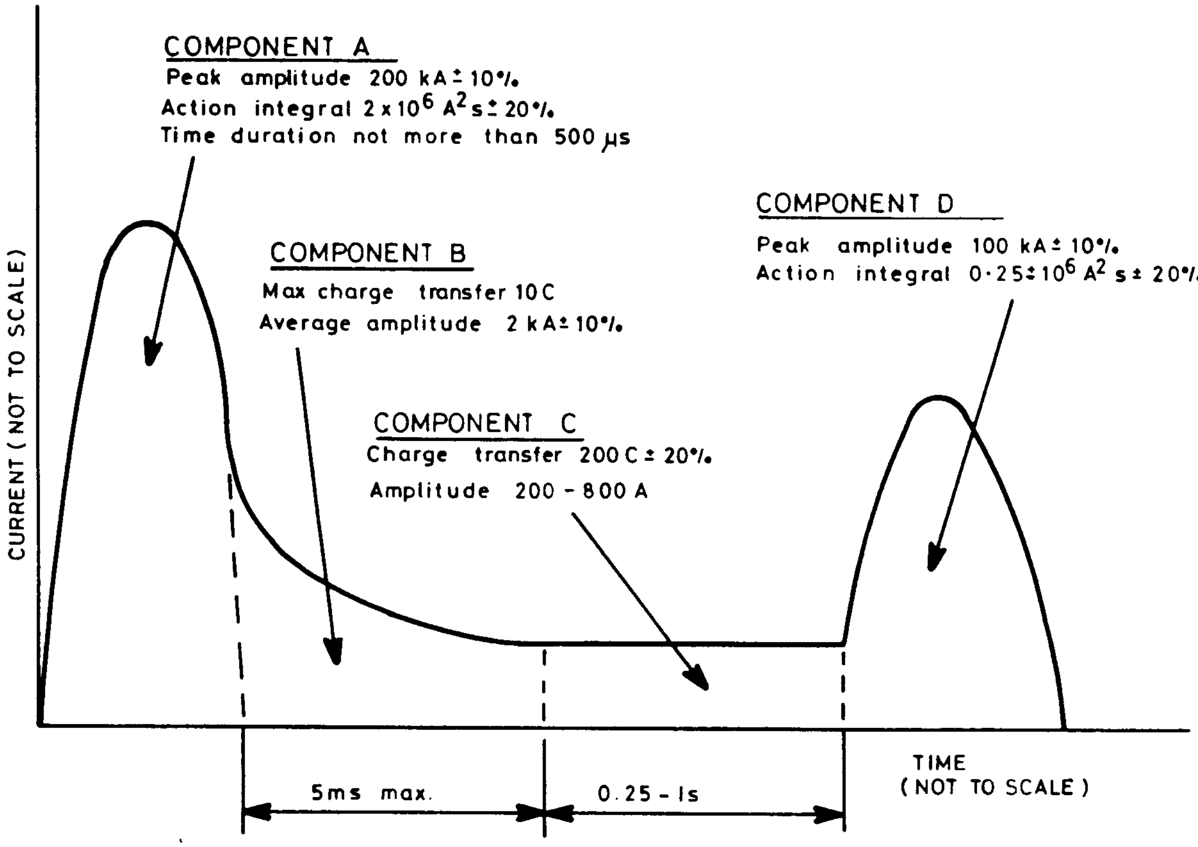


FIG. 2 - TEST CURRENT WAVEFORM

CHAPTER 709

GUN INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this chapter apply to all rotorcraft which are fitted with fixed or free movable) guns. They shall be read in conjunction with Chapter 710.

2 INSTALLATION REQUIREMENTS

2.1 DESIGN

2.1.1 The gun installation shall be designed to achieve maximum gun effectiveness throughout the specified gun firing envelope without prejudice to other rotorcraft systems and rotorcraft stability and control.

2.1.2 The gun installation shall be designed in accordance with the gun interface control document as approved by DA Arm MOD(PE).

2.2 LOCATION OF GUNS

2.2.1 The design shall be such that the lateral and vertical separations between the sight and the gun barrel bore axis are kept to a minimum.

2.2.2 The location of the guns shall be chosen with particular regard to the effects of gun firing directly on the engines, structures and other systems and indirectly, by changes to rotorcraft or equipment permanent magnetism, on compass detector units. The effects of the installation on the rotorcraft aerodynamics and the ease of servicing and replenishment of the gun installation must also be carefully considered.

2.2.3 Manually operated free guns shall be so located that, throughout the full range of gun movement, there shall be no obstruction to the line of sight or restriction of the operator equipped as provided for in Chapter 105.

3 GUN MOUNTINGS

3.1 A fixed gun mounting system shall permit angular movement of the gun to allow for harmonisation adjustments of a minimum of ½° around the nominal gun line. This is in addition to the angular displacement arising from manufacturing tolerances. Provision shall be made for locking the system to maintain the required setting.

3.2 The adjustment shall be continuous throughout the range required (i.e., not by shims) and its setting shall not vary through normal service use of the rotorcraft.

3.3 The relationship of the nominal gun line to the rotorcraft datum and flight attitude shall be such as to ensure ease of target tracking and safety of operation.

3.4 A free gun mounting system shall be designed to prevent the gun coming into contact with the airframe. When in its stowed position, the gun shall not prevent closure of adjacent doors, hatches, windows, etc.

3.5 The gun mounting system of both fixed and free guns, including fittings and associated structure, shall satisfy the strength requirements of the rotorcraft as a whole under all combinations of gun functioning and inertia loads throughout the flight envelope defined in the Rotorcraft Specification.

3.6 Gun mountings shall be designed to distribute the loads into the structure in such a manner as to prevent local distortion and fatigue failures occurring during the specified life defined in the Rotorcraft Specification.

3.7 The stiffness of the mounting system shall be such that the required gun functioning and accuracy are achieved; due regard shall be given to the effects of gun firing on the airframe and other rotorcraft systems.

4 GUN CLEARANCES

4.1 In allowing for the nominal gun line adjustment specified in para 3.1, adequate clearance of the gun from the adjacent structure/equipment shall be provided to ensure freedom from obstruction during the recoil and run-out movements. The clearances between the gun muzzle and the surrounding structure shall be sufficient to prevent damage by blast and/or projectiles at extreme positions of harmonisation adjustment.

4.2 A free gun mounting system shall be designed to provide adequate clearance of the gun from the adjacent structures/equipment and to ensure freedom from obstruction during the recoil and run-out movements.

5 RESTRICTIONS TO LINE OF FIRE

5.1 Fixed and free gun systems shall be so designed to prevent the projectiles from striking any part of the rotorcraft or stores, when the gun is fired from any of its normal operating positions.

6 GUN FIRING/SAFETY

6.1 Where a gun firing trigger is provided, it shall be protected from inadvertent operation by a safety device.

6.2 Where gun firing is remotely controlled no single failure shall cause the gun to fire inadvertently.

6.3 There shall be minimum delay between operation of the gun firing trigger/button and the guns firing (see Volume 1 Leaflet 710/1 para 7.2).

6.4 For each gun an accessible means shall be provided to inhibit firing and to enable personnel about to work on the rotorcraft to ascertain that the system is inoperative. This is in addition to the master armament safety switch (MASS) (see Chapter 710, para 9.5).

6.5 Remotely operated guns shall automatically be made inoperative when the undercarriage is selected "down". To enable the installation to be tested on the ground an override device shall be provided, the position of which is to be agreed at the armament mock-up conference.

6.6 The installation shall meet the current requirements of Electromagnetic Compatibility (EMC), Electromagnetic Pulses (EMP), Lightning and Electrostatic Hazards as defined in Chapter 710, para 9.3.

7 HARMONISATION

7.1 D A Arm will approve and issue the harmonisation diagram for each type of rotorcraft gun installation. This diagram will specify the correct gun and sighting system alignment pattern necessary to meet the Staff requirements.

7.2 It shall be possible to harmonise guns after the feed and ejection systems are connected.

7.3 Provision shall be made for the attachment of an alignment device to define the horizontal and vertical lines in relation to the airframe to facilitate the positioning of the harmonisation board.

8 GUN INSTALLATION ENVIRONMENT

8.1 A gun installation purging system shall be provided to prevent the concentration of flammable gun gas rising to a potentially explosive level.

8.2 Purging shall dilute the gas as close to the source as possible to prevent the concentration in all other areas of the installation exceeding 80% of the Lower Limit of Aircraft Hazard (LLAH) (see Volume 1 Leaflet 710/4).

8.3 The installation shall be capable of withstanding, without damage, the effects of any localised ignition that may occur at concentrations below the LLAH.

8.4 To minimise the risk of gas ignition any electrical equipment located in the vicinity of the gun installation shall be explosion proof.

8.5 HEATING AND COOLING

8.5.1 The temperature conditions in the gun and ammunition compartments shall permit the rotorcraft to utilise its full flight envelope without restrictions caused by exceeding the max/min. permissible temperatures of the gun and ammunition.

8.6 CONTAMINATION

8.6.1 With the exception of the designed openings in the installation, the gun and ammunition compartments shall be sealed to prevent the ingress of contaminants, particularly when the rotorcraft is on or near the ground.

9 AMMUNITION CONTAINERS

9.1 The design and position of the containers, in relation to the gun shall be such as to minimise friction and inertia of moving ammunition belts.

9.2 In a fully loaded container there shall be sufficient space between the top of the ammunition and the lid of the container to allow the ammunition belt to be withdrawn freely.

9.3 The position of the ammunition shall be indicated on the ammunition containers in accordance with Volume 1 Leaflet 710/1, para 3.3.

9.4 Provision shall be made for the drainage of fluids from the containers.

9.5 Ammunition containers shall be designed with a blast relief mechanism which will operate in the event of an explosion in the container.

10 AMMUNITION FEED

10.1 The design of the feed chute shall conform to the Interface Control Document and, in addition:

(i) the chute shall be as short as possible,

(ii) small radii bends shall be avoided,

(iii) changes of direction shall be minimal,

(iv) friction shall be minimised.

10.2 Access shall be provided in the chutes for loading and unloading and making and breaking of belts.

11 EXPENDED CARTRIDGE CASES AND LINKS

11.1 The gun installation shall be such that empty cartridge cases or complete rounds and links, when not collected, cannot cause damage to the rotorcraft or stores in any configuration of the rotorcraft. When empty cartridge cases or complete rounds and links are collected their collection shall not prejudice other requirements.

11.2 The cartridge case ejection tube/system shall be capable of passing complete rounds.

11.3 Collected cartridge cases, links and complete rounds shall be readily removable by personnel on the ground.

12 SERVICING

12.1 The design aim shall be to eliminate routine servicing of the gun installation during the life of the rotorcraft.

12.2 The minimum of equipment shall be required for servicing and replenishment. Whenever possible only in-service equipment shall be required.

12.3 Adequate access shall be provided to enable all servicing operations on the installation to be carried out at all times particularly when the operator is wearing protective clothing appropriate to the environment or task.

12.4 Where necessary provision shall be made for hoisting guns and loaded ammunition tanks to their installed positions in the rotorcraft.

12.5 It shall be possible to remove and replace guns without affecting harmonisation.

13 TESTING

13.1 The contractor shall demonstrate on a working rig of the gun installation and by ground and air firing trials on a development rotorcraft that the gun installation functions satisfactorily and meets the requirements of the Rotorcraft Specification and paras 1-12 above.

14 CROSS REFERENCES

14.1 A number of requirements directly related to the installation of guns appear elsewhere in this publication and the most important of these are listed in Table 1 (see also the Alphabetical Index).

TABLE 1

LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	Paragraph	Subject
100	10	Alignment of directionally sensitive equipment and weapons
107	2.2.1	Control column
107	Table 9	Armament controls
407		Protective Treatment
710		Armament installations
712	6.3.1	Fumes and vapour seals
713		Magnetic compass installations
717		Protection from the effects of nuclear explosion
802	1.3 and 1.6.1	Routine Servicing and Turn Round
1014		Flight testing of Armament Installations

Leaflet	Paragraph	Subject
Volume 1		
1014/1		Armament installations Fixed guns

LEAFLET 709/0

GUN INSTALLATIONS

REFERENCE PAGE

DEF STAN 00-970 Volume 1 Leaflets

710/1	..	..	General Recommendations
710/2	..	..	Gun Blast Loads Generated by Gun Installations
710/ 3	..	..	Sources of Gun Data for Installation Design
710/4	..	..	Gun Gas Concentrations
710/5	..	..	Gun Blast: The Effect of Gun Firing on Turbine Engines

RAE TECHNICAL REPORTS

72216	..	..	Evaluation of the French AME M621 Model F1 20mm Gun
72163	..	..	Ground Trials on the Oerlikon 304 RK 30mm gun

RAE TECHNICAL NOTES

Arm 589	..	..	The reduction of blast pressures from Aden guns by the use of obstructions in the path of gun gases
Structures 890		..	Blast from moving guns

RARDE TECHNICAL MEMORANDA

28/70	..	..	A study of gun blast in relation to that from a moving explosion
29/70	..	..	Loads on surfaces due to gun blast
34/ 72	..	..	Experiments on gun blast shields to reduce impulsive loads on nearby surfaces
17/74	..	..	A theoretical model of the blast from stationary and moving guns

Note: The foregoing documents are government sponsored; the list is not exhaustive and additional information may be available from Director, RAE and other sources

CHAPTER 710

ARMAMENT INSTALLATIONS

1 INTRODUCTION

1.1 This chapter states the design requirements for the installation of armament equipment in all rotorcraft. Armament Equipment, for the purpose of these requirements, comprises the weapon installations, associated armament equipment and electrical circuits concerned with monitoring, control, and release or firing of all stores using armament release equipment. Paras 9.9.3 and 13.1 refer to the relationship between the armament installation and Nav/Attack system. There are additional specific requirements for gun installations and these are given in Chapter 709.

1.2 All proposed rotorcraft weapon systems shall be approved by D.A. Arm MOD (PE) or the appropriate weapon system approving authority.

1.3 When weapon recording cameras, tone control and training aids are incorporated in the design of armament installations, there shall be no degradation of safety and reliability of the armament systems.

2 IN-SERVICE EQUIPMENT

2.1 Existing in-service equipment shall be utilised wherever possible provided the system performance is not prejudiced. All departures from the use of such equipment shall be approved by D.A. Arm.

3 STRENGTH AND ENVIRONMENTAL CONDITIONS

3.1 The rotorcraft installation, including attachments and associated structure, shall satisfy the strength and stiffness requirements for the rotorcraft as a whole under the conditions for carriage, operation, release and jettison of stores as quoted in the Rotorcraft Specification. Account shall be taken of rotorcraft taxying and any requirements for landing with stores fitted. This also applies to parent carriers and pylons.

3.2 The armament system shall function satisfactorily under the environmental conditions quoted in Chapter 101, or as modified in the Rotorcraft Specification.

4 DIMENSIONS OF STORES TO BE CARRIED

4.1 The dimensions assumed for stores in the design of the installation shall be in accordance with the relevant Aircraft Equipment Installation Information (AEII). Provision shall be made for the complete range of stores specified for the rotorcraft, the need for special adaptors being kept to a minimum.

5 ATTACHMENT OF STORE TO ROTORCRAFT

5.1 WEIGHT CLASS

5.1.1 At each store station the number of attachment points and their spacing shall be determined from Armament Design Memorandum 24 (Arm.D.M.24) using the maximum disposable load to determine the weight class.

5.2 RELEASE UNITS

5.2.1 Normally each store station shall be fitted with a release unit which, depending on the type of store to be carried and the role of the rotorcraft shall be either an Ejector Release Unit (ERU) or an Electromagnetic Release Unit (EMRU). The type of release unit fitted shall meet the release conditions quoted in the Rotorcraft Specification.

5.2.2 The release unit shall:

(i) fulfil the requirements of Arm. D.M.24,

(ii) satisfy the requirements of Defence Standard 13-8 (and 13-91 where applicable) with respect to the interface between the release unit and the store,

(iii) be equipped with mechanical ground safety lock, provision where applicable being made in the rotorcraft for stowage of a locking pin. It shall not be possible to lock the release unit unless it is in the cocked condition. It shall not be possible to remove the lock following:

(a) inadvertent firing of the cartridges of an ERU, until the gas pressure has fallen to a safe value,

(b) inadvertent release of the hook mechanism actuators of an EMRU, until the actuators have been reset to the cocked position,

(iv) when fitted with a Bomb Release Safety Lock (BRSL), be provided with the means to visually confirm the position of the BRSL in the locked position and with a safe means for identifying any inadvertent operation of the release mechanism with the BRSL in the locked position,

(v) when an ERU, be provided with a means for controlling the ejection and reaction loads.

5.2.3 In order to meet the requirements of para 9.9.1(i) each release unit shall be fitted with duplicate means of operating the linkage (and BRSL when fitted).

5.2.4 Indication shall be provided that the release unit linkage is correctly cocked.

5.2.5 Operation of the release unit mechanism shall:

(i) break the electrical release circuits,

(ii) activate the fuzing circuits (i.e., Supplementary Safety Devices SSD),

(iii) provide 'store on station' information as required.

5.3 BOMB RELEASE SAFETY LOCKS

5.3.1 A Bomb Release Safety Lock (BRSL) shall be fitted to each release unit which carries a nuclear store. A release unit which indirectly carries a nuclear store shall be provided with an in-flight safety locking pin (see para 5.5.1).

5.3.2 The BRST, shall:

(i) be engineered so that a single functional failure cannot prevent removal or engagement of the lock in flight,

(ii) when in the locked position prevent release of the store,

(iii) when in the locked position break the firing lines to cartridge operated release units,

(iv) when in the locked position, remain locked following:

(a) inadvertent firing of the cartridges of an ERU, until the gas pressure has fallen to a safe value,

(b) inadvertent release of the lock mechanism actuators of an EMRU, until the actuators have been reset to the cocked position,

(v) be capable of being moved into the locked or unlocked position in flight,

(vi) not be damaged if an attempt is made to operate it with the release hooks open.

5.3.3 Indication of the lock position shall be presented to the aircrew. The indicator shall show when the BRSL is finally "Locked" and completely "Unlocked", and shall be so engineered that no single failure within the system shall make the indicator inoperative or give an ambiguous or incorrect indication. The requirements of Chapter 105 para 13.4.2 shall also be observed.

5.3.4 The BRSL control system, including all associated electrical circuits, shall be completely isolated from and independent of all other weapon and rotorcraft systems except as provided for in para 9.4.3.

Note: The BRSL supplies shall be taken from the main armament feeder bars.

5.3.5 Where several BRSL's are fitted for the multiple carriage of nuclear stores, each BRSL shall have its own control and position indicator systems.

5.4 RELEASE CONTROLS

5.4.1 All the controls of both the release unit and the BRSL shall be so designed that a deliberate action is required to operate them and be so arranged and marked that they cannot be confused with each other, or with any other controls.

5.4.2 Operation of the release unit controls and the BRSL controls in the wrong sequence shall not prevent subsequent release of the store.

5.5 MULTI-STORE CARRIERS, SPECIALISED CARRIERS, ADAPTORS AND LAUNCHERS

5.5.1 Such carriers may have to be fitted to parent carriers, and shall meet the requirements of Arm. D.M.24. These carriers shall be provided with strong points for store loading, pull-off lanyards, arming wires to Defence Standard 13-91, and shall satisfy the requirements of para 11, when required by the armament installation. Carriers and adaptors for nuclear weapon release shall be provided with and have stowage for, an in-flight safety locking pin for inhibition of the parent carrier release unit.

5.5.2 It shall be possible to attach these carriers to the release units normally fitted at the store station.

5.5.3 Release units fitted to these carriers shall satisfy all the requirements of para 5.2 and, if appropriate, para 5.3.

5.5.4 The design of Multi-store carriers shall ensure that the deflections due to ERU reaction loads do not introduce release disturbance or weapon aiming problems.

5.5.5 There may be considerable advantages if Multi-store carriers, specialised carriers, adaptors and launchers can be light enough and produce insufficient drag to make it unnecessary to jettison them once the stores they carry have been jettisoned, provided this can be done without prejudice to the strength and crash landing requirements (see also Leaflet 710/2).

5.6 POSITION OF STORE STATIONS

5.6.1 The position of the stations for any of the specified combination of stores shall be chosen to minimise undesirable rotorcraft trim changes following store release.

5.6.2 The attitude and position of the loaded store in relation to the rotorcraft shall be chosen taking all the following into consideration:

(i) The overall drag of the rotorcraft when fitted with the appropriate store combinations.

(ii) The loads on the store due to aerodynamic forces induced by the rotorcraft, any aero-elastic implications for the rotorcraft and any aero-elastic implications for the store from the rotorcraft, pylon or carrier.

(iii) Any changes in aerodynamic loading on the store when it moves out of the rotorcraft field of influence after release.

(iv) The release and aiming requirements of the store as determined by the method of release.

5.6.3 The positions of carriers shall be such that any specified combination of stores may be carried with the minimum of role-change operations.

5.6.4 The position of each station shall be such that when a store is fired or released it falls, is ejected or propelled clear of the rotorcraft, adjacent stores and equipment. This requirement shall be met for all the release modes defined in the Rotorcraft Specification.

5.6.5 The positions of the store stations shall be chosen to avoid, as far as possible, high temperature efflux from engines, rockets or guided missiles impinging upon other stores and release systems. In addition, the positions of the store stations shall ensure that contamination of the stores and the release systems by engine exhaust, fuel, oil or any substance which could adversely affect the armament system is minimal. If necessary, protection shall be provided.

5.6.6 The positions of forward-firing weapon systems shall be chosen to avoid as far as possible any adverse effects upon the rotorcraft engines (see Leaflet 710/3 and Volume 1 Leaflet 710/5).

5.6.7 The firing of stores such as rockets or guided weapons shall not affect the operational function or safety of the rotorcraft (see Leaflet 710/3 and Volume 1 Leaflet 710/5). Where applicable the rotorcraft structure shall be designed to withstand the effect of blast and debris from the weapons. Precautions shall be taken to minimise the fire risk and the effects of high velocity gas streams. (Information regarding blast and fire risk may be obtained from Director RAE).

Note: The alignment requirements for directionally sensitive stores are stated in Chapter 100 para 10.

6 FUZING CONTROL SAFETY

6.1 Arming initiation of the store shall not be possible until separation from the release unit or launcher is irrevocable, although the store may still be connected to the rotorcraft by any of the devices listed in para 6. 2. Release or arming of internally carried stores shall not be possible until the weapon bay doors are fully open.

6.2 Installation of static lines, umbilicals, LEFAs (leads electrical fuze, arming), CAFAs (cable assemblies fuze arming), shear wire assemblies and lanyards between rotorcraft and store shall comply with the store AEII. They shall be laid out such that:

(i) it is not possible to make incorrect connections,

(ii) failure to make any connection properly shall not create a situation where damage to the rotorcraft can occur during carriage or after release of the store,

(iii) they are not crossed,

(iv) they do not become entangled with the lines to other stores during carriage or after release of the store,

(v) they do not become crossed or tangled upon release of the store,

(vi) they do not become entangled with, or cause damage to, the store or the rotorcraft after release of the store.

7 INSTALLATION

7.1 The layout of the complete installation shall be such that no part of the installation, loading apparatus or equipment will foul any other part of the installation, loading apparatus, equipment or the rotorcraft structure under the normal conditions of loading, operation or unloading. In the case of ship-borne rotorcraft these requirements shall be met under the conditions for the securing of naval rotorcraft given in Chapter 309, with the rotorcraft heading in any direction.

8 ACCESSIBILITY

8.1 The layout of the complete installation shall be such that it is possible to carry out, easily and quickly, all necessary operations including, the fitting and removal of loading apparatus not carried in the rotorcraft. In particular, with stores loaded there shall be good accessibility to ERU cartridges, release unit cocking levers and test plungers, crutches, stabilizers, throttles, BRSL and release unit indicators, ground safety pins, fuzing units, and for connecting plugs to their sockets.

8.2 All parts of the armament installation requiring inspection and servicing shall be readily accessible.

8.3 It shall be possible to carry out all servicing and inspection wearing the appropriate protective clothing.

9 ARMAMENT ELECTRICAL INSTALLATIONS

9.1 DEFINITIONS

9.1.1 For the purpose of this paragraph the Armament Electrical Installation is defined as comprising those electrical circuits concerned with the carriage, presetting, monitoring, fuzing , arming, firing, release and jettison of weapons, pyrotechnics and other armament stores (but see also paras 9.9.3 and 13.1).

9.1.2 In the context of the requirements of paras 9.9.1, 9.10.8, 9.10.9, 9.11.1 a failure is defined as the inability of a component to operate in the defined manner, i.e. a functional failure.

9.2 GENERAL

9.2.1 The armament electrical circuits associated with both conventional and nuclear weapons shall not be integrated unless this integration is approved by

D.A.Arm.

9.2.2 The wiring of rotorcraft installations designed specifically for nuclear weapons shall not be used for non nuclear stores unless this can be done without modification to, or degradation of the existing nuclear system and provided that approval is obtained from D.A. Arm.

9.2.3 Incorrect connection of the release and fuzing systems shall be mechanically and electrically impossible.

9.3 E.M.C., E.M.P., LIGHTNING AND ELECTROSTATIC HAZARDS[1]

9.3.1 Installations shall be engineered to minimise the risk of malfunction arising from electromagnetic interference, either generated by equipment within the rotorcraft or resulting from external RF emissions. In particular, it is necessary to guard against the possibility of explosive devices being fired inadvertently, and also to ensure that weapon control circuits are not damaged or deranged in any way by such interference. Additional steps shall be taken to ensure that armament installations do not create similar interference in other systems of the rotorcraft. The interrelation of Armament Systems and other rotorcraft systems and the achievement of an electromechanically compatible rotorcraft is the responsibility of the rotorcraft constructor. Responsibility for co-ordinating all EMC activities within the Controllerate of Aircraft rests with the Rotorcraft Project Director to whom reference should also be made at the beginning of design. Detailed advice regarding armament installations is contained in AvP.118, and installations that incorporate electro-explosive devices (EEDS) shall be engineered to the requirements of O.B. PROC 41273. Without exception, however, the guidance of AD/A Arm 2 MOD(PE) shall be sought at the beginning of design.

9.3.2 The requirements for protection against the effect of nuclear explosions stated in Chapter 717 shall be satisfied if called up in the Rotorcraft Specification.

9.3.3 Installations shall be engineered to minimise the risk of malfunction arising from electrostatic discharges and lightning strikes to the rotorcraft. Particular attention shall be paid to bonding, and screening techniques. An electrical bond which meets the requirements of Chapter 708 shall be established between the hooks of release units and the airframe. Where a sub-carrier is fitted to a main carrier, this requirement shall be achieved between the sub-carrier release unit hooks and the airframe. It is the responsibility of the store designer to ensure that his store can be adequately bonded to its lugs, or to the airframe by an alternative method. All wiring shall be routed to run as far away as possible from non-conducting panels in the outer structure of the rotorcraft in order to avoid induced effects from magnetic flux fields associated with non-conducting apertures during a lightning strike. Detailed design requirements shall be considered in consultation with the Rotorcraft Project Director.

9.4 ARMAMENT ELECTRICAL POWER SUPPLIES

9.4.1 Armament electrical power supplies shall be subject to the control of a Master Armament Safety Switch as defined in para 9.5. Any permitted exceptions shall be detailed as design deviations during the preliminary design review stage of the development and approval by D.A. Arm shall be subject to an unequivocal demonstration that the incorporation of any such design deviations does not degrade safety.

9.4.2 Armament power supplies shall be duplicated and any failure shall be indicated to the aircrew on an appropriate warning panel.

9.4.3 Two armament feeder bars, separate from the main rotorcraft bus bars shall be provided, one for each channel of the duplicated supply. For rotorcraft with nuclear weapon capability, a separate pair of feeder bars shall be provided for the nuclear installation, unless D.A. Arm directs that the conventional and nuclear installations are to be integrated. When the armament installation requires A.C. supplies, separate A.C. feeder bars shall be provided as necessary.

9.4.4 The armament feeder bars shall be controlled and protected such that any electrical or mechanical failure associated with one feeder bar will not affect the operation of the others.

9.4.5 The main armament feeder bars shall supply the circuits concerned with fuzing, arming, firing, release and jettison.

9.4.6 Unless D.A Arm MOD(PE) directs otherwise, auxiliary armament feeder bars shall be provided for armament system electronic logic supplies, 'store on station' circuits, weapon gyroscopes and armament heating purposes.

9.4.7 On all rotorcraft, one channel of each pair shall be supplied from the rotorcraft "essential" bus bar, so that in the event of load shedding following a generation failure, armament power supplies are maintained for vital services such as jettison and the making safe of certain weapons.

9.4.8 No non-armament circuits shall be connected directly or indirectly to armament feeder bars.

9.4.9 The armament feeder bars shall be situated to minimise distribution losses. They need not be adjacent to the main armament control panel.

9.4.10 Frame connections shall be in accordance with Chapter 706 para 4.5 and Volume 1 Leaflet 707/1. Each side of a duplicated circuit shall have its own independent frame connections and both shall be independent of the frame connections of other systems.

9.5 MASTER ARMAMENT SAFETY SWITCH

9.5.1 A master armament safety switch (M.A.S.S.) shall be provided in the cockpit, accessible to the pilot with harness locked, or to such other crew member as may be specified in the Rotorcraft Specification, which can isolate all armament fuzing and arming, firing, release and jettison circuits from all A.C. and D.C electrical supplies. For installations not utilising auxiliary armament supplies for logic, heating, 'store on station', gyroscopes and other agreed circuits (see paras 9.4.1 and 9.4.6) the MASS shall have two positions SAFE and LIVE. If auxiliary supplies are used a 3 position MASS shall be provided to isolate these supplies only in the SAFE position whereas the safety critical supplies shall be isolated in both the SAFE and STANDBY positions. The MASS can be used to break the supplies directly or to control relays for this purpose. Where relays are used, and to eliminate the possibility of a standing voltage to the relay shorting to one of the outputs such relays shall not normally be mounted in a junction box or armament unit which is directly connected with a release unit or EED. If this is unavoidable additional protection shall be provided in the box. The design of the MASS shall be approved by DA Arm MOD(PE).

9.5.2 The MASS shall be provided with a surround of not less than 6mm (¼in) width suitable for identification markings. The condition of the MASS in its SAFE and LIVE, and intermediate STANDBY position if applicable, shall be readily identifiable from its markings. The positions shall be both colour coded and marked by lettering. The positions shall be colour coded as follows: SAFE - green, STANDBY (if used) - amber, and LIVE - red. A tritium light source corresponding to the above colours shall be used if it is required to indicate any of the MASS positions during night operations. The tritium light source shall be compatible with current night vision goggles philosophy.

9.5.3 The MASS shall be readily identifiable by the colour and markings of its surround. This shall be matt white with matt red stripes at 45° not less than 3mm (1/8 in) or greater than 12mm (½in) wide. The width of the red stripes shall be half the width of the white bands. The preferred widths are 5mm (3/16 in) and 10mm (3/8 in) for red and white respectively. There shall be at least 2 red stripes separated by a white band.

9.5.4 Operation of the MASS to the LIVE position, from either SAFE or STANDBY as applicable to 2 or 3 position switches respectively, shall be guarded against unintentional operation by a 'gate' system such that it shall not be possible to select LIVE without a positive first action to allow passage through the 'gate'. Detent indexing shall be provided at the selectable positions of the MASS.

9.5.5 From an agreed safe location on the ground and external to the rotorcraft, it shall be ascertainable that the condition of the MASS is <u>not</u> live. The external indicator of MASS status shall, when required to uniquely identify the SAFE and STANDBY positions, show the colours green and amber respectively. In all other circumstances the SAFE position shall be identified using the colour green. The external safe location shall be ratified at the armament mock-up conference.

9.5.6 Where more than one position of the MASS is required to energise armament system circuits, the fuzing and arming, firing, release and jettison circuits shall not be made live until the MASS position furthest from SAFE is reached. Movement of the switch to this LIVE position shall be controlled by a detent system and guarded by a 'gate'. When moving the switch from the STANDBY to LIVE position, power supplies shall not be interrupted.

9.6 AIRFRAME, PYLON AND CARRIER CIRCUITS

9.6.1 All cables shall be coloured red to provide unique identification of the armament system. Armament electrical wires and cables shall be run in protective encasements (e.g., ducting, conduit or sleeving). Encasements shall be coloured red or be sufficiently transparent for the colour of the enclosed cables to be easily identified.

9.6.2 All wiring runs external to the rotorcraft shall, without exception, be adequately protected by encasement. Where camouflage is necessary, marking may be by 2 red lines of 2mm width, 180° apart running along each cable.

9.6.3 In all rotorcraft of the same type, mark, role and modification standard, the run of the cables of all armament circuits shall be identical.

9.6.4 With the exception of any armament signal lines which are covered by para 9.6.5, the armament wiring shall not be formed into cable assemblies with wires that are not associated with armament circuits (see Leaflet 710/1 para 3.3). Services other than armament services shall not be routed through armament junction boxes, terminal blocks or connectors.

9.6.5 Where digital data transmission systems form part of the armament control installation, it may be necessary to run the signal lines separately from the main armament cable runs, to minimise electromagnetic interference. If there are adequate built-in test facilities so that the armament installation can be tested easily and quickly prior to loading, without the use of any additional equipment, then the armament signal lines may be run in looms with the signal lines for other systems, and may share the same connectors. Before adopting such a system however, the designer shall satisfy D.A. Arm that the test facilities are adequate, and that the other circuits in the loom can in no way impair the safety and reliability of the armament installation. If these conditions cannot be satisfied, the armament signal lines shall be run separately from all other circuits, adequately identified and protected against damage and be routed through connectors which carry only armament signal circuits (see Leaflet 710/1 para 3.3).

9.6.6 Cable terminations in connector systems that control release lines or any form of electro-explosive device (E.E.D.) shall not occupy pin or socket connections adjacent to pins or sockets carrying standing voltages. This principle applies equally to terminal blocks and other types of connector. It is preferable to use an entirely separate connector for the sensitive circuits, and to run the associated wiring in separate looms where this is possible. The term standing voltages includes not only those which are present throughout the flight, but also those which are switched on sometime before release or firing is intended, for example, fuzing or station selection supplies.

9.6.7 In-line splices shall not normally be used. Where their use is unavoidable the approval of D.A. Arm shall be obtained. The location of these splices shall be clearly defined in the rotorcraft drawings and Air Publications.

9.6.8 Environmental protection and stowage shall be provided for unmated connectors, sockets and protection caps.

9.6.9 Circuit connections shall be of a type approved by D.A. Arm.

9.6.10 When the use of two or more connectors of the same size in the same location is unavoidable, shell orientated connectors of different orientation shall be used.

9.6.11 The armament power distribution circuits shall be so designed that the supply voltage at the input to each armament unit is maintained within the voltage limits specified for its correct operation. A circuit analysis shall be carried out to prove that this condition will be satisfied when:

(i) the main generating system is supplying the non-armament load appropriate to the moment of release,

(ii) the pulse loads at release are superimposed on the full steady state armament load,

(iii) partial failure of the generating system has occurred giving rise to conditions under which the Rotorcraft Specification requires the rotorcraft to complete its mission.

9.6.12 In addition to the above requirements, it shall be possible to operate the jettison circuits correctly despite a complete failure of the normal generating system.

9.6.13 In addition to satisfying the electrical requirements, the mechanical strength of the wire shall also be considered. Where small numbers of wires (4 or less) are run independently from or break out of the main cable runs, they are particularly vulnerable and a larger gauge of wire shall be used and protection against damage and snagging shall be provided. Wires smaller than 0.556mm^2 (cable size 20) shall not be used.

9.7 CONTROL HANDLE AND CONTROL COLUMN CIRCUITS

9.7.1 Armament circuits in control handles and control columns shall be engineered to prevent faults leading to inadvertent release or jettison. The close proximity of standing and switched voltages for armament and non-armament services, in these locations, makes it necessary to ensure that the armament circuits are properly segregated from all other services, particularly at the terminal arrangements on switches and other electrical components.

9.8 SWITCHES IN NUCLEAR WEAPON CIRCUITS

9.8.1 All switches shall be gated or guarded, the former being preferred for use in the cockpit.

9.8.2 To preserve the integrity of the electrical system following servicing, and to maintain the quiescent operational state of its components, provision shall be made for certain switches to be fitted with an irreversible tell-tale device capable of being fitted with a 'seal' to indicate that these switches have been moved from the 'off' or normal 'rest' position. (Chapter 105 para 11.6 does not apply to nuclear systems).

9.8.3 The tell-tale device shall be such that it will not unduly impede the normal operation of the switch, and shall be restorable only by servicing procedure once the switch has been moved. It shall be possible to restore the tell-tale whilst wearing appropriate protective clothing (foul weather, NBC, or arctic).

9.8.4 The choice of switches to be fitted with a tell-tale will depend upon circuit design, but it is preferable to select switches which isolate sub-systems from their power supplies. Proposals shall be submitted to D.A. Arm and agreement will be reached with the A & AEE, the Ordnance Board and appropriate Service and R & D Authorities.

9.8.5 At least one switch in each of the following sub-systems shall be fitted with a tell-tale:

(i) Fuzing and/or Arming.

(ii) Bomb Release Safety Lock.

(iii) Weapon Release.

9.9 RELEASE SYSTEM

9.9.1 The weapon release system and, when weapons are carried internally, the system for operating the weapon bay doors, shall be such that a single failure can neither:

(i) prevent release of the store(s) when required, nor

(ii) result in inadvertent release of the store(s)

9.9.2 If it becomes necessary to release weapons in a pre-determined order to ensure that no dangerous asymmetric or out-of-trim conditions occur, then the weapon control system shall achieve this order automatically. The system shall also automatically take into account the possibility of a failure to release a weapon in a "stick", without the pilot having to intervene until after the attack manoeuvre has been completed and the rotorcraft has recovered to a safe height and speed. An indication shall be provided to the aircrew if the weapon or weapons selected for release cannot be released.

9.9.3 When the weapon release signal is derived from a NAV/ATTACK system not subject to the requirements of Chapter 710, the requirements of this chapter shall apply to the armament release circuits up to the interface with the NAV/ATTACK system.

9.9.4 A manual release mode shall be provided for use in the event of failure of any NAV/ATTACK system.

9.9.5 In multi-crew rotorcraft no one crew member in his normal position shall have access to all the controls governing the release of nuclear weapons.

9.10 JETTISON SYSTEMS

9.10.1 Jettison facilities shall be provided in accordance with Chapter 100 para 15 and with the Rotorcraft Specification.

9.10.2 When armament and non-armament jettison systems are wholly or partly integrated, there shall be no degradation of safety and reliability of either system.

9.10.3 The initiation of jettison shall be as simple as possible and involve the minimum number of control actions on the part of the aircrew where a safe order of jettison is required this must be predetermined and automatic.

9.10.4 Nuclear stores shall be excluded from the jettison facility.

9.10.5 The jettison control(s) shall be independent of the normal weapon release control(s). In the case of mixed nuclear and conventional loads the jettison of nuclear stores shall be automatically inhibited (see Leaflet 710/2 paras 4.4 and 4.5). Provision shall be made for the following 2 cases:

(i) Emergency Jettison; i.e. the jettison of all non-nuclear stores as rapidly as possible, in a safe condition (unless otherwise specified) and without danger to the rotorcraft.

(ii) Selective Jettison; i.e. the jettison of individual non-nuclear stores, or groups of stores, in a safe or live condition, as selected.

9.10.6 Emergency Jettison shall always be under the control of the pilot and the controls shall be positioned in accordance with Chapter 107 Table 9.

9.10.7 Isolation of the jettison circuits from the electrical power source shall be provided by the M.A.S.S (see para 9.5.1).

9.10.8 Jettison systems shall be so engineered that no single functional failure can cause stores to be jettisoned inadvertently or prevent jettison when required.

9.10.9 No single functional failure shall cause jettisoned stores to be live when selected safe or safe when selected live.

9.11 FUZING CONTROL SYSTEM

9.11.1 The weapon fuzing control system in the rotorcraft shall be such that no single failure can:

(i) prevent the weapon being released live and in the correct condition, when required,

(ii) result in inadvertent arming of the weapon before release,

(iii) prevent the weapon being made safe after having been selected live.

9.12 CIRCUIT BREAKS

9.12.1 There shall be not less than 2 independently operated breaks in each fuzing circuit and each release circuit so that if either break alone closes to the "on" position the weapon will not be fuzed or released. These 2 breaks are additional to the M.A.S.S. required in para 9.5. The M.A.S.S. is primarily intended to ensure safety when the rotorcraft is on the ground.

9.12.2 One of these circuit breaks will be controlled by the release switch. For non-nuclear weapons the second break (sometimes called the Late Arm Switch) shall be positioned so that it can be closed during the final stages of an attack to minimise the consequences of a short circuit fault across the release switch. For rotorcraft where the pilot controls the weapon release, the second break in the circuit shall be closed automatically when the guard on the release switch is raised. For other rotorcraft and for nuclear weapons, the arrangements shall be approved by D.A. Arm at an early stage in the design. The power supply for release and non-nuclear fuzing shall not be switched to the pylons until the guard on the release switch is raised or the Late Arm Switch is operated. In the auto mode the release switch/button normally acts as an enabling device or commit button thereby allowing the weapon aiming computer to initiate release pulses which may be derived from the main computer. If at any time this switch is opened, store release is interrupted.

9.12.3 The fuzing circuit for nuclear weapons shall be in accordance with the relevant RAE Specification for Aircraft Monitoring and Control Systems. In nuclear weapon release circuits, the 2 breaks shall be in addition to that provided by para 5.3.2 (iii).

9.13 GUN, ROCKET AND GUIDED WEAPON FIRING CIRCUITS

9.13.1 In the event that engine control modulation (e.g. fuel dip, bleed, nozzle area change) is deemed desirable to minimise the adverse consequences of weapon wake ingestion should it occur, adequate provision in the weapon firing circuit for delaying weapon release or gun firing to allow implementation of the engine controlled modulation shall be provided. If flight trials prove that such a signal and/or delay in weapon firing circuit are unnecessary, then they can be omitted from the production weapon control system as appropriate (see also Leaflet 710/3 and Volume 1 Leaflet 710/5). When considering the provision of a delay in gun firing circuits, designers should be aware that a delay between trigger press and gun firing in excess of 0.1 seconds is operationally unacceptable.

9.13.2 Where the gun bay purging system employs an electrically operated inlet scoop, the scoop actuator is to be initiated by a switch on the last safety break (trigger safety flap) and not the trigger itself to ensure that full purging flow is established before firing commences.

Note: Para 9 should be read in conjunction with Chapters 706 and 708.

10 PROVISION FOR LOADING STORES

10.1 The store installation and associated parts of the rotorcraft structure shall be designed so that any of the specified stores or combinations of stores can be loaded easily and quickly, by personnel wearing the appropriate protective clothing.

10.2 Adequate clearance shall be provided for loading and unloading stores with the rotorcraft at maximum all up weight and its alighting gear compressed and tyres deflated in the most adverse manner, avoiding the need for accurate manoeuvring of the appropriate loading trolleys and loading equipment and making allowance for uneven surface conditions (see also Volume 1 Chapter 311).

10.3 Where specified, provision shall be made in the rotorcraft for hoisting stores. Wherever practicable the design shall enable existing in-service equipment to be used.

10.4 Whenever practical, it shall be possible to load and unload stores without the use of tools. Where this is impracticable, the installation shall be designed so that in-service tools can be used whenever possible.

10.5 When the rotorcraft has folding components, (e.g. rotor, fuselage, tail), to meet specified limiting dimensions, it shall be possible to fold and spread these components with stores loaded. It shall also be possible to load and unload stores with these components folded.

10.6 When rocket launchers are specified, it shall be possible to re-load individual rockets without changing the launcher.

10.7 It shall be possible to re-load the guns without having to unload any armament or other stores.

11 STRONG POINTS FOR STATIC LINES OF PARACHUTE RETARDED STORES

11.1 Readily accessible strong points shall be provided at store stations for the attachment of static lines for parachutes.

11.2 The strong points shall take the form of a fork end with a quick-release pin of not less than 6.35mm (¼in) dia. The strong point and adjacent structure shall not fail under a load of 1.5 times the specified maximum working load acting in any direction within a cone of semi-angle 30° with its apex at the strong point and its axis perpendicular to the fuselage datum.

12 CREW STATION CONTROLS

12.1 The controls to be fitted at crew stations shall be in accordance with Chapter 107.

13 SIGHTING SYSTEMS

13.1 There shall be a clearly defined interface between the Nav/Attack and Weapons Systems and the arrangement shall be approved by D.A. Arm.

13.2 The Nav/Attack and Weapon Systems Designers shall agree on the type and form of signals required.

14 WEAPON BAY ILLUMINATION

14.1 One or more lights shall be provided in the weapon bay to give illumination for loading stores at night. The lights shall be arranged to give general illumination round the store and specific illumination of store controls and attachment points. The lights shall be controlled by a switch accessible from the ground, and by movement of the weapon bay doors, so that opening or closing the doors will automatically extinguish the lights which shall remain off until manually selected on. The system shall be so engineered that no single functional failure can cause inadvertent illumination.

15 OPERATION OF WEAPON BAY DOORS

15.1 A positive means, under the control of the pilot, shall be provided for opening and closing the weapon bay doors in flight; spring operated doors are not acceptable.

15.2 Care shall be taken in the design stage to ensure that correct operation of the weapon bay doors will not be prejudiced by:

(i) deformation of the rotorcraft structure by external aerodynamic load,

(ii) jamming caused by deformation of the doors or surrounding structure,

(iii) incorrect alignment of the hinges under load,

(iv) environmental conditions.

15.3 Means shall be provided to prevent inadvertent opening of the doors under all possible flight conditions (see also Chapter 306).

15.4 MAXIMUM TIMES FOR OPERATION

15.4.1 Unless aerodynamic, weapon aiming, or other considerations dictate otherwise, the operation of opening or closing the doors shall be completed within 6 seconds from initiation of the opening or closing signal.

15.5 EMERGENCY OPERATION

15.5.1 An emergency method of opening and closing the doors shall be provided.

15.5.2 The emergency system shall be independent of the main system and be operable regardless of any previous setting or failure of the main system. Power for both systems shall be from independent sources.

16 SAFETY INTERLOCK

16.1 Means shall be provided to prevent release, arming or jettison of the stores until the weapon bay doors are fully open.

16.2 Means shall be provided to inhibit the firing of systems while any part of the rotorcraft (e.g. undercarriage), its equipment or other stores obstructs the line of fire or is likely to be damaged by blast or debris. In cases where a weapon is moved from a flight carriage position to a firing position, firing shall be inhibited until the weapon is correctly positioned. An 'over-ride' facility for ground testing may be required (see para 18).

17 TESTS

17.1 Wind tunnel and rig tests shall be made at an early stage to determine the suitability of the rotorcraft to meet the requirements of para 15.4.1.

17.2 On prototype rotorcraft (and on the first of all rotorcraft where a new installation is used, or use of special equipment has been approved) the Contractor, on completion of the installation shall subject it to such tests as considered necessary by D.A. Arm.

18 IN-SERVICE TEST REQUIREMENTS

18.1 Provision shall be made for meeting the user's general requirements for testing in service and in particular for NO VOLTS testing. The designer shall agree these test facilities with the NATEC special maintenance party or CSDE servicing project team at an early stage in the design (see Leaflet 800/1 para 1).

19 CROSS REFERENCES

19.1 A number of requirements directly related to the armament installation appear elsewhere in this publication and the most important of these are listed in Table 1 (see also the Alphabetical Index).

20 GENERAL SAFETY AND RELIABILITY

20.1 The requirements of Chapter 710 aim to ensure that the equipment has the degree of safety and reliability adequate for the carriage of weapons in rotorcraft. The requirements have been developed on the assumption that two failures are unlikely to occur on associated equipment during the period between the departure of the serviceable rotorcraft from its base and its arrival at the target.

20.2 The second assumption is that structural failure of components can be virtually eliminated by careful procedure, e.g., by careful material control and careful inspection during manufacture. The design strength, i.e., the required strength to carry a particular store, is known or can be established by suitable testing; it is therefore possible to ensure adequate strength of all the components in a given installation.

20.3 Once it has been accepted that structural failures are covered by the procedure described, the requirements of Chapter 710, para 9.9.1 can be met by preventing mishap occurring as a result of a single functional failure. This type of failure is difficult to deal with procedurally; although of course, good design is extremely important in minimising the occurrence of such failures.

TABLE 1: LIST OF OTHER IMPORTANT REQUIREMENTS

Chapter	Paragraph	Subject
100	10	Alignment of directionally sensitive weapons
100	15	Jettisoning of stores
101		Operation in various climatic conditions
103		Operational colouring and markings
105	13.4	Colour of illuminated indicators
107		Cockpit controls and instruments
306		Doors and locks
309		Picketing, tie-down and rapid securing systems
311 VOL 1		Ground clearance
706		Electrical installations
708		Bonding and screening
709		Gun installations
713		Magnetic Compass Installations
717		Protection from the effects of nuclear explosion
720		Air launched weapon installations
1014		Flight testing of armament installations
Leaflets 710/5 VOL 1		The effect of gun firing on turbine engines
800/1		General maintenance requirements

REFERENCES

Reference	ASCC Air Standard
1	20/16

LEAFLET 710/0

ARMAMENT INSTALLATIONS

REFERENCE PAGE

AGARD REPORTS

AGARD-AR-107 .. .. Drag and other aerodynamic effects of external stores

AGARDOGRAPH 202 .. Store separation

AIR PUBLICATIONS

AP110A-0001-1 .. .. Guide to armament air publications

A R C REPORTS

R and M 3438 .. .. Wind tunnel experiments on the flow over rectangular cavities at subsonic and transonic speeds

R and M 3503 .. .. Aerodynamic Loads on external stores. A review of experimental data and method of prediction

ARMAMENT DESIGN MEMORANDA

Arm DM24 .. .. Strength requirements for bombs and similar stores and their carriers, release mechanisms and ejector units for use in aircraft

DEFENCE STANDARDS (DEF STAN)

10-21 .. .. Aircraft ejector release units for conventional munitions (implements STANAG 3575)

13-8 .. .. Dimensional requirements of airborne stores (implements STANAG 3441 and 3558)

13-37 Pt 1 .. .. Airborne weapon fuzing systems (implements STANAG 3525)

13-91 .. .. Mechanical arming wire connections between airborne armament stores and associated suspension equipment (implements STANAG 3605)

59-36 Pt 2 .. .. Electronic components for defence purposes - procedure for selection and specification

59-71 Pt 1 .. .. Crimped electrical connectors

66-6 .. .. Ohm meter safety multi range battery powered

66-16 .. .. Test set safety voltage detection

MOD(PE) PUBLICATIONS

AvP 118	.. ..	Guide to electromagnetic compatibility in aircraft systems

ORDNANCE BOARD PROCEDURES (OB Procs)

41273	.. ..	Principles of design and use for electrical circuits incorporating explosive devices
41754	.. ..	Safety of fuzing systems. Design safety principles

RAE REPORTS

Aero 2511	.. ..	Low speed wind tunnel tests on the flow in bomb bays and its effect on drag and vibration
Tech Report 68086	.. ..	The measurement of fluctuating pressures in and behind the bomb bay of a Canberra aircraft

RAE SPECIFICATIONS

WE 659	.. ..	Design requirements for aircraft systems fitted with electro-initiated explosive devices

LEAFLET 710/1

ARMAMENT INSTALLATIONS

WEAPON RELEASE AND FUZING SYSTEMS; FACTORS INFLUENCING SAFETY AND RELIABILITY OF ELECTRICAL CIRCUITS

1 INTRODUCTION

1.1 The dangers associated with the carriage of weapons on rotorcraft demand that special attention be paid to the electrical release and fuzing circuits. The problems are accentuated on rotorcraft that can carry nuclear weapons. The aim must be that conditions can never arise which could result in accidental release of weapons, live or safe, or, on the other hand, in failure to release the weapons, live or safe as selected. This is the objective of the requirements of Chapter 710 para 9 and the purpose of this leaflet is to provide a guide to the design and engineering of armament electrical systems to meet these requirements, and to indicate acceptable practices. As stated in Chapter 710 the requirements apply to all rotorcraft.

2 TYPES OF FAILURE

2.1 The types of failure that need to be guarded against may be sub-divided as follows:

(i) Faults in equipment and components:

(a) those which make equipment operate inadvertently,

(b) those which prevent equipment operating,

(c) those which cause equipment to operate incorrectly.

(ii) Faults in cables and connectors:

(a) short circuits,

(b) open circuits,

(c) cross connections during assembly and servicing,

(d) low insulation resistance between circuits,

(e) high resistance connections,

(f) variation of insulation and contact resistance with environment and age.

(iii) Faults in power supplies:

(a) voltage below acceptable limits,

(b) voltage above acceptable limits,

(iv) Malfunctions from:

(a) electrical transients in excess of acceptable limits,

(b) induced currents,

(c) conducted and radiated interference.

3 AIRFRAME, PYLON AND CARRIER CIRCUITS

3.1 Each circuit of a duplicated system should be independent and connected to its own feeder bar, so that a failure of either circuit will leave the other fully operative.

3.2 Care should be taken to ensure that each side of a duplicated system (e.g. fuzing, bomb release safety locks (BRSL), release) is connected via separate plugs and sockets and that each side is connected to the appropriate feeder and independent frame connection (see Chapter 710, para 9.4.10). Circuits supplied from different feeders should not be run in the same loom.

3.3 To minimise the risk of failure by inadvertent damage or enemy action, duplicate services should be separately routed wherever practicable, subject to the limitations imposed by rotorcraft size and design. Note that Chapter 710 para 9.6.3 requires the run of the cables associated with the circuits for firing, fuzing, BRSL, release and jettison to be identical in all rotorcraft of the same type, mark, role and modification standard. When it becomes necessary to run a cable loom carrying armament circuits alongside other rotorcraft service looms, the length of runs in close proximity should be kept to a minimum and additional protection provided at the points where they touch. If however, general services circuits are encased, these may run alongside armament circuits in separate encasements without further protection.

3.4 It is desirable that cables controlling the weapon fuzing, BRSL and release systems should not be loomed together. Individual conduits may, if necessary, be laid up alongside one another provided there are no overriding requirements in the weapons specification.

3.5 Chapter 710 para 9.6.1 requires that cables are encased; however, exceptionally (subject to agreement of the Rotorcraft Project Director) encasement may not be necessary in the following instances:

(i) Where physical constraints imposed by the rotorcraft structure severely limit the space available, the wiring is well protected mechanically by the structure itself, and the area is free from the risk of contamination.

(ii) Where encasement would introduce a servicing penalty disproportionate to the protection provided.

The omission of encasement under the terms of this para constitutes a Design Deviation, any requests for which should be made to D.A. Arm in accordance with the Technical Procedures applicable to the rotorcraft.

3.6 Junction boxes used for armament wiring should meet the following requirements:

3.6.1 Internal wiring should be loomed and secured such that hinge points do not occur at cable terminations.

3.6.2 Cable end fittings should be of an approved type of crimped terminal.

3.6.3 Junction boxes and terminal blocks should be so designed and mounted that there is adequate protection against the ingress of fuel oils and moisture or other contaminants (see Volume 1 Leaflet 707/1, para 7).

3.6.4 The location of junction boxes and terminal blocks within the rotorcraft structure should be such that they are at all times easily accessible for servicing.

3.6.5 Ideally, when the M.A.S.S. is set to safe, there will be no power supplies to any armament junction box or unit, to avoid the possibilities of faults developing between the power input and the outputs. When this is not possible, for example when Bomb-on-Station or Logic supplies are required when the M.A.S.S. is set to safe, special precautions should be taken to ensure that these supplies cannot be short-circuited to the outputs from the junction box or unit. There is no objection to mounting the M.A.S.S. relays in a junction box which acts as a power distribution box, provided that the outputs to release units etc., are not subsequently routed through that box.

3.6.6 The wires carrying the standing, voltage supplies to the M.A.S.S. should not be formed into any loom which includes release or other sensitive circuits.

3.7 The size of the conductors in cable runs to individual store stations, in addition to having the required electrical characteristics, should be such that the cables have adequate mechanical strength in their operational environment. Voltage drop in distribution circuits should be within the limits specified in the Rotorcraft Specification or BS 3G 100 Part 3 as appropriate.

3.8 Particular care should be taken to protect cables, connectors, plugs and sockets from damage likely to result from adjacent moving parts and the movements of aircrew and servicing personnel.

3.9 Connector systems used for inter-connecting looms in the airframe should be of a type approved by D.A. Arm. Crimped cable connections should be employed. Glands should be used rather than bulkhead connectors to minimise the number of discontinuities in the circuits. Where plugs and sockets are used, these should be to the requirements of Defence Standard 59-56 (Pattern 602) or alternative types approved by D A. Arm.

3.10 Approved connector modules carrying armament services should be segregated from modules used for non-armament services. Where space and weight consideration make it impracticable to mount the armament modules in a separate frame, they should be kept apart from the non-armament modules by a spacer and clearly labelled "ARMAMENT".

3.11 Inter-connections between rotorcraft and carriers or pylons should be by connectors of types complying, with para 3.9 above. Construction of the interconnections should be in accordance with RAE Specification Arm 942 except that any plug and socket approved by D.A. Arm. may be used. Cable connections between airframe and carriers should be positioned and secured so as to prevent damage.

3.12 Care should be taken to ensure adequate flexibility of cable assemblies where physical movement takes place in flight or during servicing.

3.13 The number of connectors of all types in any one circuit should be kept to a minimum.

3.14 For non-nuclear weapons there should, ideally, be no standing voltage supplies in the pylon carrier or station until such time as the release switch guard is raised. Low voltage or current limited sources should be used for Bomb-on-Station signalling, Built In Test Equipment (BITE) and similar circuits so that in the event of a fault there will be insufficient power to operate the release unit. For Guided or other weapons, where it is not possible to meet these requirements, there must be complete physical separation between the circuits associated with these weapons and the normal release and fuzing circuits, and separate connectors should be provided for the two groups of circuits.

3.15 The use of components with exposed connection tags is to be avoided whenever possible.

LEAFLET 710/2

ARMAMENT INSTALLATIONS

JETTISON SYSTEMS

1 INTRODUCTION

1.1 Design features associated with any jettison facility can only be considered in conjunction with the particular Rotorcraft Specification. However, some features are common to most jettison systems and these are discussed in the following paragraphs.

2 NUCLEAR STORES

2.1 Chapter 710 para 9.10.4 requires that nuclear stores be excluded from the jettison system. It is desirable that the act of loading a nuclear weapon at any station should inhibit the jettison circuits at that station.

3 GUIDED WEAPONS

3.1 These are normally jettisoned safe unless otherwise specified, either by firing the motor of the weapon, by ejection or by gravity release.

3.2 Rail launched missiles can often be jettisoned by firing the rocket motor with the missile in a safe configuration. Alternatively it may be possible to jettison the missile and launcher as a single unit. For emergency jettison, the second method is likely to be the quickest and safest, but for selective jettison, there are advantages in being able to fire the missile as the first option, whilst retaining the ability to jettison the missile complete with launcher as a secondary method, should the rocket motor fail to fire.

4 EMERGENCY JETTISON

4.1 The emergency jettison of launchers/carriers alone or with pylons may be required, in addition to the separate release of the stores themselves. In such cases consideration should be given to the feasibility of jettisoning multi-store carriers complete with stores.

Note: It will be necessary to consider the consequences of jettisoning both fully and partly loaded carriers.

4.2 Chapter 710 para 9.10.5. (i) requires that the rotorcraft be cleared of all non-nuclear stores as rapidly as possible and without danger to the rotorcraft. It is therefore desirable that the stores, carriers and pylons where necessary, be jettisoned simultaneously. However, simultaneous jettison may endanger the rotorcraft for any one or all of the following reasons:

(i) The jettison of all stores together may result in aerodynamic disturbances leading to undesirable behaviour of the rotorcraft or even temporary loss of control.

(ii) Unacceptably high reaction loads may be generated in the rotorcraft structure.

(iii) The stores may contact the airframe after leaving the carriers.

(iv) The stores may collide with each other after leaving the carriers.

(v) Jettisoned pylons, and jettisoned carriers complete with stores, may behave violently after release due to their aerodynamic characteristics.

Consequently it may be necessary to release the stores (including carriers and pylons if appropriate) in a predetermined safe order. In this case control of the order of release must be automatic, and require no more than a single operation of the emergency jettison switch for complete jettison.

4.3 In some configurations it may be desirable to jettison the store complete with its carrier or launcher although release of the unloaded carrier or launcher may endanger the rotorcraft. In such cases means must be provided to inhibit automatically the jettison of the unloaded carrier or launcher.

4.4 In the case of internally carried stores operation of the emergency jettison control should release these stores (except nuclear weapons) only if the weapon bay doors are open. It is not considered necessary to provide automatic opening of the weapon bay doors specifically for emergency jettison, unless otherwise specified. If, however, an automatic door opening system is provided for normal operation this could be used for emergency jettison if necessary.

4.5 It is desirable that emergency jettison should be possible at any point within the flight envelope. However, in practice, there may be limitations and it is desirable that these be determined at an early stage and that D.A. Arm., be notified of any such limitations.

4.6 Considerable electrical power can be required to operate a comparatively large number of release units simultaneously. Consideration should therefore be given to the capacity of the rotorcraft emergency battery to ensure that this is sufficient to perform an emergency jettison in the event of a complete rotorcraft power generation system failure.

5 SELECTIVE JETTISON

5.1 The selective jettison facility may form part of the stores management system provided that the mandatory requirements are met. The operation may be controlled by the pilot or a second crew member, as appropriate.

5.2 As stated in para 4.1 there is sometimes a requirement to jettison carriers or carriers and pylons in addition to the stores themselves. In such cases, at any station carrying a nuclear store the jettison of the pylon and carrier, as well as the store itself, must be inhibited, as indicated in para 2.1.

LEAFLET 710/3

ARMAMENT INSTALLATIONS

THE EFFECT OF FIRING AIR WEAPONS ON THE BEHAVIOUR OF TURBINE ENGINED ROTORCRAFT

1 INTRODUCTION

1.1 This Leaflet discusses those factors which are known or suspected of causing engine malfunctioning when guns, rockets or guided weapons are fired. In addition to the information contained herein much of the content of Volume 1 Leaflet 710/5 is also applicable to the firing of rockets and guided missiles. It must be considered in conjunction with this leaflet.

1.2 Malfunctioning depends upon the type of weapon, its installation and proximity to the engine air intake system, the engine and its associated air intake and the flight and engine conditions under which the weapons are fired.

2 CAUSES OF MALFUNCTION

2.1 Engine malfunction is known to be caused by a multiplicity of factors which include the entry of hot gases from the rocket/propulsion motor efflux into the air intake system, the variations in total pressure at engine entry and the total temperature distortion at the engine entry. The problems are extremely complex and although the actual mechanism of the subsequent effect is not understood, the individual effects in a total complex set of effects are (see Volume 1 Leaflet 710/5 para 2.1).

2.2 There can be adverse effects due to the chemical composition of the weapon gases, (e.g., chemical reaction on engine components or aspiration of inert gases and see para 4.3 (v) on fouling of optical or IR measuring systems).

3 FLIGHT CONDITIONS

3.1 The flight conditions under which the weapons are fired (i.e., altitude, attitude, forward speed, throttle position, rotorcraft incidence, side slip and ambient temperature) affect the problem in various ways in that they:

(i) can have a critical influence on the expansion of the weapon gases and hence the characteristics of the pressure wave,

(ii) can have a critical effect on the efficiency of the air intake system directly and also indirectly by their influence on rotorcraft attitude, and

(iii) can have a critical effect on the engine compressor surge margin. (The surge margin at a particular engine speed although basically a function of engine design will vary with change in altitude, forward speed, ambient temperature and air intake performance).

3.2 In general, increasing altitude and decreasing speed will increase the likelihood of engine malfunctioning because the reduction of air inlet mass flow causes a higher proportion of weapon exhaust to engine airflow to be ingested.

3.3 Manoeuvres aggravate the problem in their effect on attitude and hence on air intake performance.

3.4 During supersonic flight conditions, blast waves from a fired weapon can modify instantaneously the organisation of the rotorcraft forebody/engine inlet airflow and cause engine malfunction.

4 ROCKET INSTALLATIONS

4.1 The remarks in this para apply equally to unguided rockets and to guided weapons.

4.2 The type of propellant used in a rocket motor, its rate of burning and the flight conditions under which it is launched are the important factors in determining the chemical composition, concentration, temperature gradient and pressure wave characteristics of the efflux gases.

4.3 Characteristics of the overall weapon design which can affect the engine problem are:

(i) acceleration - in determining the time the rocket efflux may act within the vicinity of the air intake system,

(ii) aerodynamic stability and/or guidance system - in determining the motion of the weapon in the early stages of its trajectory. (Any motion which causes the efflux to be directed towards the intake system is undesirable),

(iii) the number and size of rockets fired at any one time and firing sequence (i.e., ripple or salvo),

(iv) the positioning of the weapon installation in relation to the air intake system. (This is clearly a most critical factor),

(v) the base material of the propellant, for example it is known that an aluminium base can cause serious engine turbine blading contamination, deterioration, and fouling of optical and IR sensors.

5 GUN INSTALLATIONS

5.1 The type of propellant used in gun ammunition and flight conditions under which it is fired are important factors in determining the chemical composition, concentration and pressure wave characteristics of the discharge gases.

5.2 Characteristics which can effect the engine are:

(i) muzzle blast pressure - may create a shock wave which is critical to the intake airflow,

(ii) firing rate - the rate and rhythm of firing may be critical to the intake airflow,

(iii) the positioning of the gun and/or gas discharge port(s),

(iv) the composition of the propellant - it is known for example that use of phosphorous compounds as a flash suppressant can cause contamination of turbine blades and fouling of IR and optical sensors,

(v) vibration - excessive vibration may upset airframe mounted engine control systems to the extent that physical damage can occur which may then result in malfunctioning of the engine.

6 RECOMMENDATIONS

6.1 At this stage only broad recommendations can be made and every endeavour is being made to improve knowledge of the problem by intensive investigation.

6.2 The aim should be to separate the area of weapon disturbances from the engine air intake as far as possible, thus avoiding weapon effects.

6.3 For rocket propelled weapons, the maximum separation between weapons and air intakes should be provided. In addition, the initial trajectory should be studied in relation to the air intake system, so as to avoid directing the rocket efflux towards the intake if it can be avoided.

6.4 Where design necessitates a close relationship of the weapons and engine intakes, due consideration should be given to the fact that it will not be possible to design an engine with enough basic surge margin to cater for serious weapon wake ingestion, although the characteristics of the engine and the intake and fuel systems should be such as to minimise the effects of that ingestion.

6.5 An unambiguous statement of weapon firing/release operational envelope and sortie profile requirements must be prepared during the initial weapon system design study. It must for example include statements as to whether engine acceleration/deceleration, and rotorcraft incidence are relevant factors, and also whether the sortie profile combines the use of weapons and Air Sea Rescue.

CHAPTER 711

ICE PROTECTION

1 INTRODUCTION

1.1 This chapter contains the requirements for protecting rotorcraft against the accretion of ice, snow and slush.

1.2 This chapter applies to all rotorcraft irrespective of the extent of ice protection provided, which are required by the Rotorcraft Specification to enter, or operate in, icing conditions.

1.3 Although general requirements for engine ice protection are defined elsewhere (for example, in DEF STAN 00-971), requirements for engine and auxiliary air intake ice protection are contained in this chapter.

1.4 The atmospheric conditions in which ice accretion may occur, and their probable extent, are defined in Leaflet 711/2. If the Rotorcraft Specification requires operation at altitudes above 3000 m the appropriate icing standards shall be as defined therein.

2 OPERATIONAL REQUIREMENTS

2.1 The rotorcraft shall be capable of meeting the Service operational roles for the durations stipulated in the Rotorcraft Specification under the following conditions:

(i) the Continuous Maximum icing conditions,

(ii) the Periodic Maximum icing conditions,

(iii) the Mixed icing conditions, continuous,

(iv) the Mixed icing conditions, periodic,

(v) falling snow, continuous,

(vi) falling snow, periodic,

(vii) blowing or recirculating snow,

(viii) freezing fog,

(ix) freezing rain/drizzle,

(x) any additional conditions stipulated in the Rotorcraft Specification.

Notes: (1) See Leaflet 711/2 for the definition of conditions (i) to (ix).

(2) Condition (x) might include, for example, operation from or onto runways contaminated with ice, snow (both lying and hard-packed), slush or standing water.

2.2 In order to satisfy the requirements of this Chapter it shall be shown that, when the rotorcraft is operated in the stipulated conditions and for the required durations, there will be no hazard to the rotorcraft or its crew, and no unacceptable degradation in:-

(i) the performance of the rotorcraft or its systems,

(ii) the handling qualities of the rotorcraft,

(iii) the performance of weapon systems carried in or on the rotorcraft.

Note: Any degradation would be unacceptable if it resulted in the rotorcraft or its systems being unable to meet the requirements of the Rotorcraft Specification appropriate to flight in ice-forming conditions.

2.3 The flight envelope for flight in icing conditions shall not produce loads in the mechanical systems of the rotorcraft that exceed the design loads at Vne under non-icing conditions. An allowance may be required in the fatigue spectrum for the manoeuvre envelope to cater for flight in icing conditions, and the Design Authority shall evaluate the effects of icing on the flight manoeuvre loads. Any limitations in the flight envelope for flight in icing conditions shall not be more restrictive than permitted by the Rotorcraft Specification.

3 ICE PROTECTION SYSTEMS

3.1 BASIC SYSTEM TYPES

3.1.1 The two basic types of ice protection systems are:

(i) Anti-Icing: Systems used to prevent the formation of ice on critical parts of the rotorcraft surface. This is usually achieved by the continuous heating of the relevant parts, but use can also be made of freezing point depressant fluids in certain applications.

(ii) De-Icing: Systems used to periodically remove the ice from parts of the rotorcraft surface, before it reaches a size that could cause an unacceptable degradation of rotorcraft or system performance, or hazard the rotorcraft during shedding.

3.1.2 Anti-icing and de-icing systems can be used in combination.

3.2 ICE PROTECTION METHODS

3.2.1 A brief description of the various means of ice protection is given in Leaflet 711/3.

4 SYSTEM REQUIREMENTS - GENERAL

4.1 The protection system shall be capable of satisfactory operation throughout the rotorcraft flight envelope, unless otherwise stated in the Rotorcraft Specification.

4.2 The system shall be designed so far as is reasonable to provide symmetric protection of dynamic systems, eg rotors, in order that the problems associated with asymmetric ice accretion and/or shedding may be avoided. In designing to achieve this objective the effects of possible system failures shall be considered (see para 4.5).

Note: This is of particular importance in respect of main rotors where the asymmetric accretion or shedding of ice must not result in unacceptable levels of vibration or excessive structural stresses.

4.3 Notwithstanding the formation of ice, the rotorcraft services and auxiliary equipment, (e.g undercarriage, aerodynamic control surfaces, generators, rotor droop stops, pitch change links and flight instruments) that are required for safe flight, emergency operation, and landing, shall continue to function satisfactorily.

4.4 Ice protection power requirements shall be kept to the minimum necessary to achieve the required level of protection.

4.5 Back-up systems/power sources must be provided where failure of the protection system could lead to unacceptable flight safety problems or operating restrictions. On multi-engined rotorcraft where protection is provided by the engine, alternative sources must be provided to cater for the effect of engine failure.

4.6 Ice protection systems shall not introduce secondary ice accretion (run-back) unless it can be shown to be of no consequence. Specific attention shall be paid to the effect of runback on the rotor blades as this may cause a loss of aerodynamic performance or a degradation of the dynamics of the rotor due to the aft movement of the blade c.g.

4.7 When a de-icing system is used, the ice accretion formed during the de-icing off-time must not produce unacceptable rotorcraft or system performance degradation.

4.8 The ice protection systems, whether functioning or not, shall not:

(i) cause corrosion or deterioration of any part of the structure, or associated and adjacent systems,

(ii) cause unacceptable levels of vibration, loss of control power, or excessive structural stresses due to ice accretion on, or shedding from, the main or tail rotors.

4.9 The ice protection systems shall be so designed that toxic fluids or vapours can not enter the cabin in normal flight, or as a result of enemy action, to the extent required by the Rotorcraft Specification.

4.10 Ice protection systems for engine intakes, pilots' windscreens, air data probes and any other system requiring protection prior to take-off shall be designed for operation during start-up, manoeuvring on or near the ground and take-off as well as in flight.

4.11 The Design Authority must show that the protection system meets the Rotorcraft Specification by means of some combination of modelling, rig and tunnel testing, and flight tests in natural or simulated icing conditions. (See para 9 and Chapter 1006). Where computer models or simulated icing conditions are used to prove compliance, they must be supported by substantiation and/or validation evidence.

4.12 The manufacturer shall show, by analysis or test, that the performance of the ice protection system(s) will not be degraded to an unacceptable extent over the required life and predicted usage of the rotorcraft.

5 ICE PROTECTION COVERAGE

5.1 In determining the areas requiring protection the designer shall give particular consideration to the following:

(i) rotors, rotor hubs, associated control linkages, and blade-limiting stops,

(ii) aerofoil surfaces,

(iii) moving surfaces,

(iv) engine and auxiliary intakes,

(v) pitot and static heads and masts,

(vi) aerials, sensors and radomes,

(vii) weapons and weapon carriers,

(viii) transparencies,

(ix) vents and drains,

(x) other items as specified.

5.2 In determining the extent of ice protection coverage and the degree of protection (eg heat per unit area) required, the designer shall consider droplet trajectories and hence rate of water droplet impingement under all critical combinations of airspeed, altitude, ambient temperature, liquid water (or ice crystal) content, and water droplet size. For any but the simplest shapes mathematical modelling and/or rig testing may be required to provide quantitative results.

Note: For most shapes and ambient conditions there is a worst or least favourable speed which results in the worst accretion rate or greatest heat flux required. This speed is not normally either the lowest or the highest speed of which the rotorcraft is capable.

5.3 The extent of ice protection coverage shall also take account of the predicted effects of the maximum likely ice accretion on the unprotected areas.

5.4 The design of the rotorcraft and its systems, inclusive of protective systems such as debris guards, shall be such that the probability of damage from any ice or slush shed from unprotected or de-iced areas of the rotorcraft is minimised. In addition, such shed ice or slush shall not hazard the rotorcraft occupants, nor - so far as is within the control of the system designer - cause hazard to ground personnel.

6 ROTORCRAFT ICE PROTECTION

6.1 GENERAL

6.1.1 The design features of the rotorcraft shall be such as to minimise the hazards of flight in ice forming conditions. Surface discontinuities and excrescences shall be avoided if possible.

6.2 ROTORS

6.2.1 Excessive ice accretion on rotors shall be prevented, unless it can be shown by rotor icing tests that, in the meteorological conditions of para 2.1:

(i) the rotors and their associated control linkages operate satisfactorily,

(ii) loss of lift or directional control, resulting from ice accretion, is within acceptable limits,

(iii) vibration levels and structural stresses resulting from accretion or shedding of ice are within acceptable limits, and

(iv) there is no hazard to the rotorcraft or its systems, its occupants, or - so far as can reasonably be prevented - to ground personnel due to ice shed from the rotors.

6.2.2 If ice accretion is to be prevented, then protection may need to be provided for the rotor hubs, associated controls, and blade limiting stops, as well as for the blades.

6.2.3 Rotor ice protection may be continuous, cyclic, or a combination of both in order to meet the requirements of the Rotorcraft Specification. Operation of the ice protection system shall be accomplished either automatically or manually (as specified) unless a continuous system is provided. Irrespective of the type of system installed, continuous operation of the ice control system in flight shall not damage or affect the life of the rotor or the system unless the design of the system is such that it cannot operate manually in conditions where damage would result (see para. 7.1 (iv)). Indication of the operation of the ice control system, and of any failure to operate, shall be provided in accordance with the Rotorcraft Specification.

6.2.4 The airframe, and the rotors themselves, shall be protected against unacceptable damage by ice thrown from the rotors.

6.3 AEROFOIL SURFACES

6.3.1 Aerofoil surfaces (such as vertical and horizontal stabilizers, stub wings, faired sponsons etc.) may require some degree of ice protection in order to meet the requirements of para. 2. The Design Authority shall give due attention to the effects that such protection may have on the operational performance of the rotorcraft.

6.4 ENGINE AND AUXILIARY AIR INTAKES

6.4.1 Engine air intakes, and any components such as sensor probes which may be located within or in close proximity to the induced airflow, shall be protected against the accumulation of ice in such quantity as to interfere with the performance or safe running of the engine, or which, upon separation, could cause damage to the engine or the airframe.

Note: The engine air intake is assumed to begin at the upstream end of the duct traversed by the air which eventually passes into the engine, and to include the whole of such duct.

6.4.2 Air intakes shall be protected against the ingestion of ice, compacted snow and slush shed from the airframe or the rotors, or picked up from the ground, of a quantity sufficient to cause engine damage or malfunction. They shall also be protected against clear air icing when flying in clouds containing ice crystals. (See Leaflet 711/2, Conditions III and IV.)

6.4.3 The adhesion of ice to any part of the interior of the engine intake shall be completely prevented under the conditions specified in para 2.1 unless tests have shown that the engine can safely accept such ice as might form in the air intake and subsequently be ingested, and there is no significant deterioration in engine performance.

6.4.4 The engine intake anti-icing system shall be designed, as far as is reasonably possible, to "fail-safe"; that is to say, to remain in or revert to the anti-icing mode in the event of a partial system failure, with the system selected on. Continuous operation of the anti-icing system throughout the operational envelope of the rotorcraft shall not damage or affect the life of the engine or the system.

6.4.5 Auxiliary air intakes and their associated components shall be protected against ice accretion (and any subsequent shedding) which could cause damage to, or unacceptable loss of efficiency of, any rotorcraft system.

6.4.6 Debris guards and grilles or screens fitted to engine or auxiliary intakes shall be designed or protected against blockage by ice, snow or slush which could result in abnormal engine or system operation, or an unacceptable engine performance degradation. They shall also be designed to minimise the hazard to engines, airframe or rotorcraft systems due to ice shed from them. Appropriate advice and procedures shall be included in the rotorcraft operating and maintenance instructions to avoid damage so caused.

6.5 AIR DATA SENSORS

6.5.1 Pitot, Static, and other similar air data sensors shall be protected against ice accretion unless it can be shown that such accretion will not cause a hazard, or result in an unacceptable loss of performance of the sensor, its associated systems, or any other part of the rotorcraft.

6.5.2 Where protection is required, it should be automatically selected, unless otherwise stated in the Rotorcraft Specification. The pilot shall be given a cautionary warning (see Chapter 107 para 14.3) of the failure of any air data sensor heating system.

6.6 RADOMES

6.6.1 Even though radomes are manufactured from low thermal conductivity materials, such structures, depending on shape and position, readily accrete ice and slush. Account shall be taken of the possible effects of ice and slush build-up which may attenuate the radar signal, and may also degrade rotorcraft performance and handling.

6.6.2 When siting radomes, account shall be taken of the possibility of damage being caused by ice shed from the rotors and from other parts of the airframe on the radome, and from the radome itself on other parts of the rotorcraft.

6.7 WEAPONS AND WEAPON CARRIERS

6.7.1 Weapons (system, sights, stores, pods, guns etc.) whilst being carried by rotorcraft in icing or snow, must not be adversely affected by accretions during use, deployment or jettison, unless the weapon or rotorcraft specification permits otherwise. Furthermore, ice accretion on such systems must not hazard the rotorcraft during use of the weapon system.

6.8 TRANSPARENCIES

6.8.1 Adequate areas of the transparencies used by the crew for the safe and effective operation of the rotorcraft in its designated role shall be maintained free of ice, snow and slush. (See also Chapter 104, para. 5.1)

6.8.2 The ice protection system shall be capable of correct operation during engine warm-up, ground manoeuvring, hover, take-off and landing, as well as throughout the normal flight envelope of the rotorcraft.

6.8.3 Redundant systems shall be provided if the failure of a single system would result in an inability to comply with para 6.8.1.

6.8.4 The ice protection system shall be tested to demonstrate compliance with para. 6.8.1 without causing overheating or other detrimental effects on the transparency. (See Chapter 715.)

6.9 VENTS AND DRAINS

6.9.1 Airframe and systems vents and drains shall be designed or protected against blockage by ice, snow and slush.

6.9.2 Water systems (where installed) shall be so designed that the waste water is either collected and retained in the rotorcraft, or is discharged in such a way as to prevent it eventually falling from the rotorcraft in the form of lumps of ice. (See also Leaflet 711/1)

7 ICE PROTECTION SYSTEM CONTROLS AND INDICATORS

7.1 All ice protection systems should, unless otherwise stipulated in the Rotorcraft Specification:

(i) operate automatically when ice begins to form (see para. 7.3(v))

(ii) automatically cease to operate as soon as it is no longer required,

(iii) be capable of being switched on-off manually to override the automatic control (see Chapter 107, Table 3, Item 2, for location of manual controls), and

(iv) be capable of operation throughout the flight envelope of the rotorcraft without causing damage, or, alternatively, be so designed that it cannot operate automatically in conditions when damage would result. (In such cases, the pilot shall be given warning that, if the system is operated manually, damage will result.), and,

(v) incorporate back-up systems where flight safety may be impaired by the failure of primary systems.

7.2 The following indications shall be provided to the pilot:

(i) accurate indication of the onset of icing (see para 7.3 (v)), unless otherwise stated in the Rotorcraft Specification,

(ii) confirmation that the protection systems are operating,

(iii) system failure indication,

(iv) indication of overheating if the construction of the component is such, (e.g. by bonding process), that overheating would be a serious hazard,

(v) indication of the severity of the icing conditions, unless this is shown to be unnecessary or impractical.

7.3 Ice detectors shall:

(i) provide an accurate indication of ice build-up or impending ice formation (depending on whether they are of the accretion or inferential type) over the full range of temperatures and associated conditions required by the Rotorcraft Specification unless otherwise specified therein,

(ii) be located so as to give an accurate indication in all phases of flight and associated rotorcraft configurations,

(iii) if they project into the airstream, present the least obstruction to airflow and be constructed to withstand damage from the impact of ice or slush shed from the rotors or other parts of the airframe,

(iv) in the case of accretion type detector probes, be positioned with due regard to the probable trajectories of ice shed from them in order to obviate damage to engines, or other systems or components, and

(v) have a response time such that the safety of the Rotorcraft, and the correct functioning of its engines and systems, are not jeopardised due to ice accretion.

7.4 Ice detectors, and temperature or thermostat probes if used, shall be designed for minimum thermal lag.

8 DESIGN AND CONSTRUCTION

8.1 The ice protection system shall conform to the applicable requirements for equipment for use in aircraft. See DEF STAN 00-35 "Environmental Handbook for Defence Materiel".

8.2 The protection system and its installation in the airframe shall meet the relevant static strength and fatigue damage tolerance requirements of Chapters 200 and 201, in addition to any life requirements contained in the Rotorcraft Specification.

8.3 All electrical installations shall be designed in accordance with Chapter 706.

8.4 Materials used must not be adversely affected by the de-icing or anti-icing medium.

8.5 Temperature limiting devices, insulation, or other methods shall be used, as necessary, to prevent any system or component of the Rotorcraft from exceeding any of the following:

(i) a surface temperature of 70°C, or that which would be hazardous to the occupants,

(ii) its design temperature limitations,

(iii) the self-ignition temperature of any flammable material (solid or liquid) to which it may be exposed either normally or accidentally, including during ground handling and maintenance.

8.6 Any insulation which is not naturally impervious to combustible fluids and which is used in areas where combustible fluids are present shall be covered and sealed with abrasion and fluid resistant covering.

8.7 The system shall be designed with due regard to:

(i) reliability,

(ii) availability,

(iii) maintainability (including inspection and servicing, see Chapter 800), and

(iv) testability,

and shall be shown to meet the requirements of the Rotorcraft Specification in these respects.

8.8 If ground checking requires the use of external supplies or test equipment, then suitable test connection facilities shall be provided within the system.

8.9 The internal diameter of any filling orifice shall not be less than 44mm[1].

9 TESTING

9.1 The Design Authority shall show by ground and flight tests that the requirements of this chapter are complied with, and in particular that:

(i) sufficient ice protection is provided for the icing conditions and durations required by the specification,

(ii) the ice detector will detect, and respond rapidly to, icing conditions,

(iii) the automatic controls function satisfactorily when flying into and out of icing conditions,

(iv) automatic system operation does not occur in conditions where this might hazard the Rotorcraft or system, and

(v) significant failures of the protection system are indicated to the pilot and any back-up systems operate satisfactorily at such times.

9.2 Testing shall be performed in accordance with the provisions of Chapter 1006.

9.3 Where flight testing of untried systems or applications may involve some risk, consideration shall be given to first conducting ground tests on representative test specimens using rigs, icing tunnels or other suitable facilities.

9.4 The ice protection systems under compliance test shall be representative of the standard to be used on production Rotorcraft.

9.5 With the ice protection system installed and working, the handling and performance requirements of Part 6 must be complied with.

9.6 The test schedule shall be so drafted, and suitable instrumentation provided, as to ensure (by calculation on the basis of the tests that are performed) that the Rotorcraft and its ice protection systems will also meet the requirements of the Rotorcraft Specification under those conditions which may not be available during the designated test period.

10 CROSS REFERENCES TO OTHER CHAPTERS

10.1 A number of requirements directly applicable to ice protection appear elsewhere in this publication and the most important of these are listed below. (See also the Alphabetical Index.)

Chapter	Subject
100	General Requirements.
101	Operation in various Climatic Conditions.
104	View and Clear Vision.
105	Crew Stations - General Requirements.
107	Pilot's Cockpit - Controls and Instruments.
200	Static Strength and Deformation.
201	Fatigue.
400	General Design Data.
407	Precautions against Corrosion and Deterioration.
700	Propulsion System Installations.
706	Electrical Installations.
716	Static and Pitot Pressure Systems.
724	Instrument/Display Installations.
800	General Maintenance Requirements.
1006	Ice Protection Systems.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	25/11	3212

LEAFLET 711/0

ICE PROTECTION

REFERENCE PAGE

MOD Specifications

DTD/RDI 3961	Windscreen de-icing pumps.

RAE Reports

SME 3380	Protection of aircraft against ice.
Mech Eng 2	Kinetic temperature of propeller blades in conditions of icing.
Mech Eng	Design of heat exchangers.
TR 77090	Calculation of surface temperatures and ice accretion rate in a mixed water droplet/ice crystal cloud.
Memo MAT/ST 1004	An investigation into the anti-icing of a heated cylinder in mixed conditions.
TR 82128	Ice accretion on aerofoils in 2-dimensional compressible flow - a mathematical model.
TR 84060	Calculation of water droplet trajectories about an aerofoil in steady 2-dimensional compressible flow.
TR 87013	Measurement of drag increase due to ice accretion on aerofoils of NACA 0012 and RAE 9645 section.
TR 88052	HOVACC - An aerofoil ice accretion predicition program for steady, two-dimensional, compressible flow conditions.
TR 90054	TRAJICE2 - A combined droplet trajectory and ice accretion prediction program for aerofoils.

RAE Technical Notes

Eng 124	Ice guards on engine air induction systems.
Mech Eng 12	Icing tests on engine air induction systems.
Mech Eng 19	Provision of heat on aircraft for protection against ice and for cabin heating.
Mech Eng 57	The maintenance of clear vision through aircraft transparencies.
Mech Eng 58	Tests of water spraying for simulating icing conditions ahead of a turbine engine in flight.

Mech Eng 62	The problem of icing as it affects modern military aircraft.
Mech Eng 203	Impingement of water droplets on aerofoils.
Mech Eng 283	The analysis of measurements of free ice and ice-water concentrations in the atmosphere in the equatorial zone (Tables 1 - 6).

<u>NGTE Memorandum</u>

M 106	The anti-icing of compressor blades by surface heating - Part 1 - Stator blades.

<u>ARC Report</u>

R & M 2805	Evaporation of drops of liquid (formerly RAE Report Mech Eng 1).

<u>RAeS Journal</u>

August, 1959	Water and ice in the atmosphere.

<u>AGARD</u>

AGARD 16	Icing problems and recommended solutions.
AGARD AR127	Aircraft icing.
AGARD AR223	Rotorcraft icing - progress and potential
AGARD CPP236	Icing tests for aircraft engines.

<u>Defence Standards</u>

00-35	Environmental handbook for defence materiel
00-971	General Specification for aircraft gas turbine engines.
01-5	Fuels, lubricants, and associated products.

Note: The following US reports contain useful bibliographies on the subject of aircraft icing and ice protection:

(i) Advisory Circular AC 20-73, Aircraft Ice Protection, (Appendices 1 and 2). FAA, 1971.

(ii) NASA TM 81651, Selected Bibliography of NACA - NASA aircraft icing publications.

(iii) Society of Automotive Engineer (SAE) Aerospace Information Report (AIR) 4015, Icing technology bibliography.

LEAFLET 711/1

ICE PROTECTION

PRECAUTIONS TO PREVENT WASTE WATER LEAVING ROTORCRAFT AS ICE

1 INTRODUCTION

1.1 Investigation into dangerous falls of ice from certain aircraft in flight has led to the conclusion that, in many instances, the ice was formed by the discharge of waste water on to the cold airframe structure; the ice then became detached in lumps when the aircraft entered warmer air.

1.2 It is, therefore, required for Rotorcraft which have waste water systems installed that waste water shall not be so discharged as to leave the Rotorcraft in the form of lumps of ice of such a size as to cause a hazard. This Leaflet gives recommendations for compliance with this requirement.

2 COLLECTION OF WASTE WATER

2.1 The surest way of meeting the requirements is to collect all waste water (by use of a soil tank, if necessary) and retain it in the Rotorcraft.

3 DISCHARGE OF WASTE WATER IN FLIGHT

3.1 Where the method of para 2.1 is not adopted, the waste water should be prevented from freezing in the outlet pipe and should be discharged clear of the Rotorcraft in such a manner that impingement of the waste water on any part of the Rotorcraft downstream of the outlet cannot occur.

3.2 The chances of ice forming on the airframe structure will be further reduced if the waste water is discharged in large amounts infrequently rather than in small amounts frequently. Hence it is desirable to store, at a suitable temperature, as much waste water as feasible and then to discharge it quickly.

4 RECOMMENDATIONS FOR DESIGN OF WASTE WATER OUTLETS

4.1 The water drain mast should either be heated to provide de-icing, or manufactured of low thermal conductivity materials if not de-iced.

4.2 The outlet pipes should cause the least possible interference with the air flow and should be long enough to clear the boundary layer (see also para 3.1).

5 FLIGHT TESTS

5.1 Where it is doubtful whether waste water discharged in flight will be carried clear of the Rotorcraft, flight tests should be made with whitewash on the Rotorcraft surfaces near to, and downstream of, the outlet and colour dye in the water.

LEAFLET 711/2

ICE PROTECTION

ICING CONDITIONS

1 GENERAL

1.1 The design atmospheric icing conditions are defined in this leaflet.

1.2 These conditions represent standards which a Rotorcraft and its equipment may be required to meet in order to ensure the ability to fulfil the Service operational requirements stated in the Rotorcraft Specification in inclement weather.

1.3 The extent to which these conditions are applicable to any particular Rotorcraft or operational role fit will be stipulated in the Rotorcraft Specification.

1.4 The design atmospheric icing conditions in Table 1 extend up to 3000m altitude to cover the normal operational altitude range of Rotorcraft. Where Rotorcraft are required to operate at higher altitudes, the associated design atmospheric icing conditions shall be stipulated in the Rotorcraft Specification.

1.5 These design conditions have been based on statistical analyses of thousands of observations obtained over several decades in many geographic regions.

1.6 In natural icing the conditions experienced are unlikely to correspond precisely to any one of these design conditions; indeed, natural icing conditions may well be mixed, and may change rapidly in a short distance (or time).

2 METEOROLOGICAL FACTORS INFLUENCING ICING

2.1 Atmospheric icing is a complex phenomenon influenced by many interdependent factors. The more significant of these are as follows.

2.2 Ambient Temperature: Icing can occur at ambient temperatures between -40°C (-80 in ice crystal cloud) and +5°C or above. At positive ambient temperatures icing can occur in engine air intakes, carburettor venturis, and other places where the air experiences adiabatic cooling due to expansion of the airflow.

2.3 Liquid Water Content: The design icing conditions specify liquid water content (LWC) from 0.2 to 1.5 g/m^3. The higher concentrations are associated with the higher temperatures within the icing range. The liquid water content in layer (stratiform) cloud seldom exceeds 1 g/m^3, whereas much higher concentrations can occur in convective (cumuliform) cloud. However, convective cloud is normally much more limited in horizontal extent than layer cloud.

2.4 Ice Crystal Content: Clouds containing ice crystals can occur at temperatures from 0°C down to -80°C, and at altitudes up to 18000 m. At temperatures down to, and possibly below, -20°C (but no lower than -40°C) the ice crystals usually occur in combination with supercooled water. At lower ambient temperatures ice crystals and snow will not normally adhere to a cold surface. However, in ice crystal cloud, accumulation of slush can occur even on heated surfaces such as turbine engine air intakes, and on pitot heads and other sensor probes. Furthermore, the use of anti-icing fluids in ice crystal cloud and snow may cause unwanted accretions.

2.5 Droplet Size: The median volume diameter droplet size is a function of the icing condition. Reference to Table 1 shows a range of 10 microns (in freezing fog) to 1500 microns (in freezing rain). Within each condition the droplets are assumed to be distributed in size about the median diameter. The distribution is defined in Table 2. Water droplet and ice crystal size has a marked influence on both the extent and the severity of ice accretion on a body (see para 3.5). In calculating accretion characteristics it is recommended that, in addition to using the distribution defined in Table 2, the analysis should also be performed assuming a constant droplet diameter (equal to the median volume diameter appropriate to the Condition).

2.6 Pressure Altitude: The altitude ranges in which icing can occur depend considerably on the condition. For example, freezing fog often extends no higher than 15m (50 ft) above ground level, and seldom above 100m. However the severe icing associated with the Periodic Maximum Condition (normally associated with cumuliform cloud) is most likely to be experienced above 1200m.

3 THE INFLUENCE OF ICING CONDITIONS ON ICE ACCRETION CHARACTERISTICS

3.1 The distribution, and the type, of ice accretion is strongly dependent on air temperature, but is also influenced by the following factors:

(i) the shape, size, and attitude relative to the airflow of the accreting body,

(ii) the surface temperature of the body and its thermal conductivity if in contact with a source of heat or cold,

(iii) the liquid water or ice crystal content,

(iv) the size of the water droplets, and the airspeed,

(v) the atmospheric pressure at the altitude the rotorcraft is flying.

3.2 At the lowest temperatures in the icing range the supercooled water freezes on impact on a cold surface, normally in a narrow band centred on the stagnation point, to form rime ice. This is usually white, opaque and relatively streamlined in profile when small, but can become sharply pointed when large.

3.3 At the highest temperatures in the icing range, close to 0°C, the supercooled water does not freeze immediately on impact but runs back, losing heat by evaporation, conduction, and convective cooling until ice forms. This is glaze ice which may be smooth and fairly transparent. Since the runback of the impinging water occurs on both sides of the stagnation point, the ice formation grows as two horns which may be separated by a relatively ice-free area. Because of the high subsonic Mach number conditions existing on the outboard section of the rotor, it is possible for ice to grow only in the low pressure region close to the leading edge on the upper surface. Such accretion has been termed beak ice, and resembles a slushy ridge of ice which self-sheds. This accretion is thought to be specific to helicopter rotors. While it rarely grows to a large size (usually less than 2% chord) its effect on the rotor lift and drag characteristics cannot be neglected.

3.4 At intermediate temperatures in the icing range the extent and type of ice accretion lie between those described in paras 3.2 and 3.3. The ice may take on an arrowhead or a mushroom shape, and the ice texture may range from rime through cloudy ice to glaze, depending on the temperature.

3.5 Since there is a spanwise velocity gradient along the blade of a rotor, it is possible for rime, glaze and beak ice to exist on the same rotor. Rime ice will tend to form at inboard stations followed by glaze ice, with beak ice forming at outboard stations. The spanwise extent of these accretions will vary with the icing conditions, but particularly with ambient temperature and, in the case of glaze ice, with LWC.

3.6 The combined influences of body shape and size, liquid water content, water droplet diameter, and airspeed determine the rate at which supercooled water impacts with the surface, and the extent of the impact area. The larger the water droplets, the smaller the body, and the higher the airspeed - the greater the 'rate of catch' per unit frontal area, and thus the greater the rate of ice accretion if the air and surface temperatures are sufficiently low.

Note: Great caution should be exercised in extrapolating ice accretion rate data, whether obtained from calculation or by test. The very process of accreting ice modifies the profile of the accreting body. This alters the rate of catch, as well as changing such factors as the rate of convective cooling due to flow acceleration around rapid changes of profile. These factors modify the subsequent rate of ice accretion, sometimes resulting in an accelerating rate of ice accumulation.

4 RELATIONSHIP BETWEEN ICING SEVERITY STANDARDS

4.1 There is no simple or precise relationship between the design icing conditions defined in this leaflet and the Meteorological Office forecasters' terminology, viz. "light", "moderate" and "severe".

4.2 Reference to Table 1 shows that the LWCs appropriate to the Continuous Maximum and Periodic Maximum design icing conditions are functions of OAT and altitude. For the Continuous Maximum condition LWC ranges from 0.9 g/m^3 at +5°C to 0.2 at -30°C, whereas the Periodic Maximum LWCs range from 1.35 at +5°C to 0.3 at -30°C.

4.3 As a rough guide the Meteorological Office forecasters' "light icing" covers LWCs up to approximately 0.5 g/m^3, "moderate icing" covers LWCs from 0.5 to 1.0, and "severe icing" from 1.0 to 4.0 g/m^3.

Note: The US Army definitions are similar, viz.:

TRACE:	0	to 0.05 g/m^3
LIGHT:	0.05	to 0.5 g/m^3
MODERATE:	0.5	to 1.0 g/m^3
HEAVY:	1.0	to 2.0 g/m^3

5 AMPLIFICATION OF DESIGN ATMOSPHERIC CONDITIONS

5.1 The design atmospheric icing conditions are defined in Table 1 of this Leaflet. The amplification which follows is provided to assist the design of ice protection systems.

5.2 SUPERCOOLED WATER DROPLET CONDITIONS I AND II

5.2.1 In establishing the design icing conditions for Rotorcraft in the 0 - 3000m altitude range, it was clear that both the Instantaneous Maximum and the Intermittent Maximum Conditions used for aeroplanes were not relevant as they applied above 3000m. The validity of the Continuous Maximum was also questioned, as it was based on data covering a wider altitude band than was appropriate for Rotorcraft.

5.2.2 Re-examination of the original data (References 1 to 4), excluding those for altitudes above 3000m, and the inclusion of some additional data (see References 5 to 9), confirmed the applicability of the Continuous Maximum Condition to altitudes below 3000m. It was noted that periodic encounters with more severe LWC values had also been recorded. These formed the basis for Condition II - Periodic Maximum Icing where they were set at 1.5 times the LWC for the Continuous Maximum Icing Condition, but with the horizontal extent reduced to 6 km in every 100 km of Condition I.

5.3 MIXED CONDITIONS III AND IV

5.3.1 Icing trials have clearly demonstrated that mixed snow, or ice crystal, and water droplet conditions are encountered in the 0 - 3000m altitude range, and can have hazardous consequences. The frequency of these encounters, their horizontal extents, total water content, solid/liquid ratio and solid particle size have not been fully determined. The values presented in Table 1 for Conditions III and IV are based on reasoned judgement and will be reviewed in the light of future data.

5.3.2 Due to the wide range in shape and size of ice crystals and snowflakes, a unique size cannot be specified. In estimating impingement limits both large and small crystals should be considered. The relationship between mass (m), in milligrams, and maximum linear dimension (d) in millimetres, may be assumed to be $m = 0.010d^2$ for powder snow, and $m = 0.004d^2$ for plane dendrites in the range $1<d<5$ millimetres.

5.4 SNOW CONDITIONS V TO VII

5.4.1 Two snow conditions are defined: falling snow, and recirculating snow generated by hovering flight in ground effect over loose lying snow.

5.4.2 The water content for recirculating snow is based on References 10 and 11 together with other unpublished data, but no data is available on particle size (see para 5.3.2).

5.4.3 The water content for recirculating snow is based mainly on visual assessment and may be an underestimate. However, the recommended value of 1.5 g/m^3 is likely to last only for a minute or less until the loose snow has blown away, unless the Rotorcraft is moving slowly in ground effect over loose snow (which is not a recommended manoeuvre).

5.4.4 In designing ice protection systems it should be noted that snow can form slush on, or downstream of, heated surfaces. The additional heating requirements for anti-icing systems due to the latent heat of the snow should be considered.

5.5 FREEZING FOG CONDITION VIII

5.5.1 The values of 10 and 20 microns quoted for droplet size are for radiation fog and advection fog respectively. Both conditions should be considered in ice protection system design.

5.6 FREEZING RAIN/DRIZZLE CONDITION IX

5.6.1 The condition of freezing rain is based on a rainfall rate of 4 mm/hr and covers the temperature range 0 to -10° with 1.5 mm raindrops. The freezing drizzle condition covers the range +5 to -15°C with LWC decreasing linearly with temperature from 0.3 g/m^3 at +5°C to zero at -15°C. The droplet size is assumed to be 0.2 mm (200 microns).

REFERENCES

Ref	Author	Title etc.
1	A R Jones William Lewis	Recommended values of meteorological factors to be considered in the design of aircraft ice prevention systems. NACA Tech Note 1855, March 1949.
2	P T Hacker R G Dorsch	A summary of meteorological conditions associated with aircraft icing and a proposed method of selecting design criteria for ice protection equipment. NACA Tech Note 2569, 1951.
3	William Lewis N R Bergrum	A probability analysis of the meteorological factors conducive to aircraft icing in the US. NACA Tech Note 2738, 1952.
4	C G Abel	Report of the first year's flying on the development of flight testing techniques for finding and measuring natural icing conditions. ARC Current Paper 221, 1953. Also CP 222 and CP 223 for second and third years respectively.
5	D T Brown et al	Engineering summary of airframe icing technical data. FAA Tech Report ADS-4, March 1964.
6	P J Perkins	Summary of statistical icing cloud data measured over United States, North America, Pacific and Arctic Oceans during routine aircraft operations NACA Memo 1-19-59E, January 1959.
7	F S Atkinson	A report on icing trials with the S61N, January/April 1971. BEAH/ENG/TD/R/113, February 1972.
8	R K Curtis	Wessex Mk 5, XS 481 Icing Trials at Kirina, Sweden during winter 1972/73. AAEE Interim Report, 1973.
9	A Wilson C J Green	Wessex Mk 5, XT 762 and 768. Icing Trials in Canada during winter 1972/73. AAEE Interim Report, 1973.
10	J R Stallabras	Preliminary measurements of Snow Concentration. NRC Report LTR-LT-42, 1972.
11	R N Hardy	Rates of fall of snow, mixed precipitation and freezing rain. Met Office Investigation Division TN9, 1974.

TABLE 1

DEFINITION OF DESIGN ICING CONDITIONS

Condition	Air temp °C	Water content g/m^3	Horizontal extent km	Droplet median volume dia. microns	Altitude Range $m \times 10^3$	Notes
I Continuous Maximum Icing	+5 0 -10 -20 -30	0.90 0.80 0.60 0.30 0.20	Continuous	20	0 - 3	1, 2
II Periodic Maximum Icing	+5 0 -10 -20 -30	1.35 1.20 0.90 0.45 0.30	6 km every 100 km of Condition I	20	0 - 3	1,3,4

continued

TABLE 1 (Cont.)

DEFINITION OF DESIGN ICING CONDITIONS

Condition	Air temp °C	Water content g/m^3	Horizontal extent km	Droplet median volume dia. microns	Altitude Range $mx10^3$	Notes
III Mixed Conditions Continuous	0	(0.20) LWC (0.60) Ice				
	-10	(0.25) LWC (0.45) Ice	Continuous		0 - 3	1,5,6
	-20	(0.10) LWC (0.20) Ice				
	-30	0.20 Ice				
IV Mixed Conditions Periodic	0	(0.30) :WC (0.90) Ice				
	-10	(0.20) LWC (0.70) Ice	6 km every 100 km of Condition III.		0 - 3	1,3,5
	-20	(0.15) LWC (0.30) Ice				
	-30	0.30 Ice				

continued

TABLE 1(Cont.)

DEFINITION OF DESIGN ICING CONDITIONS

Condition	Air temp °C	Water content g/m^3	Horizontal extent km	Droplet median volume dia. microns	Altitude Range $m \times 10^3$	Notes
V Falling Snow Continuous	+3 to -30	0.8	Continuous		0 - 3	7
VI Falling Snow Periodic	+3 to -30	1.5	8 km every 100 km of Condition V		0 - 3	3,7
VII Blowing or Recirculat-ing snow	0 to -30	1.5			Hover, and manoeuvre on or near the ground	3
VIII Freezing fog	0 to -30	0.3		10 to 20	0 - 15m AGL.	3
IX Freezing Rain/ Drizzle	+5 to -15 0 to -10	0.3 at 0°C to 0 at -15°C 0.3	100km	200 1500	0 - 3 0 - 3	3,8

Notes

1. At altitudes below 1200 m (4000 ft), water content is assumed to decrease linearly with decreasing altitude to zero at sea level, except that below 300m (1000 ft) the content for 300 m (1000 ft) applies.

2. In determining the limits of the impingement area, droplet sizes up to 30 microns shall be considered.

3. The Rotorcraft Specification will define the durations for which protection is required against Conditions II, IV, and VI to IX inclusive.

4. In determining the limits of the impingement area, droplet sizes up to 40 microns shall be considered.

5. In the temperature range 0 to -20°C the ice crystals are likely to be mixed with water droplets (with a maximum diameter of 2 mm) up to a content of 1 g/m^3 or half the total content, whichever is the lesser, the total content remaining numerically the same. Below -20°C all the water present may be assumed to be in the form of ice crystals.

6. When the horizontal extent is shown as 'continuous' it is acceptable to show that the rotorcraft functions satisfactorily for the normal endurance of the rotorcraft, or until an equilibrium state has been achieved, or as otherwise stipulated in the Rotorcraft Specification.

7. See Figure 1 - Empirical Relationship between Snow Concentration and Observed Visibility - for a guide to the severity of snow conditions.

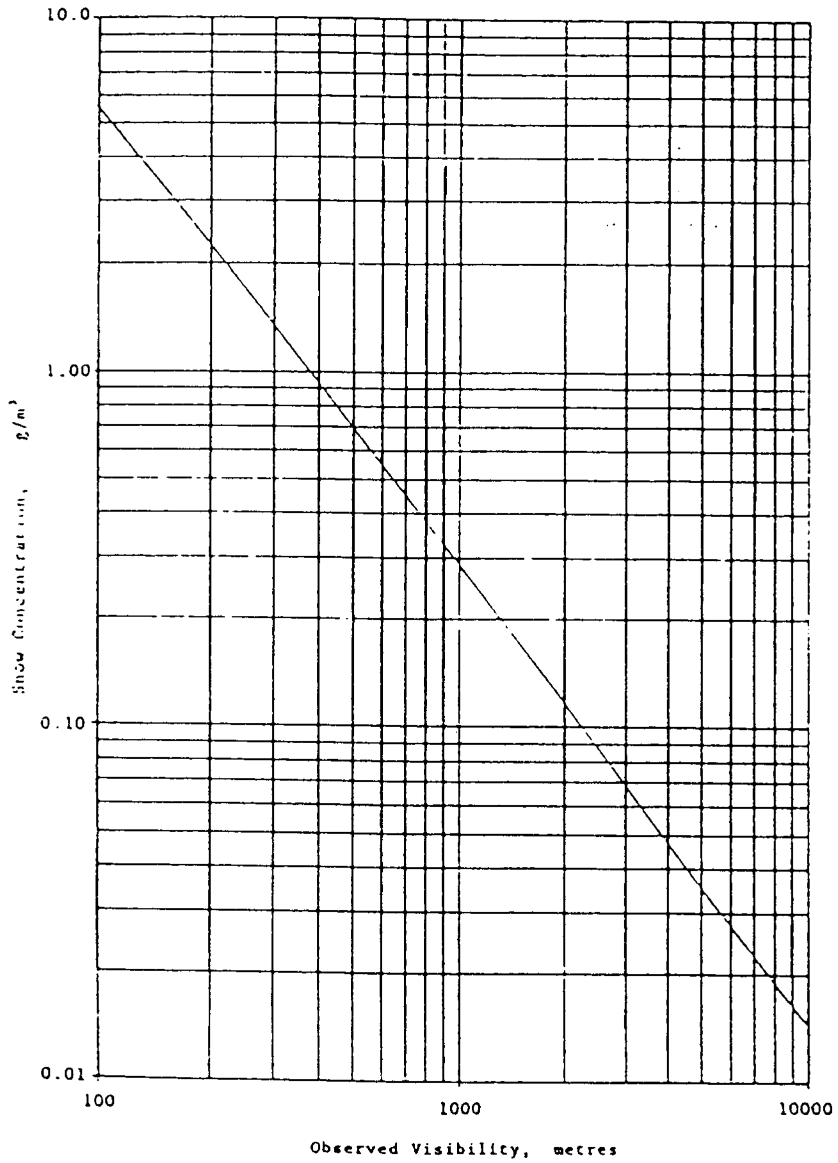
8. At air temperatures above 0°C the water content at 0°C shall be assumed to apply.

TABLE 2

RANGE OF DROPLET SIZES

% by weight of total water content contained in droplets of diameter d_F	3	8	20	30	20	10	5	4
Droplet diameter ratio, - d_F/d_V	0.27	0.55	0.83	1.10	1.39	1.67	1.95	2.22

Note: The droplet sizes quoted in Table 1 are the volume median diameters (d_V) for the distribution shown in Table 2; d_F is the particular droplet diameter under consideration.



NOTES:

1. Falling Snow

The snow severity definitions used within CA Release Limitations are as follows:

Severity .. Visibility

TRACE SNOW > 2000m
LIGHT 1200 to 2000m
MODERATE 400 to 1200m
HEAVY less than 400m

2. Recirculating Snow

HEAVY RECIRCULATION - reduction in visibility such that objects outside the recirculating snow cloud produced by the rotorcraft are obscured or can not be recognised by the pilot.

LIGHT RECIRCULATION - visibility may be impaired but objects outside the recirculating snow cloud can be recognised by the pilot.

Fig 1 Empirical Relationship between Snow Concentration and Observed Visibility

LEAFLET 711/3

ICE PROTECTION

ICE PROTECTION SYSTEMS

1 GENERAL

1.1 The design or selection of ice protection systems requires consideration of the following:

(i) The Design Icing Conditions in which the Rotorcraft is required to be capable of meeting its Service operational requirements stated in the Rotorcraft Specification,

(ii) The areas over which ice accretion can occur, and the effects of ice accretion on Rotorcraft or systems performance,

(iii) The rate at which ice accretion can occur, and the possible damage or hazards from shed ice.

1.2 The Design Icing Conditions are defined in Leaflet 711/2. The Rotorcraft Specification will stipulate which of the conditions shall be met, and for what durations.

2 AREA OF ICE ACCRETION

2.1 The area over which ice can accrete on an object is determined by:

(i) its shape, size and disposition relative to the airflow,

(ii) its surface temperature when exposed to the icing condition, and

(iii) the water droplet or ice crystal size and air velocity.

2.2 The shape of the ice accretion is largely a function of icing surface temperature and freezing fraction. These are complex functions of the parameters (i) and (iii) above, and of liquid/solid water concentration, ambient pressure, temperature and relative humidity. A concise relationship between accretion shape and surface temperature and freezing fraction has not been established. In general a freezing fraction of unity and low icing surface temperature gives a sharp pointed rime ice growth. As temperature rises to zero the ice type changes to glaze ice and the accretion shape changes with decrease in freezing fraction through arrowhead, blunt arrowhead and mushroom to double or single horned at low freezing fraction.

2.3 For fixed aerofoil surfaces the area and chordwise limits can be calculated using suitable computer programs, (see for example Reference 11). Further relevant information is contained in References 1 to 10 inclusive.

2.4 While a comprehensive theoretical treatment of main rotor ice accretion is still not possible, research to date suggests that computer programs developed for fixed-wing applications (e.g. References 10 or 11) may be used with reasonable confidence to assess the likely rate, shape, and both spanwise and chordwise extents of ice accretion. Photographic evidence gathered during flight tests in natural icing conditions has confirmed a variation in ice type and profile along the rotor due to the spanwise velocity gradient. Results from tests on aerofoils in icing wind tunnels confirm that cyclic variations in pitch increase the limits of droplet impact but do not significantly effect the size or the profile of the ice, when compared to an equivalent accretion on a static aerofoil set at the mean incidence. Cyclic variations in airspeed are predicted to have a small effect on the threshold for rime, glaze, and beak type ice accretion, but this remains to be confirmed by flight tests. On the knowledge available to date it is therefore recommended that evaluation of ice accretion limits at a given spanwise station be made for both the advancing and the retreating blade in forward flight and for the hover condition. For the evaluation of the ice accretion profile and size the use of the local mean blade incidence and airspeed is recommended.

3 RATE OF ICE ACCRETION

3.1 The rate at which ice may form on a surface is determined by:

(i) the shape, size and disposition of the surface relative to the airflow,

(ii) the water droplet or ice crystal size,

(iii) the airspeed, ambient air temperature and pressure,

(iv) the amount of water blown off,

(v) the liquid water or ice crystal content,

(vi) the surface freezing fraction (which depends on the balance of the heat transfer from the surface).

Items (i) to (iv) control the surface water catch efficiency, which with item (v) determines the total rate of catch of water/ice. Item (vi) determines how much of the collected water/ice freezes.

3.2 The rate of catch of water drops on aerofoil sections can be calculated using computer programs. In the absence of a suitable program the methods of Reference 1 may be used. An allowance should be made for the water which blows off and therefore does not require evaporation, and for kinetic heating. The kinetic temperature rise should be taken as half the value assumed for dry air conditions. The rate of catch of droplets on a radome may be assumed to be equal to that of a sphere with the same frontal area.

4 NOTES OF ICE PROTECTION SYSTEMS

4.1 GENERAL

4.1.1 Ice protection systems can be either of two types:

(i) Anti-icing Systems, where the surface is maintained free from ice accretion at all times, or

(ii) De-icing Systems, where accretion is allowed to occur and is periodically removed before its effects, and those of the shed ice, are hazardous.

4.1.2 Since there may be operational penalties associated with the provision of ice protection systems, and also with flight in icing conditions even when protection systems are fitted, it is essential that the requirements of the Rotorcraft Specification are carefully examined at an early stage in the design.

4.2 ANTI-ICING SYSTEMS

4.2.1 Anti-icing can be achieved by continuous heating, employing either electrical or hot air systems, or by the use of freezing point depressant fluids.

4.2.2 If a continuous heating method is employed, either an adequate surface should be maintained above 0°C, so that freezing of any run-back water which might cause a hazard is prevented or sufficient heat should be applied to the wetted area to evaporate all impinging water not blown off. Computer programs have been developed to perform these calculations (References 12 and 13). In the absence of a suitable program the heat required may be calculated by the methods given in References 2 and 6. The amount of heat should be evaluated for the droplet sizes given in Table 2 of Leaflet 711/2 with due allowance for the latent heat of fusion of ice for the mixed and snow conditions.

4.2.3 When methods employing freezing point depressants only are used as anti-icing systems, the quantity of fluid required to depress the freezing point below the local temperature of the surface should be calculated. The amount of fluid applied should be 1.25 times the calculated quantity for local variations both in the mixing process and in distribution (see Reference 7). The pumps should be duplicated, each capable of providing the full requirements. Allowance should also be made for the mixed conditions of Table 1 of Leaflet 711/2, and for the effects of the fluid when flying in snow.

Note: The use of fluid anti-icing systems may be ineffective in certain snow conditions; their use can aggravate and increase unwanted slush and ice accretions.

4.3 DE-ICING SYSTEMS

4.3.1 De-icing can be achieved by the application of heat through electrical or hot air systems to weaken the bond between the ice and the surface, by the use of freezing point depressant fluids, by mechanical means or by the use of low adhesion (ice-phobic) coatings or pastes.

4.3.2 If a heating method is employed, care should be exercised in the selection of heating power and duration to ensure that free water on the surface, after shedding of the ice, does not result in an unacceptable amount of run-back icing on unprotected parts of the surface. Methods of calculating heating requirements are given in Reference 7.

4.3.3 If eletro-impuse methods are used to provide de-ice protection, the airworthiness of the resulting structure must be fully substantiated.

REFERENCES

To assist the design of ice protection systems, the following references may be consulted:

Ref No	Author	Title
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2	Hardy J K	Protection of aircraft against ice. RAE Report SME 3380, 1946.
3	Brun A E	Icing problems and recommended solutions. Agardograph 16, November 1957.
4	Dorsch R G Brun R L Gregg J L	Impingement of water droplets on an ellipsoid with fineness ratio 5. NACA TN 3099, 1954.
5	Coles W C	Icing limit and wet-surface temperature variation for two airfoil shapes under simulated high speed flight conditions. NACA TN 3396, 1955.
6	Bowden D T et al	Engineering summary of airframe icing technical data. FAA Tech Report ADS-4 1964.
7	Messinger B L	Equilibrium temperature of an unheated surface as a function of airspeed. J Ae Sc January 1953 Vol 20 No 1.
8	Cansdale J T	Calculation of surface temperature and ice accretion rate in a mixed water droplet - ice crystal cloud. RAE Technical Report TR 77090, 1977.
9	Cansdale J T Gent R W	Ice accretion on aerofoils in 2-dimensional compressible flow - a mathematical model. RAE Technical report TR 82128, 1982.
10	Gent R W	Calculation of water droplet trajectories about an aerofoil in steady, 2-dimensional, compressible flow. RAE Technical Report TR 84060, 1984.
11	Gent R W	TRAJICE2 - A combined droplet trajectory and ice accretion prediction program for aerofoils. RAE Technical Report, 1990.
12	Gent R W	HRB1D - A computer program for the design and assessment of electrothermal rotor de-icing systems. RAE Technical Report TR 87047, 1987.
13	Gent R W	HRB2D - A computer program for the detailed assessment of electrothermal rotor protection systems. RAE Technical Report TR.

CHAPTER 712

FIRE PRECAUTIONS

1 INTRODUCTION

1.1 The requirements of this chapter define a standard of fire precautions which shall be met on all rotorcraft conditional on their operational role and associated combat environment.

1.2 Definitions essential to the accurate interpretation of the requirements of this chapter are given in Leaflet 712/1.

2 GENERAL REQUIREMENTS

2.1 The Rotorcraft Designer shall designate the fire zones and define the fire and explosion vulnerability of the rotorcraft with respect to all fire hazards.

2.2 A survey shall be conducted to determine the temperatures occurring in all relevant areas with reference to the risk of spontaneous ignition of flammable fluids and the provision of fire and overheat warning systems.

2.3 Non compliance with the fire precaution requirements appropriate to rotorcraft when operating in combat environments shall be agreed with the Rotorcraft Project Director.

2.4 In each area where flammable fluids or vapours might be present due to leakage of a fluid system, there shall be means to minimise the probability of ignition of the fluids and vapours and the resultant hazards if ignition does occur.

2.5 Those controls and services located in a designated fire zone which are necessary for the detection and suppression of fires shall be fireproof (for electrical cables etc., see paras 2.10 and 3.11.6).

2.6 Essential flight controls and services and any pipes carrying flammable fluids shall not pass through or be located in a designated fire zone or in adjacent areas which would be subjected to the effects of fire in a designated fire zone, unless they are fireproof or shielded. It shall not be difficult to operate such controls during or after the fire.

2.7 Filling points for flammable fluids shall be designed to prevent the entry of fluids into any portions of the rotorcraft other than tanks.

2.8 The cooling air supply for any electrical or electronic equipment shall be conveyed and discharged so that it does not create a hazard following failure of the equipment. Where necessary the cooling duct shall be fireproof.

2.9 The fire extinguisher system, the quantity of the extinguishing agent, the rate of discharge and the discharge distribution shall be adequate to extinguish fires and minimise the probability of re-ignition.

2.10 Electrical cables and terminals, in designated fire zones, that are used during an emergency shall be at least fire resistant.

3 PRECAUTIONS IN DESIGNATED FIRE ZONES

3.1 The requirements of this para are applicable to, the installation of conventional piston or turbine-engined power units, auxiliary power units (APU's) and any fuel-burning heater or other combustion equipment installation.

3.2 Each designated fire zone shall meet the requirements of paras 3.8 to 3.12 below inclusive.

3.3 Designated fire zones are:

3.3.1 Piston Engine Installations.

(i) Engine Power Section.

(ii) Engine Accessory Section.

3.3.2 Turbine Engine Installations

(i) Engine Compressor and Accessory Sections.

(ii) Engine Combustion, Turbine and Exhaust System Sections.

3.3.3 Auxiliary Power Unit Installations (also para 3.12.4).

(i) Auxiliary Power Unit Compartment and Exhaust System Section.

3.3.4 Combustion Heater Installations.

(i) The region surrounding the heater.

(ii) The ventilating air passages surrounding the combustion chamber.

3.3.5 Any other region which may be specified by the Rotorcraft Project Director.

3.4 In addition to compliance with the detailed requirements of precautions against fire, all reasonably practical measures shall be taken to minimise the probability of fire both in flight and on the ground with due regard to the rotorcraft's operational role.

3.5 No fire in any one engine of a multi-engined rotorcraft shall result in stoppage or critical power loss of the other engine(s)

3.6 TORCHING FLAMES

3.6.1 Precautions shall be taken where necessary to protect the rotorcraft from the hazardous consequences of a torching flame burning through the engine.

3.6.2 The resistance of components and structure to the effects of torching flames associated with a combustion chamber or exhaust system burn-through shall be in compliance with para 3.13. Where torching flame testing is relevant the torching flame shall be produced by a burner possessing characteristics consistent with a combustion chamber/exhaust system burn-through in an engine of the type under consideration under its most critical operating conditions. Details of the torching flame characteristics to be simulated in these tests shall be obtained from the engine manufacturer, and such data shall be acceptable to the approving authority. Tests to demonstrate compliance may not be necessary if similarity can be shown with other components which have been tested in accordance with para 3.6.3.

3.6.3 BS 3G100 Part 2, Section 3, Sub-section 3.13 paras 4, 5 and Appendix D give data applicable to demonstrating compliance with torching flame requirements. However, due to variability in specific engine operation conditions etc., the burner characteristics in para 4 Table 1 of the BS shall be amended as necessary to reflect the characteristics of the engine type being considered.

3.7 FLAMMABLE FLUID SYSTEMS

3.7.1 No parts of exhaust systems, jet pipes or other potential sources of ignition, including electrical equipment, shall be located in hazardous proximity to any systems carrying flammable fluids which may be subject to leakage, unless suitable design precautions are taken.

3.7.2 Each pipe, fitting, and other components carrying flammable fluids in any area subject to engine fire conditions and each component which conveys or contains flammable fluid in a designated fire zone shall be at least fire resistant, except that flammable fluid tanks and supports in a designated fire zone shall be fireproof or be enclosed by a fireproof shield unless damage by fire to any non-fireproof part will not cause leakage, spillage or risk of excessive temperature/pressure build up of flammable fluid. Components shall be shielded or located to safeguard against the ignition of leaking flammable fluid.

3.7.3 Vents and drain pipes and their fittings, whose failure will not result in, or add to, a fire hazard need not comply with the requirements of paras 3.7 and 3.8.

3.7.4 Components and systems, within a designated fire zone shall be fireproof if, when exposed to or damaged by fire, they could result in fire spreading to other regions of the rotorcraft or cause unintentional operation of, or inability to operate, essential services or equipment.

3.7.5 Except for integral oil sumps, no tank or reservoir that is a part of a system containing flammable fluids or gases shall be in a designated fire zone unless the fluid contained, the design of the system, the materials used in the tank, the shut-off means, and all connections, pipes and control provide a degree of safety equal to that which if the tank or reservoir were outside such a zone.

3.7.6 Absorbent materials near to flammable fluid system components that might leak shall be covered or treated to prevent the absorption of hazardous quantities of fluids.

3.8 DRAINS, VENTS AND VENTILATION

3.8.1 There shall be complete drainage of each designated fire zone to minimise the hazards resulting from failure, malfunction or combat damage of any component containing flammable fluids. The drainage means shall be:

(i) Effective under conditions expected to prevail when drainage is needed, and

(ii) Arranged so that no discharged fluid will cause an additional fire hazard.

The requirement of (i) and (ii) need not be applied in a design case where it is impossible to prevent fuel leakage into a fire (e.g., projectile penetration of fuel tank wall).

3.8.2 Each designated fire zone shall be ventilated to prevent the accumulation of flammable vapours, under all rotorcraft design conditions.

3.8.3 No ventilation opening shall be where it would allow the entry of flammable fluids, vapours, or flame from other compartments.

3.8.4 Each ventilation means shall be arranged so that no discharge vapours will cause an additional fire hazard.

3.8.5 Unless the extinguishing agent capacity and rate of discharge are based on maximum air flow through a zone, there shall be means to allow the crew to shut-off sources of forced ventilation to any designated fire zone.

3.8.6 It shall be demonstrated that the ventilation and drainage arrangements are adequate under all rotorcraft design conditions.

3.9 FLAMMABLE FLUID SHUT-OFF

3.9.1 Provision shall be made for shutting off the flow of hazardous quantities of flammable fluids into or through each designated fire zone, except that shut-off means are not required for:

(i) Pipes, lines or ducts forming an integral part of the engine, and

(ii) Systems in which all external components of the system including oil tanks are fireproof.

This requirement need not be applied in a design case where prevention of leakage into a fire is impossible (e.g., projectile penetration of fuel tank wall or components carrying flammable fluids).

3.9.2 The closing of the shut-off valve for any engine shall not affect the fuel available to the remaining engine(s) or interfere with the later emergency operation of other equipment.

3.9.3 Each flammable fluid shut-off valve/control shall be located so that any fire in a designated fire zone will not affect its operation.

3.9.4 Each tank to engine shut-off valve/control shall be located so that the operation of the valve will not be affected by power-plant or engine mounting structural failure and means shall be provided to relieve excessive pressure accumulation.

3.10 FIREWALLS

3.10.1 Each engine, APU, combustion heater and other fuel burning equipment intended for operation in flight and the combustion turbine and exhaust system section of turbine engines shall be isolated from the rest of the aeroplane by firewalls, shrouds, or equivalent means.

3.10.2 Each firewall and shroud shall be:

(i) Fireproof

(ii) Constructed so that no hazardous quantity of air, fluid, or flame can pass from the compartment to other parts of the rotorcraft.

(iii) Constructed so that each opening is sealed with close fitting fireproof grommets, bushes or firewall fittings, and

(iv) Protected against corrosion.

3.10.3 Systems passing through firewalls shall be fireproof to maintain the integrity of the firewall.

3.11 FIRE DETECTION AND FIRE WARNING

3.11.1 There shall be a rapid response optical surveillance and/or continuous heat detectors in each designated fire zone, and in the combustion, turbine and exhaust system section of turbine engine installations to ensure prompt detection of fire in those zones. The system shall be of the resetting type.

3.11.2 The fire detection system shall provide the crew with an adequate visual and audible warning of fire and these warnings shall be incorporated in the standard warning system defined in Chapter 107, paras 14.2 and 14.5.

3.11.3 Each fire detector system shall be constructed and installed so that:

(i) It will withstand the vibration, inertia and other loads to which it may be subjected in operation.

(ii) There is a means to warn the crew in the event that the sensor, control unit, or associated wiring within a designated fire zone is severed at one point, unless the system continued to function as a satisfactory detection system after the severing.

(iii) There is a means to warn the crew in the event of a short circuit in the sensor, control unit or associated wiring within a designated fire zone unless the system continues to function as a satisfactory detection system after the short circuit.

3.11.4 No fire or overheat detector shall be affected by contamination by any oils, fuels, water, other fluids or fumes that might be present.

3.11.5 There shall be means to allow the crew to check in flight, the function of each fire or overheat detector electrical circuit.

3.11.6 Wiring and other components of each fire or overheat detector system in a designated fire zone shall be at least fire resistant.

3.11.7 Detector system components for any one designated fire zone shall not pass through or be close to other designated fire zones, unless they are protected against false warnings and being rendered inoperative from fires in such zones. This requirement shall not be applicable with respect to zones which are simultaneously protected by the same warning and extinguishing system.

3.11.8 Each fire detection system shall be constructed so that when installed it will not exceed the alarm activation time approved for the detectors using the response and sensitivity criteria quoted in the specification for the detector type.

3.11.9 Fire detector systems shall be designed and installed so that their efficiency is not impaired by exposure to environmental temperatures encountered in normal operation, solar or other false warning stimuli which may be present.

3.12 FIRE EXTINCTION

3.12.1 There shall be a fire extinguishing system serving each designated fire zone, except for combustion, turbine and tail pipe sections of turbine engine installations that contain pipes or components carrying flammable fluids or gases for which it is shown that a fire originating in these sections can be controlled.

3.12.2 Fire systems may not be needed where engine(s) are uncowled or mounted in free air. In such cases it shall be demonstrated that an engine fire will not affect the functioning of essential systems or give rise to a catastrophic structural failure.

3.12.3 Except for auxiliary power units or combustion heater installation and certain other exceptions agreed with the Rotorcraft Project Director, it shall be possible to direct a second adequate discharge of extinguishant to any designated fire zone for which the first discharge of extinguishant has been used. The extinguishant provided for another designated zone may be used for this purpose provided that selection of zones requires distinct action.

3.12.4 There shall be a fire extinguishing system serving the APU, fuel burning heater, and other combustion equipment. Each of these units shall have an individual system which may be an individual 'one-shot' system. When the APU is not used in flight except in an emergency, and the installation is readily accessible to groundcrew the provision of a fire extinguishing system may be waived at the discretion of the Rotorcraft Project Director.

3.12.5 Fire detectors shall not operate main engine power unit extinguisher systems automatically unless the extinguishing systems are designed to respond to normal and combat induced fires via a crew operated mode selection switch.

3.12.6 Automatic operation of the fire extinguishing system under crash conditions shall be provided (also Chapter 307).

3.12.7 It shall be demonstrated by test that the required concentration of extinguishant is achieved simultaneously at all parts of the compartment (see para 3.8.5 and Leaflet 712/2 para 4).

3.12.8 A visual detector shall be provided to indicate to maintenance personnel that the extinguishant has been discharged.

3.13 COMPLIANCE

3.13.1 Unless otherwise specified, compliance with the requirements of para 3 shall be shown by a full-scale fire test or by one or more of the following methods:

(i) Tests of similar powerplant configurations.

(ii) Tests of components or systems.

(iii) In Service experience of rotorcraft with similar powerplant configurations.

4 PRECAUTIONS IN OTHER ZONES

4.1 For precautions relating to combat induced fires see para 11.

4.2 FLAMMABLE FLUID FIRE PROTECTION

4.2.1 In each area where flammable fluids or vapours might escape by leakage of as fluid system, there shall be means to minimise the probability of ignition of the fluids and vapours, and the resulting hazards if ignition does occur.

4.2.2 Compliance with para 4.2.1 shall be shown by analysis or tests and the following factors shall be considered.

(i) Possible sources and paths of fluid leakage, and means of detecting leakage.

(ii) Flammability characteristics of fluids, including effects of any combustible or absorbent materials.

(iii) Possible ignition sources, including electrical faults, overheating of equipment, and malfunctioning of protective devices.

(iv) Means available for controlling or extinguishing a fire, such as stopping flow of fluids, shutting down equipment, fireproof containment, or use of extinguishing agents.

(v) Ability of rotorcraft components that are critical to flight safety to withstand fire and heat.

4.2.3 If action by the flight crew is required to prevent or counteract a fluid fire (e.g., equipment shut-down or actuation of a fire extinguisher) quick acting means shall be provided to alert the crew.

4.2.4 Each area where flammable fluids or vapours might escape by leakage of a fluid system shall be identified and defined.

4.3 Flammable fluid drains, vents and ventilation shall comply with the requirements of paras 3.7 and 3.8.

4.4 No vent or drainage provision shall end at any point where the discharge of fuel from the vent outlet in flight or on the ground would constitute a fire hazard or from which fumes could enter personnel compartments.

4.5 AREAS ADJACENT TO DESIGNATED FIRE ZONES AND ENGINE NACELLE ATTACHING STRUCTURES

4.5.1 Each area immediately adjacent to a firewall and each portion of any engine nacelle attaching structure containing flammable fluid lines shall meet the requirements of paras 3.7 and 3.8.

4.5.2 Engine nacelle attaching structures need not contain fire detection or extinguishing means.

4.5.3 For each area covered by para 4.5.1 that contains a retractable landing gear compliance with that para need only be shown with the landing gear retracted.

4.5.4 Engine mountings and other critical structures located in designated fire zones or in adjacent areas which would be subject to the effects of fire in the fire zone shall be constructed of fireproof material or shielded so that they are capable of withstanding the effects of fire.

4.6 In addition to complying with the requirements of paras 3.9 and 4.5 for designated fire zones, components and structure used immediately adjacent to the outside face of firewalls and in engine pod attachments shall be of such materials and at such a distance from the designated fire zone that they will not suffer damage that could hazard the rotorcraft if the inner surface of the firewall is enveloped in flames at 1100°C for 15 mins.

4.7 There shall be adequate airspace between each tank or reservoir and each firewall or shroud isolating a designated fire zone such that the possibility of ignition of liquids or vapours is minimised in the event of a fire in the designated fire zone.

4.8 ROTOR BRAKE AND TRANSMISSION SYSTEM INSTALLATIONS

4.8.1 Means shall be provided to eliminate the inadvertent application of the brake when the rotor condition levers are in the flight position or following a component or system failure.

4.8.2 Flammable fluid pipes shall not be routed in the area of the rotor brake where leakage or combat damage could result in a fire developing due to high component temperatures and/or incandescent particles being released on application of the rotor brake.

4.8.3 If compliance with para 4.8.2 is impractical then the relevant flammable fuel pipes and components with fluid content shall be shielded and complete drainage away from the rotor brake area provided or an optical fire detection and rotor brake fire suppression system be installed.

4.8.4 When the rotor brake is powered hydraulically or installed adjacent to transmission gear-boxes or other systems using flammable fluids means shall be provided to ensure that fluids cannot contaminate the rotor brake disc or brake friction pads and that any build up of fluid due to leakage or drainage is drained from the rotor brake area.

5 OXYGEN INSTALLATION

5.1 Oxygen equipment and pipes shall not be located in any designated fire zone.

5.2 Oxygen equipment pipes shall be protected from heat that may be generated in, or escape from, any designated fire zone.

5.3 Oxygen equipment and pipes shall be installed so that escaping oxygen cannot cause ignition of grease, fluid, or vapour accumulations that are present in normal operation or as a result of failure, or malfunction of any system. Design precautions shall be taken to minimise hazards due to battle damage.

5.4 Oxygen pressure sources and pipe lines between the sources and shut-off means shall be protected from unsafe temperatures.

5.5 Pipes carrying flammable liquids shall be positioned at as great a distance as practical from the oxygen installation. Precautions shall be taken to prevent fluid impinging on the oxygen or oxidant system.

6 ELECTRICAL SYSTEM FIRE AND SMOKE PROTECTION

6.1 Electrical components in regions immediately adjacent to firewalls and in engine pod attachment structures shall be of such materials and at such a distance from the firewall that they will not suffer damage that could hazard the rotorcraft if the surface of the firewall adjacent to the fire is heated to 1100°C for 15 minutes.

6.2 Electrical equipment shall be constructed and/or installed so that in the event of failure no hazardous quantities of toxic or noxious (e.g., smoke) products will be distributed in the crew or passenger compartments.

6.3 Electrical equipment which may come into contact with flammable vapours shall be designed and installed to the requirements of BS 3G100 Part 2 Sect 3, Sub-section 3.5, to minimise the risk of the vapours exploding under both normal and fault conditions or as approved by the Rotorcraft Project Director.

6.4 Insulated electrical wire and cable installed in any region of the rotorcraft shall comply with the flammability requirements in BS G212 para 2.28A Flammability Test Method 3, or BS G230, para 6.29.1, Test 28(a) Method 1 as appropriate.

7 INHABITED AREAS

7.1 There shall be at least the following number of hand fire extinguishers conveniently located in passenger compartments:

Passenger Capacity	Minimum Number of Hand Fire Extinguisher
7-30	1
31-60	2
61 or more	3

The number and location of hand fire extinguishers shall be such as to provide adequate availability for use, account being taken of the number and size of the inhabited compartments and the location of toilets and galleys etc. These considerations may result in the number being greater than the minimum prescribed. Hand held fire extinguishers shall comply with para 14.4.

7.2 There shall be at least one hand held fire extinguisher suitable for both flammable fluid and electrical equipment fires conveniently located on the flight decks of aeroplanes normally occupied by 2 or more persons.

7.3 Where a built-in fire extinguishing system is used it shall comply with para 14.5.

7.4 Materials (including finishes or decorative surfaces applied to the materials used on each compartment occupied by the crew or passengers) shall as far as practicable be chosen to minimise the production of smoke or noxious fumes when over-heated or burned. For test requirements see para 11.2.

7.5 A notice stating that smoking is prohibited in the inhabited areas of military rotorcraft shall be displayed in the appropriate compartments of the rotorcraft.

7.6 Each disposal receptacle for towels, paper, or waste shall be fully enclosed and constructed of at least fire resistant materials and shall contain fires likely to occur in it under normal use. The ability of the disposal receptacle to contain these fires under all probable conditions of wear, misalignment, and ventilation expected in service shall be demonstrated by test.

7.7 Lavatories shall have "No Smoking" or "No Smoking in Lavatory" notices located conspicuously on each side of the entry door.

7.8 'No Smoking' notices shall comply with Chapter 103 para 5. A 'No Smoking' symbol may be included in the notice.

8 CARGO AND BAGGAGE COMPARTMENTS

8.1 Compartments not occupied by passengers or crew shall be constructed of, or lined with, materials that are at least equivalent in fire resistance to that of the adjacent airframe structural materials.

8.2 Compartments inaccessible in flight shall be designed so that fires will be contained until a safe landing can be made, or equipped with fire detecting, indicating and extinguishing apparatus unless it can be demonstrated that the presence of fire will be made known immediately.

8.3 Instructions for the user of extinguishing apparatus fitted in compliance with para 8.2 shall be placarded at the appropriate crew member's station.

8.4 It shall be demonstrated that hazardous quantities of smoke, flame, extinguishing agents or other noxious gases produced as a result of a fire in a cargo or baggage compartment are excluded from any crew or passenger compartment.

8.5 Each cargo or baggage compartment that is not sealed to contain compartment fires completely without endangering the safety of the rotorcraft or its occupants shall be designed or shall have a device to ensure detection of fires by a crew member while at his station and to prevent the accumulation of harmful quantities of smoke, flame, extinguishing agent and other noxious gases in any crew compartment. This shall be demonstrated in flight.

8.6 No compartment shall contain any controls, wiring lines, equipment, or accessories whose damage or failure would affect safe operation, unless those items are protected so that they cannot be damaged by the movement of cargo in that compartment, and their breakage or failure will not create a fire hazard.

8.7 There shall be means to prevent cargo or baggage from interfering with the functioning of the fire-protection features of the compartment.

8.8 Sources of heat within the compartment shall be shielded and insulated to prevent ignition of the cargo. Lights shall be switched off automatically when the cargo doors are closed.

8.9 Cargo compartments shall meet the requirements of para 8.1 above. In addition, flight tests shall be conducted to show:

(i) Compartment accessibility.

(ii) The entry of hazardous quantities of smoke or extinguishing agent into compartments occupied by the crew or passengers.

(iii) The dissipation and concentration of the extinguishing agent in Class C compartments.

(iv) Compartment leakage air flow.

During these tests it shall be shown that no inadvertent operation of smoke or fire detectors in any compartment would occur as a result of fire contained in any one compartment either during or after extinguishment, unless the extinguishing system floods each such compartment simultaneously.

8.6 If the Rotorcraft Specification requires a cargo compartment fire detection system to be installed, then the following shall be met for each cargo compartment with these provisions:

(i) The detection system shall provide a visual indication to the flight crew.

(ii) The system shall be capable of detecting a fire at a temperature significantly below that at which the structural integrity of the rotorcraft is substantially decreased.

(iii) There shall be means to allow the crew to check in flight, the functioning of each fire detector circuit.

(iv) The effectiveness of the detection system shall be demonstrated in all approved operating configurations and conditions.

9 BOMB BAYS

9.1 A separate approved smoke detector or fire detection system shall be installed in bomb bays to give warning at the pilot or flight engineers station.

9.2 If a fire suppression system in the bomb bay is required by the Rotorcraft Specification:

(i) An approved built-in fire extinguishing system shall be installed controllable from the pilot or flight engineers station

(ii) There shall be means to exclude hazardous quantities of smoke, or extinguishing agent, from any compartment occupied by the crew or passengers.

(iii) There shall be means to control ventilation and draughts within the bomb bay so that the extinguishing agent used can control any fire that may start in the bomb bay.

(iv) Flight tests agreed by the Rotorcraft Project Director shall be conducted to demonstrate that hazardous quantities of smoke or extinguishing agent are prevented from entering into compartments occupied by the crew or passengers (for dissipation of the extinguishing agent see Leaflet 712/2).

10 HAZARDOUS SYSTEMS

10.1 High speed rotating equipment shall be examined for potential fire hazards such as the ignition of flammable fluids by high casing temperatures during normal operation or following a failure. Protection shall be provided if a hazard exists. Such equipment shall be adequately supported. High speed drive shafts shall be encased, if necessary to protect flammable fluid components, fuel tanks, explosives, oxygen containers etc.

10.2 High pressure air compressors and related equipment shall be located, whenever possible, to minimise damage to flammable fluid components, explosives, oxygen containers and fuel tanks, by flying fragments in case of any explosion.

10.3 Explosives shall comply with the fire protection requirements of the following specifications:

Installation and test of Aircraft Pyrotechnic Equipment, General Spec for MIL-I-8672

Design and Evaluation of Cartridges for Cartridge Actuated Devices MIL-D-21625

Initiators, Electric, Design and Evaluation of MIL-I-23654

Ordnance Board Proceedings 41273 dated November 1972

10.4 Explosives shall not be installed or stowed in the proximity of heat sources if these heat sources can cause ignition of the explosive under any normal condition, or if a "single failure" can cause ignition of the explosives. Explosives shall not be installed or stowed in the proximity of fire zones. If explosives must be located close to real or potential heat sources for justifiable reasons they shall be adequately protected by permanently installed insulation or shrouds.

10.5 Cartridges used in Cartridge Starter Systems shall comply with the fire protection requirements of MIL-D-21625.

10.6 Fuel tanks shall not be located immediately adjacent to gun compartments; they shall be separated from such compartments by at least one liquid and vapour tight bulkhead in addition to the tank boundary structure.

10.7 External rockets shall be installed so that the rocket exhaust will not be a hazard to fuel tank vent lines. If fuel vent exits cannot be located at a safe distance from the rocket exhaust, the vent line exits shall be protected by flame arrestors or other effective means to prevent flame propagation into the tanks.

10.8 Protection shall be provided, if necessary, for flammable fluid components, fuel tanks, explosives etc., located in compartments exposed to the rocket exhaust wake.

10.9 Decoy flare dispensing systems shall be installed so that the ignited flare will not be a hazard to flammable fluid drain and fuel tank vent lines. If such drain or vent lines cannot be located at a safe distance from the ignited flare the fluid exits shall be protected by flame arrestors or other effective means to prevent flame propagation into the fluid systems.

10.10 Hot bleed air ducts and other hot gas ducts and components which can be an ignition source due to high surface temperatures or to leaking hot air or gas shall not be located in compartments containing flammable fluid components, unless suitable safeguards are provided e.g., insulation, shields, overheat sensors.

10.11 Sealing shall be provided to prevent the passage of flammable vapours into or out of the gun bay other than through a purpose designed venting system.

10.12 Appropriate measures shall be taken to ensure that the gas-air mixture within the gun compartment does not fall within the explosive range.

11 PRECAUTIONS: COMBAT INDUCED FIRES

11.1 Components shall be designed and located to minimise the risk of ignition from battle damage.

11.2 If fire suppression in dry bay areas adjacent to flammable fluid tanks is specified by the Rotorcraft Project Director as a result of studies undertaken in compliance with para 2.1, the following requirements shall be met:

(i) Void filling materials used in passive combat fire suppression installations shall comply with the requirements of Specification DTD 5624. It is essential that the void is completely filled by using either shaped blocks, cubes or spheres of the filling material used. (also Leaflet 712/3 para 5.2).

(ii) Active fire protection measures shall be installed in dry bays which are congested due to the fitment of systems components, fluid lines, electrical looms, etc., or where the dimensions, geometry or maintenance considerations make the embodiment of void fillers, vapour or powder packs impractical.

(iii) The method of fire detection incorporated in active systems installations in which the rapid suppression of incipient fires is essential shall be by optical surveillance or dynamic pressure sensing.

(iv) Optical surveillance fire detectors shall conform to the Design Specification for Radiation Sensing Combat Fire Detectors and Amplifiers (non-discriminating) issued by Director RAE.

(v) Automatic operation of the fire suppression system by the detector(s) or crash trip/switch shall be provided for all combat fire protection active system installations.

(vi) No crew fire warning is required for automatically activated dry bay fire suppression systems, but an indication shall be provided for servicing personnel that the system has operated.

(vii) Fluid drainage and the ventilation of dry bay areas shall meet the requirements of para 3.8 above.

(viii) A fire suppression system shall be provided for each group of tanks.

(ix) Dry bay boundary walls shall be at least fire resistant.

11.3 Fuel lines shall be routed, as far as is practical, through fuel tanks and close to heavy structure to minimise the fire and explosion risk due to combat damage.

11.4 Where fuel tanks are located above an engine compartment a drained and ventilated interspace shall be provided.

11.5 A fuel tank and/or vent pipe inerting/in situ flame arrestor system shall be incorporated in the rotorcraft if the requirement for explosion protection has been established from studies undertaken in compliance with para 2.1, specified in the Rotorcraft Specification or by the Rotorcraft Project Director. Fuel tank and/or vent pipe inerting/in situ flame arrestor systems shall comply with the following requirements:

(i) When in situ flame arrestors are specified the requirements of DTD Specification 5627 shall be met. It is essential that the void is filled completely by using either shaped blocks, cubes or spheres of the filling material (Leaflet 712/3 para 5.1.1).

(ii) The inflow of any inerting agent to the tank or vent pipe, the distribution of the agent, and the oxygen evolution from the bulk fuel shall be so controlled that the maximum allowable pressure will not be exceeded.

(iii) The protection shall be effective for the time and conditions specified in the Rotorcraft Specification or as approved by the Rotorcraft Project Director.

(iv) When combustor gas is used as the inerting agent, a flame arrestor shall be embodied to prevent flame from entering the fuel tank and a check valve shall prevent the flow of fuel or fuel vapour into the inerting gas supply system.

(v) The temperature of the inerting agent at the tank inlet shall not exceed 50°C.

(vi) Inerting systems shall be so designed that serviceability can be pre-flight checked and that failure of the system to maintain the oxygen concentration in the ullage space at or below the safe design limits shall be indicated to the crew during flight.

12 COMPARTMENT INTERIORS - TEST CRITERIA

12.1 Materials (including finishes or decorative surfaces applied to the materials) used in each compartment occupied by the crew or passengers shall meet the following test criteria, as applicable:

12.2 Interior ceiling panels, interior wall panels, partitions, galley structure, large cabinet walls, structural flooring and materials used in the construction of stowage compartments (other than underseat stowage compartments and compartments for stowing small items such as magazines and maps) shall be self-extinguishing where tested vertically in accordance with the applicable portions of Leaflet 712/4 (JAR 25.853 Appendix F) or other equivalent methods approved by the Rotorcraft Project Director. The average burn length shall not exceed 150 mm and the average flame time after removal of the flame source shall not exceed 15 secs. Drippings from the test specimen shall not continue to flame for more than an average of 3 secs after falling.

12.3 Floor covering, textiles (including draperies and upholstery), seat cushions, padding, decorative and non-decorative coated fabrics, leather, trays and galley furnishings, electrical conduit, thermal and acoustic insulation and insulation covering, air ducting, joint and edge covering, cargo compartment liners, insulation blankets, cargo covers, and transparencies, moulded and thermoformed parts, air ducting joints, and trim strips (decorative and chafing), that are constructed of materials not covered in para 12.4 shall be self extinguishing when tested vertically in accordance with the applicable portions of Leaflet 712/4 (JAR 25.853 Appendix F) or other equivalent methods approved by the Rotorcraft Project Director. The average burn length shall not exceed 200 mm and the average flame time after removal of the flame source shall not exceed 15 secs. Drippings from the test specimen shall not continue to flame for more than an average of 5 secs after falling.

12.4 Acrylic windows and signs, parts constructed in whole or in part of elastomeric materials, edge lighted instrument assemblies consisting of two or more instruments in a common housing, seat belts, shoulder harnesses, and cargo and baggage tiedown equipment, including containers, bins, pallets, etc., used in passenger or crew compartments shall not have an average burn rate greater than 64 mm per min when tested horizontally in accordance with the applicable portions of Leaflet 712/4 (JAR 25.853 Appendix F) or other equivalent methods approved by the Rotorcraft Project Director.

12.5 Except for electrical wire and cable insulation, and for small parts (such as knobs, handles, rollers, fasteners, clips, grommets, rub strips, pulleys and small electrical parts) that would not contribute significantly to the propagation of a fire, materials in items not specified in paras 12.2, 12.3 and 12.4 shall have a burn rate not greater than 100 mm per min when tested horizontally in accordance with the applicable portions of Leaflet 712/4 (JAR 25.853 Appendix F) or other equivalent methods approved by the Rotorcraft Project Director.

12.6 In addition to meeting the flammability requirements of para 12.3 each seat cushion (squab and back support) except where fitted to seats located within the flight deck shall meet the requirements of FAR 25, Appendix F, Part 11 (at amendment 25.59) or an equivalent fire test criteria agreed with the Rotorcraft Project Director (CAA Airworthiness Notice No. 59).

13 CARGO COMPARTMENTS: THERMAL AND ACOUSTIC INSULATION

13.1 Thermal and acoustic insulation (including coverings) and liners used in cargo and baggage compartments not occupied by passengers or crew, shall be constructed of materials that at least meet the test requirements of para 12.3.

(i) Liners must be constructed of materials that at least meet the requirements set forth in para 12.3, must be separate from (but may be attached to) the rotorcraft structure, and shall be tested at a 45^{o} angle in accordance with the applicable portions of Leaflet 712/4 (JAR 25.853 Appendix F) or other equivalent methods approved by the Rotorcraft Project Director. In the course of the 45° angle test, the flame shall not penetrate (pass through) the material during application of the flame or subsequent to its removal. The average flame time after removal of the flame source shall not exceed 15 secs and the average glow time may not exceed 10 secs.

(ii) Insulation blankets and cargo covers used to protect cargo in compartments not occupied by passengers or crew must be constructed of materials that at least meet the requirements of para 12.3 and tiedown equipment (including containers, bins and pallets) used in each cargo and baggage compartment not occupied by passengers or crew must be constructed of materials that at least meet the requirements set forth in para 12.5.

14 MISCELLANEOUS FIRE PRECAUTIONS

14.1 Pipes, tanks or equipment containing flammable fluids shall not be installed in passenger, crew, cargo or baggage compartments unless adequately shrouded and the interspace vented and drained.

14.2 Whenever lagging is used in compartments in which pipes, tanks or equipment containing flammable fluids are installed, suitable precautions shall be taken to prevent the wetting of the lagging by flammable fluids as a result of normal operation, damage, failures of the equipment or leakages from joints or unions.

14.3 No essential service outside any designated fire zone shall be rendered ineffective by a fire within the fire zone.

14.4 For hand held fire extinguishers the following apply:

(i) Each hand held fire extinguisher including bracket must be approved to BS 5423.

(ii) The types and quantities of extinguishing agent used shall be appropriate to the kinds of fires likely to occur and the materials involved. The use of Methyl Bromide is prohibited.

(iii) Hand held fire extinguishers shall be easily accessible and mounted to facilitate quick removal, and their locations marked in accordance with AP119A-0601 Chapter 3.

(iv) Extinguishers used in personnel compartments shall be designed to minimise the hazard of toxic gas concentrations.

(v) The design of the extinguisher bracket shall be such as to minimise the possibility of inadvertent opening of the release by snagging on clothing etc.

14.5 If a built-in fire extinguishing system is specified:

(i) The capacity of each system, in relation to the volume of the compartment where used and the ventilation rate, shall be adequate for any fire likely to occur in that compartment.

(ii) Each system shall be installed so that no extinguishing agent likely to enter personnel compartments will be hazardous to the occupants.

(iii) Discharge of the extinguisher shall not cause structural damage.

(iv) The use of Methyl Bromide is prohibited.

14.6 FIRE ACCESS PANELS

14.6.1 Spring loaded fire access doors in the main power plant and auxiliary power unit bays shall be provided as defined in the Rotorcraft Specification. The spring system shall be sufficiently strong to hold the door shut against air loads and the door shall be marked "Fire Access" (see also Leaflet 712/2 para 7).

14.7 EXTERNAL ACTUATION OF FIRE EXTINGUISHING SYSTEMS ON MULTI ENGINES ROTORCRAFT

14.7.1 A means shall be provided for operating the rotorcraft fire extinguishing systems when the rotorcraft is on the ground with the batteries disconnected or removed and the normal external power supply not connected. This requirement shall apply to the fire extinguishers for all designated fire zones and to the fire extinguisher system which provides external protection for flammable fluid tanks, but need not apply to bomb bay, cargo and baggage bays and pyrotechnic installations in the rotorcraft.

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FIRE PRECAUTIONS

REFERENCE PAGE

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DTD/RDI 3964	.. ..	Electrically-driven fuel booster pumps for aircraft low pressure fuel systems
*DTD/RDI 3968	.. ..	General requirements for crash resistant self-sealing and non-self-sealing flexible fuel tanks for use in fuel systems installed in aircraft
DTD 5627	.. ..	Fibrous polyamide material for use as an explosion suppressant and as a baffle material in aircraft fuel tanks
DTD 5624	.. ..	Fibrous polyester material for use as a flame and fire suppressant in aircraft dry bays
*DTD/RDI 3966	.. ..	General requirements for flexible tanks for use in fuel and water methanol systems installed in aircraft
*DTD/RDI 3967	.. ..	General requirements for self-sealing covers for flexible fuel tanks for use in fuel systems installed in aircraft
D Eng RD 2453 Avtur	..	Aviation Turbine Fuel (with FSII). NATO Code F34
D Eng RD 2494 Avtur	..	Aviation Turbine Fuel. NATO Code F35
D Eng RD 2454 Avtag	..	Aviation Turbine Fuel.(with FSII) NATO Code F40
D Eng RD 2490	.. ..	Aviation Turbine Oil. NATO Code 0.135
D Eng RD 2491	.. ..	Thrust Augmentation Water Methanol NATO Code S1744
D Eng RD 2497	.. ..	Synthetic Lubricating Oil. NATO Code 0.160
D Eng RD 2498 Avcat	..	Aviation Turbine Fuel. NATO Code F43
D Eng RD 2485 Avgas	..	Aviation Petrol. NATO Code F18
D Eng RD 2492 Avpin	..	Isopropyl Nitrate Starter Fuel. NATO Code S-746
D Eng RD 2452 Avcat	..	Aviation Turbine Fuel (with FSII). NATO Code F44

*In process of being superseded by DEF STAN 15-2.

<u>Defence Standards</u>

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65138 .. .. Spontaneous Ignition of Kerosene (AVTUR) Fuel Vapour: the effect of vessel size

65191 .. .. Detection and Measurement of Flammable Vapours in Aircraft

67107 .. .. Spontaneous Ignition of Kerosene (AVTUR) Vapour: the effect of the ratio: vessel surface area to volume

67162 .. .. Ignition of Aviation Fuel by Hot Pipes.

69002 .. .. Survey of Aviation Kerosene Vapour Pressures and Flash Points

69120 .. .. Reticulated Material for Prevention of Internal Fuel Tank Explosions

70044 .. .. An assessment of Explosion Protection of Fuel Tanks with Reticulated Foam Filling

71012 .. .. Spontaneous ignition of Avtur vapour in various oxygen - nitrogen mixtures

71194	.. ..	Environmental Tests of Reticulated Plastic Foam for use in Aircraft Fuel Tanks
72059	.. ..	Spontaneous ignition of aircraft fuels
72199	.. ..	An assessment of the spontaneous ignition hazards of M86 hydrazine monofuel
73068	.. ..	Fuel Tank Explosion Suppression with Block and Ball Fillings of Reticulated Plastic Foam
74066	.. ..	Spontaneous ignition of aircraft hydraulic fluids
80002	.. ..	Examination of a Fibrous Flame Suppressant for Explosion Protection of Aircraft Fuel Tanks

<u>RAE Technical Notes</u>

Mech Eng 96	.. ..	The effect of Air Evolution from Fuel on the Inert Gas Protection of Aircraft Fuel Tanks
Mech Eng 210	.. ..	Treatment of Aircraft Fuel with Nitrogen to reduce Tank Explosion Hazards (Parts I, II and III)
Mech Eng 241	.. ..	Flight Tests of Valiant Nitrogen System using Normal and Treated Fuels
Mech Eng 273	.. ..	An Investigation into the effectiveness of Existing and Experimental Anti-fire Installations for Aircraft
Mech Eng 341	.. ..	Spontaneous Ignition within a Heated Fuel Tank when Pressured with Air under Simulated Diving Flight Conditions
Mech Eng 379	.. ..	Spontaneous Ignition of Kerosene (Avtur) Fuel Vapour within a 4 inch Cylindrical Vessel
Mech Eng 393/ CPM 41	.. ..	The Detection and Measurement of Oxygen in Aircraft Fuel Tanks
Mech Eng 120	.. ..	Installation of Lightweight Flexible Fuel Tanks

<u>RAE Test Notes</u>

Test Note EP 1094	.. ..	Damage to Liquid Filled Pipes and Surrounding Structure by 12.7mm API/T Ammunition
Test Note EP 1098	.. ..	Initiation and suppression of Fuel Based Fires resulting from Small Arms Fire against Fuel filled Pipes

<u>British Standards</u>

BS 5423	.. ..	Portable Fire Extinguishers

BS C6	..	..	Aircraft engine nacelle fire extinguisher doors
BS 3G 100	..	..	Specification for general requirements for equipment for use in Aircraft
BS G 212	..	..	General requirements for aircraft electrical cables
BS G 230	..	..	Specification for general requirements for aircraft electrical cables

MIL Standards

MIL-B-83054	..	..	Baffle Material for Aircraft Fuel Tanks
MIL-P-46111A	..	..	Plastic Foam, Polyurethane for Aircraft Use
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FAA Technical Standard Orders

TSO-C11d	..	..	Fire Detectors, Thermal Sensing and Ionisation Sensing Types

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20-53A	..	..	'Protection of Aircraft Fuel Systems against Fuel Vapour Ignition due to Lightning'

20-100	..	..	General Guidelines for Measuring Fire Extinguishing Agent Concentrations in Power Plant Compartments

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AS 401	..	..	Minimum Requirements for Fire and Heat Detection Instruments for use in Aircraft

FAR Requirements

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BCAR Section G Rotorcraft

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AFWAL-TR-80-	..	..	Evaluation of Explosafe Explosion Suppression System for Aircraft Fuel Tank Protection

AGARD

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LEAFLET 712/1

FIRE PRECAUTIONS

DEFINITIONS

1 INTRODUCTION

1.1 This Leaflet gives definitions of the terms used in Chapter 712 and the other Leaflets on fire precautions.

2 DEFINITIONS

2.1 DESIGNATED FIRE ZONE

2.1.1 A designated fire zone is defined as a region in which a single failure of an installation or any part of it could result in a fire or breakout of an existing controlled fire (e.g., combustion chamber) into the rotorcraft.

2.2 FIRE PROOF

2.2.1 Fireproof with respect to components and equipment means the capability to withstand for a period of 15 mins, the application of heat by the standard flame.

2.2.2 In respect of materials and structural parts, means the capacity to withstand the heat associated with fire at least as well as steel in dimensions appropriate for the purpose for which they are used.

2.3 FIRE RESISTANT

2.3.1 Fire resistant with respect to components and equipment means the capability to withstand for a period of 5 mins the application of heat by the standard flame.

2.3.2 With respect to sheet materials and structural members, means the capacity to withstand the heat associated with fire at least as well as aluminium alloy in dimensions appropriate for the purposes for which they are used.

2.4 STANDARD FLAME BURNER

2.4.1 A burner giving a nominal flame temperature of 1100°C and the characteristics specified in ISO/TR 2685.

2.5 FLAME RESISTANT

2.5.1 Flame resistant means not susceptible to combustion to the point of propagating a flame, beyond safe limits, after the ignition source is removed.

2.6 FLAMMABLE MATERIALS

2.6.1 Flammable Materials are materials which will ignite readily or explode.

2.7 FLASH RESISTANT

2.7.1 Flash resistant means not susceptible to burning violently when ignited.

2.8 SELF EXTINGUISHING

2.8.1 A substance is self-extinguishing if it ceases to burn within a given time after removal of the igniting source.

2.9 TORCHING FLAME

2.9.1 A torching flame is defined as the flame characteristic of that breaking-out from the primary zone of a defective engine combustor and derived from the ignition of a rich mixture of kerosene and air. (See also BS 3G100 Part 2: Section 3 Section 3.13: 1973 para 2 and 4.2).

2.10 TORCHING FLAME RESISTANT

2.10.1 Components and equipment to this grade shall be capable of withstanding for a period of at least 2 mins, the application of heat by a torching flame without any malfunction that would jeopardize the safety of the rotorcraft, or failure that would aggravate an existing hazard (BS 3G100).

2.11 CONTINUOUS HEAT DETECTORS

2.11.1 Continuous heat detectors are those employing continuous lengths of heat sensing elements connected to a monitoring device.

2.12 OPTICAL SURVEILLANCE DETECTORS

2.12.1 Optical Surveillance detectors are radiation sensing devices which operate in the ultra-violet and/or infra-red wave bands.

2.13 HAZARDOUS PROXIMITY OF POTENTIAL IGNITION SOURCES

2.13.1 Potential ignition sources in proximity to flammable fluid lines, containers or components such that fluid leakage could result in the development of a fire which would hazard the safety of the rotorcraft.

2.14 ALARM ACTIVATION TIME

2.14.1 Alarm activation time is the time taken from the initial emission of radiation or heat from a specified fire source and the activation of the fire warning.

2.15 VULNERABILITY

2.15.1 See Chapter 112.

2.16 ESSENTIAL CONTROLS AND SERVICES

2.16.1 Essential controls and services are those controls and services essential for the safe operation of a rotorcraft during and after the extinguishing of a fire and include:

(i) fire detection system.

(ii) aerodynamic controls.

(iii) controls needed for emergency services.

(iv) fuel supply and controls.

(v) cooling air control and actuating gear.

(vi) oil supply and control.

LEAFLET 712/2

FIRE PRECAUTIONS

GENERAL RECOMMENDATIONS

1 INTRODUCTION

1.1 This leaflet gives recommended or acceptable methods of meeting certain of the basic requirements stated in Chapter 712.

2 DESIGNATED FIRE ZONE ISOLATION (See Chapter 712 para 3.10)

2.1 Firewalls should be made from stainless steel or titanium at least 0.4mm (0.016in) thick, and their outer surfaces should not be painted.

2.2 Firewalls may distort severely under heat and hence access doors and joints should be avoided. If this is not practical then closely spaced fasteners should be used. High strength fasteners with aluminium parts are not recommended.

2.3 Firewalls should be capable of sustaining loads likely to be experienced during their service life.

2.4 Attention is drawn to the difficulties in handling light gauge material which may be specified for firewalls and to the chances of damage during servicing particularly where there is any appreciable unsupported area.

2.5 Air ducts passing through a firewall might give a fire the opportunity to travel from one fire zone to another or to the rest of the rotorcraft and may also allow a high air mass flow to enter the fire zone and feed the fire. It is recommended that such air ducts be made of stainless steel or titanium not less than 0.40mm thick or equivalent material for a sufficient distance beyond the fire barrier to ensure that any fire can be contained within the duct. Shut off means should be provided to prevent air being fed into the designated fire zone.

2.6 Precautions should be taken to limit the consequences of an explosion in a fire zone. Compartment walls should be strong enough to withstand a pressure differential and, if natural relief is not provided, pressure relief panels should be considered. Ducts inside a fire zone, e.g., intake duct, exhaust pipe, should be capable of withstanding an explosion without hazarding the safety of the rotorcraft.

3 FLAMMABLE FLUID TANKS

3.1 Where possible flammable fluid tanks should be located so that they will not be damaged by an uncontained failure of an engine, or so that a collapsing undercarriage does not cause a major tank leak.

3.2 For protection against lightning see Chapter 708.

4 FIRE EXTINGUISHING SYSTEM

4.1 The means by which the effectiveness of the fire extinguishing system can be demonstrated will be agreed with the Rotorcraft Project Director, but should normally be confined to the measurement of the rate of discharge of the extinguishant and distribution of the discharged extinguishant. In certain circumstances rig tests or a fire tunnel test including the extinguishing of representative engine fires may be required by the Rotorcraft Project Director. Acceptable methods of establishing the adequacy of the fire extinguishing system are given in FAR Advisory Circular 20-100.

4.2 Recommended extinguishants are bromochlorodifluoromethane (BCF) or, where a high discharge rate is required, bromotrifluoromethane (BTM). The extinguishing agent should have thermal stability over the temperature range likely to be experienced in the compartment in which it is stored.

4.3 To ensure that the minimum concentration of BCF (7.1% by volume) required for extinction of a fire is achieved throughout the fire zone and maintained over the required period a safety factor is applied to give an average concentration by volume of 10.5%. The factored concentration of BCF should be reached within 2 secs, and should be maintained for a minimum period of 2 secs.

4.4 In the case of BTM usage a concentration of at least 6% by volume should be achieved in all parts of the fire zone. This concentration should be reached within 1 sec and be maintained in each part of the fire zone for at least 0.5 secs.

4.5 QUANTITY ASSESSMENT

4.5.1 The quantity of extinguishant required for any one fire zone is governed by:

(i) Zone volume.

(ii) Ventilating air flow.

(iii) Period of average required concentration.

(iv) Period of discharge of extinguishant.

During discharge the average concentration increases in accordance with the following formula:

$$C = \frac{B}{A+B}\left(1-e^{-t(A+B)}\right)$$

C = concentration by volume at time t

A = Ventilating airflow per unit volume of zone in $m^3/sec/m^3$

B = extinguishant discharge rate per unit volume of zone is $m^3/sec/m^3$

t = time from start of discharge in seconds.

4.5.2 When the discharge has been completed, the concentration decreases in accordance with the following formula:

$$C = C_{max}e^{-A(t-t_{max})}$$

C_{max} = concentration at end of discharge
t_{max} = time at end of discharge in seconds.

Note: To obtain the weight of extinguishant required, it may be assumed that at 0°C at a pressure of 1.01325 x 10^5N/m^2, the volume of 1 kg of:

(a) BCF is 0.129 m^3.

(b) BTM is 0.148 m^3.

4.5.3 The amount of extinguishant calculated in accordance with paras 4.5.1 and 4.5.2 may have to be increased by a factor dependent upon the method of discharge. The factor will vary between a value of 1.4 if the method of discharge is a single nozzle in a narrow annular zone the length of which does not exceed approximately two diameters, and a value of 1.0 if the method is a suitably routed spray pipe system. The purpose of this factor is to increase the discharge rate; the discharge period should remain unchanged from the original design value. Discharge nozzles should be located near ventilating air inlets and, in an annular zone, should be positioned to direct the spray spirally around the zone.

4.6 Fire extinguisher bottles are likely to burst if subjected to high temperatures and will normally have a pressure relief device fitted to prevent bursting of the container by excessive internal pressures. Great care should be given in their installation to ensure that there is minimum risk of this occurring before the system has operated.

4.7 Chemical reaction between the extinguishant and the materials used in the fire extinguisher system should not degrade the system.

4.8 A visual indicator should be provided to indicate to maintenance personnel that the extinguishant has been discharged.

5 FIRE DETECTION SYSTEMS

5.1 Fire detection systems should only respond to a fire and the mis-interpretation of a lesser hazard such as engine overtemperature, harmless exhaust gas and bleed air leakage should not be possible. If indication of a lesser hazard is desirable, an independent system should be installed. A fire detection system should be reserved for a condition requiring immediate measures such as engine shutdown, fire extinguishing etc. Detection systems should not be combined with other systems, which if faulty, could prevent normal operation of the detector system. Diodes should not be used to separate detector systems.

5.2 The fire detection system should be designed so that it will:

(i) Indicate fire immediately after ignition and show the compartment in which the fire is located.

(ii) Remain 'ON' for the duration of the fire.

(iii) Indicate when the fire is out.

(iv) Indicate re-ignition of a fire.

5.3 Temperature sensing fire detector elements should be located as close as practicable to sources of flammables such as fuel/hydraulic components and ignition sources such as electric generators and alternators, where the proximity of these flammables and ignition sources constitute a possible source of fire. They should also be located at points where the ventilation air leaves the compartments so that temperature indication can be obtained with a minimum length or minimum number of sensing elements.

5.4 Radiation detectors should be located so that any flame within the compartment is sensed, bearing in mind the scope of vision of the sensor and that reflected flames may be sensed depending on the reflective surface and sensor wave length. Detectors should not be located directly adjacent to combustion sections or any area where in the event of 'burn through' the high temperature would incapacitate the system prior to providing alarm. They should be located so that they will indicate the 'burn through'. The need to shield detectors from seeing reheat flames/weapon flames/engine wet start flames should be considered, also possible changes in performance characteristics at elevated environmental temperatures.

5.5 Acceptable test methods to demonstrate the adequacy of continuous wire type of fire and overheat systems are given in US Mil Spec F-7872 para 4.6.

6 DRAINS AND VENTS

6.1 Drain outlets should be located to avoid any resultant adverse differential pressure that would prevent effective drainage. Drain lines should not normally be manifolded together except at points of overboard drainage. In cases where manifolding is necessary, pressure differentials in drained compartments, or equipment cavities and their possible cause of a fire or explosion, and the ability to identify drained fluids should be carefully considered. Drained fluids which are manifolded should be compatible. All drain line sizes should be such that accumulation of flammable liquids is prevented at the highest leakage rate caused by a single failure.

6.2 Consideration should be given to the possibility of accidental drainage into the jet pipe zone following a false start. Ideally:

(i) the jet pipe or jet pipe shroud if fitted, should protect aft beyond the nacelle.

(ii) the shroud should be sealed to the pipe and to the nacelle and should be separately drained, and,

(iii) the internal and external airflows around the jet pipe zone should always be in a rearward direction.

6.3 Consideration should be given to the elimination of the blockage of vents and drains by ice or snow which may give rise to a fire hazard during engine start-up.

6.4 Drains, vents and fill points should be identified with external markings (see AP119A-0601-0 Chapter 3).

6.5 A container should be provided for collecting fuel drainage on ship-based rotorcraft. The container should be designed and located so that flammable vapours from the container cannot enter the engine compartment or any other compartment which contains a potential ignition source and that the fluid in the container cannot be ignited. If the possibility of ignition in the container cannot be eliminated, the drainage system should be designed to contain a fire without causing a hazard to drained components.

7 ACCESS PANELS

7.1 The access panels for the engine designated fire zone and, where applicable, the APU designated fire zone (see Chapter 712 para 14.6) should be located in a cowling in such a position that when the nozzle fitted to the standard Service ground fire fighting equipment is inserted the extinguishant will effectively reach the engine fuel system and the engine combustion area of the APU.

7.2 Each access panel should be approximately 100mm square (see also BS C6) and should be so designed that a sharp blow with the nozzle will give immediate entry of the nozzle (e.g., a hinged panel split horizontally or vertically and lightly spring loaded in the closed position or a frangible hood of transparent material).

8 APU EMERGENCY STOP

8.1 If an APU is operated without ground crew attendance it is recommended than an emergency stop button be provided for a rapid shut-down.

LEAFLET 712/3

FIRE PRECAUTIONS

COMBAT INDUCED FIRES

1 INTRODUCTION

1.1 This leaflet amplifies the requirements relating to combat induced fire precautions in Chapter 712 para 11.

2 COMBAT INDUCED FIRE HAZARDS

2.1 A large part of a military combat rotorcraft is used for the storage of fuel. It is, therefore, important to assess the risk of ignition of the fuel from combat action and the fire and explosion that could arise.

2.2 It is fundamental to ignition that fuel be vaporised either by expansion following high shock pressures or by the thermal energy of the ignition source. The impact area and the ammunition used are important factors in obtaining the appropriate spark or incendiary initiation. A distinction is drawn between a primary fire which is a direct consequence of projectile impact and a secondary fire which follows flammable fluids leaking into those regions of the rotorcraft containing ignition sources. The likelihood of a primary fire in combat depends on parameters such as the type of fuel, fuel and air temperatures, degree of void ventilation, ammunition used and the rotorcraft configuration.

2.3 Fluid systems are vulnerable to primary fires in compartments and voids adjacent to fuel tanks or the risk of an internal fuel tank explosion.

2.4 The mechanisms of ignition relative to inert fragments/bullets and incendiary/high explosive projectiles impacting a metal airframe is summarised below:

2.4.1 INERT FRAGMENTS AND BULLETS. The fragments or bullet penetrates the rotorcraft structure and a spray of incandescent metallic particles appear at entry and exit lasting approximately 2 milliseconds, the temperature of the flash is about 2300°C. Fragments traverse the intervening air gap penetrating the metal tank wall and producing further incandescent particles. Very high shock pressure is generated in the fuel adjacent to the projectile in fractions of a millisecond. The existence of sub-atmospheric pressure in the wake of the projectile leads to a rapid phase change from liquid to vapour. Thus the initial ejection pulse at the tank wound discharges vapour with such rapidity that it can arrive within the adjacent dry bay during the period when incandescent particles from the initial impact flash have sufficient heat energy to cause an immediate ignition and development of a fire ball.

2.4.2 INCENDIARY AND HIGH EXPLOSIVE PROJECTILES. In this form of attack there is considerable ignition energy available to vaporise the fuel and this increases the fire risk period up to approximately 20 milliseconds. The fire ball associated with the vaporising of the fuel can stabilise into a fuel fed fire, the suppression of which necessitates discharge of the extinguishant within 10 milliseconds of the ignition and the concentration held at least 20 milliseconds.

Projectile penetration of the fuel tank can give rise to in-tank explosion when the stoichiometric ratio of fuel/air in the ullage space attains the appropriate conditions compatible with a rapid rise in pressure and flame front travel.

2.5 The use of composites modifies the mechanism of ignition due to the reduction of incandescent particles following penetration of the structure. Whilst there is no flash generated when a projectile impacts composite structures, the embodiment of numerous structural fastenings, usually manufactured in titanium, and the possibility of encountering multiple fragment strikes may not be significantly different to the probability of ignition associated with metallic structures. Projectiles incorporating incendiary products will generate internal fires irrespective of the type of structural material selected for the airframe.

2.6 Thermal effects from nuclear weapons on the cockpit area makes the crew vulnerable to thermal effects exceeding a level of 5 to 10 Cal/cm^2. Additionally a fire hazard may exist due to the ignition of inflammable materials within the crew compartment and from the discharge of toxic combustion products from smouldering plastics and paints. Higher levels of thermal radiation can have a damaging effect on rotorcraft structures and may give rise to integral fuel tank explosions due to the ignition of flammable vapours.

3 PRECAUTIONS: COMBAT INDUCED FIRES

3.1 In the majority of rotorcraft the following installations and areas will normally be designated as fire zones although it is accepted that in special cases this may not be so:

(i) Piston and Turbine Engine Installation

(ii) Auxiliary Power Plant Installations.

(ii) Combustion Heater Installations.

There is however, the risk that a severe danger of fire may be present in other regions due to the combat environment involving the high probability of a projectile strike. Special consideration should be given to threat induced fire protection with emphasis on the optimum location and shielding of flammable fluid systems and components to threat characteristics and the data derived from rotorcraft vulnerability studies. (See Chapter 112).

3.2 A relocation of equipment could reduce the need for fire precaution due to projectile strike (See Chapter 112).

4 FIRE DETECTION

4.1 If optical detection is used for warning of combat induced fires in dry bays the following should be considered:

(i) Minimising the effects of contamination of the detector heads by oil, dirt etc.

(ii) Locating the detector head and any associated high voltage cables so that a turbine blade, compressor blade, or disc failure could not result in any malfunctioning of the fire detection system.

(iii) Shielding the optical detector head from spurious stimuli associated with the rotorcraft's operational role.

(iv) Ensuring that the polar diagram relevant to the field of view of the detector or its degradation by the intrusion of components and/or structures within the dry bay will not result in the inability to detect a fire.

4.2 RAE studies have shown that severe fires can develop in 4-5 seconds and that the optimum time to extinguish a combat fire is within 1-5 ms of its initiation. Sensors based on detecting ultra violet and/or infra-red radiations emitted from flames have been developed for use in combat fire protection systems. These detectors can respond to impact flashy incendiary products and hydrocarbon fires. They also offer rapid response to fire (1-2 ms) and have a low probability of producing a false warning for a wide range of stimuli.

5 FIRE AND EXPLOSION SUPPRESSION

5.1 EXPLOSION SUPPRESSANT FOAM

5.1.1 If it is decided to use explosion suppression foam to DTD 5627 in fuel tanks consideration should be given to:

(i) Provision of gauze shielding around fuel pumps, level switches, inlet and outlet ports etc., to prevent malfunction due to fouling by the foam, and to minimise the difficulties in removing the foam during maintenance.

(ii) Filling the tank with foam will depend on the tank geometry and type of foam used. Where shaped blocks are used it is recommended that the fill be 110% by volume; for sphere or small cubes a high percentage fill is recommended to eliminate voids.

(iii) Explosion suppressant foam in fuel tanks will attenuate the output signal from any in tank pressure sensor used to operate suppression systems fitted in adjacent dry bays. It may be necessary to amplify the output signal to a level suitable for the fire suppression system.

5.2 Where passive fire suppression foam material to DTD 5624 is used in a congested dry bay area adjacent to fuel tanks, a minimum thickness of 75mm of foam should be attached to the outer surface of the fuel tank walls and retained in place to ensure the void is completely filled and that close contact is maintained between the foam and the tank walls.

5.3 ACTIVE FUEL TANK EXPLOSION SUPPRESSION

5.3.1 Active fuel tank/vent pipe explosion suppression systems rely on the limitation of oxygen concentration in a fuel-air mixture by dilution with an inert gas for the prevention of an explosion or the limitation of explosion pressures. The maximum allowable oxygen concentration in a hydrocarbon fuel-air mixture depends on the total pressure of the mixture before ignition and on the maximum tolerable pressure rise after ignition.

5.3.2 The in-flow of any inerting agent to the fuel tank or vent-pipe should be such that the oxygen content in the ullage or vent-pipe is maintained at a value below 9% under all conditions of flight when the rotorcraft is operating in a combat environment. The oxygen evolution rate is low from a quiescent fuel, but increases rapidly with agitation.

5.3.3 The inerting gas (see Chapter 712 para 11.5) when entering the tank should be free of harmful amounts of water, corrosive material, and material which contaminates the fuel or fuel systems. The gas should not adversely affect pumpability and the burning and electrical characteristics of the fuel.

5.4 The response time of explosion suppression systems from initiation of the explosion to quenching of burning should be such that a safe pressure in the protected tank or pipe is not exceeded.

LEAFLET 712/4

FIRE PRECAUTIONS

AN ACCEPTABLE TEST PROCEDURE FOR SHOWING COMPLIANCE WITH CHAPTER 712 PARAS 12 AND 13

1 CONDITIONING

1.1 Specimens must be conditioned to 21°C ± 5°C, and at 50%, relative humidity until moisture equilibrium is reached or for 24 hours. Only one specimen at a time may be removed from the conditioning environment immediately before subjecting it to the flame.

2 SPECIMEN CONFIGURATION

2.1 Except as provided for materials used in electrical wire and cable insulation and in small parts, materials must be tested either as a section cut from a fabricated part as installed in the aeroplane or as a specimen material or a model of the fabricated part. The specimen may be cut from any location in a fabricated part; however, fabricated units, such as sandwich panels, may not be separated for test. The specimen thickness must be no thicker than the minimum thickness to be qualified for use in the rotorcraft, except that:

(i) Thick foam parts, such as seat cushions, must be tested in 13mm thickness.

(ii) When showing compliance with Chapter 712 para 12.5 for materials used in small parts that must be tested, the materials must be tested in no more than 3mm thickness.

2.2 In the case of fabrics, both the warp and weft direction of the weave must be tested to determine the most critical flammability condition. When performing the tests prescribed in paras 4 and 5 of this Leaflet, the specimen must be mounted in a metal frame so that;

(i) In the vertical tests of para 4 below the two long edges and the upper edge are held securely;

(ii) In the horizontal test of para 5 below the two long edges and the edge away from the flame are held securely;

(iii) The exposed area of the specimen is at least 50mm wide and 304mm long, unless the actual size used in the rotorcraft is smaller; and

(iv) The edge to which the burner flame is applied must not consist of the finished or protected edge of the specimen but must be representative of the actual cross section of the material or part installed in the rotorcraft.

2.3 When performing the test prescribed in para 6 of this Leaflet, the specimen must be mounted in a metal frame so that all four edges are held securely and the exposed area of the specimen is at least 203mm by 203mm.

3 APPARATUS

3.1 Tests must be conducted in a draught free cabinet in accordance with Federal Test Method Standard 191 Method 5903 (revised Method 5902) for the vertical test, or Method 5906 for horizontal test or other approved equivalent methods. Specimens which are too large for the cabinet must be tested in similar draught free conditions.

4 VERTICAL TEST, IN COMPLIANCE WITH CHAPTER 712 PARAS 12.2 AND 12.3

4.1 A minimum of three specimens must be tested and the results averaged. For fabrics, the direction of weave corresponding to the most critical flammability conditions must be parallel to the longest dimension. Each specimen must be supported vertically. The specimen must be exposed to a Bunsen or Tirrill burner with a nominal 10mm inside diameter tube adjusted to give a flame of 38mm in height. The minimum flame temperature measured by a calibrated thermocouple pyrometer in the centre of the flame must be 843°C. The lower edge of the specimen must be 19mm above the top edge of the burner. The flame must be applied to the centre line of the lower edge of the specimen. For materials covered by para 12.2 the flame must be applied for 60 secs and then removed. For materials covered by para 12.3 the flame must be applied for 12 secs and then removed. Flame time, burn length, and flaming time drippings, if any, must be recorded. The burn length determined in accordance with para 7 of this Leaflet must be measured to the nearest 2mm.

5 HORIZONTAL TEST IN COMPLIANCE WITH CHAPTER 712 PARAS 12.4 AND 12.5

5.1 A minimum of three specimens must be tested and the results averaged. Each specimen must be supported horizontally. The exposed surfaces when installed in the rotorcraft must be face down for the test. The specimen must be exposed to a Bunsen burner or Tirrill burner with a nominal 10mm inside diameter tube adjusted to give a flame of 38mm in height. The minimum flame temperature measured by a calibrated thermocouple pyrometer in the centre of the flame must be 843°C. The specimen must be positioned so that the edge being tested is 19mm above the top of, and on the centre line of, the burner. The flames must be applied for 15 seconds and then removed. A minimum of 254mm of the specimen must be used for timing purposes, approximately 38mm must be burnt before the burning front reaches the timing zone and the average burn rate must be recorded.

6 45°C TEST, IN COMPLIANCE WITH CHAPTER 712, PARA 13.1

6.1 A minimum of three specimens must be tested and the results averaged. The specimens must be supported at an angle of 45° to a horizontal surface. The exposed surface when installed in the rotorcraft must be face down for the test. The specimens must be exposed to a Bunsen or Tirrell burner with a nominal 10mm inside diameter tube adjusted to give a flame of 38mm in height. The minimum flame temperature measured by a calibrated thermocouple pyrometer in the centre of the flame must be 843°C. Suitable precautions must be taken to avoid draughts. One third of the flame must contact the material at the centre of the specimen and must be applied for 30 secs and then removed.

Flame time, glow time, and whether the flame penetrates (passes through) the specimen must be recorded.

7 BURN LENGTH

7.1 Burn length is the distance from the original edge to the farthest evidence of damage to the test specimen due to flame impingement, including areas of partial or complete consumption, charring, or embrittlement, but not including areas sooted, stained, warped, or discoloured, nor areas where material has shrunk or melted away from the heat source.

CHAPTER 713

MAGNETIC COMPASS INSTALLATIONS

1 INTRODUCTION

1.1 This chapter deals with the installation of magnetic compasses and states those requirements which shall be met to ensure the satisfactory operation of the compass.

1.2 There are 2 main types of compass in use:

(i) the direct indicating compass, and

(ii) the remote indicating compass.

1.3 Where a remote indicating compass is provided as the primary heading reference, a direct indicating compass shall also be fitted for use as a standby. A high order of accuracy is required for a remote indicating compass which feeds ancillary equipment such as automatic direct reading instruments; this is particularly necessary for navigation computers which combine compass heading with Doppler information.

1.4 A lesser degree of accuracy may be sufficient for a remote indicating compass which does not feed automatic direct reading equipment or for the direct indicating compass (whether this is the normal or standby compass), but reliability is still necessary.

Note: The term standby is taken to include both the normal and emergency uses of the standby compass.

2 POSITION OF COMPASS

2.1 The direct indicating compass shall be installed so that it can be easily read from each pilot's seat position with a minimum of parallax error under all operating conditions of flight and on the ground. In the case of bearing plates, repeaters or compasses, due regard shall be given to the arcs of vision specified for the particular rotorcraft.

2.2 The position of any compass of the set heading type shall be such that each pilot can adjust the grid ring, clamping arm or set heading device without difficulty during flight.

2.3 Adequate clearance shall be allowed around the compass bowl to ensure that the movement of the bowl inside the container is unrestricted.

2.4 Sufficient space shall be provided, where appropriate, for the fitting of a deviation corrector (ie when an internal corrector has not been incorporated in the compass system).

2.5 Easy access shall be provided to the deviation corrector to permit adjustments to be made. The movement of the corrector key shall not be restricted.

2.6 The direct reading compass and the detector unit of a remote indicating compass shall be mounted in such a position that at all times the installation complies with the deviation limits given in Table 2 (See also Para 6). To this end, they shall be mounted in a position as remote as possible from magnetic materials. As a guide, the minimum safe distance of these materials from the detector units of remote indicating compasses is 1.5 metres (5ft).

3 ALIGNMENT OF COMPASS AND CORRECTOR

3.1 A separate deviation corrector shall be provided when an internal unit has not been incorporated in the compass system.

3.2 All direct indicating compasses, the detector units of remote indicating compasses, standards and correctors which are not already built into the compass shall be attached rigidly to the structure by means of their securing points so that in the normal flying attitude:

(i) the locating plane of the unit is horizontal or vertical, according to the type of unit concerned, and

(ii) the vertical plane through the fore and aft line of the unit is parallel to the vertical plane through the fore and aft axis of the rotorcraft.

3.3 No 3 and No 4 correctors for P type compasses shall be mounted on the compass bracket with their fixing lugs towards the compass. The vertical centre lines of each compass and its corrector shall be coincident. The correctors should not be separated from the compass by more than 3 mm.

4 DEVIATION CARDS

4.1 Holders suitable for standard compass deviation cards shall be provided at positions agreed for each type of Rotorcraft.

5 INTER-COMPASS SAFE DISTANCES

5.1 The distances between the magnetic elements of compasses/detector units shall not be less than those shown in Table 1.

6 PERMISSIBLE LIMITS

6.1 The design of the structure and position of the equipment and electrical wiring in the vicinity of the compass and detector unit shall be such that the figures given in Table 2 are not exceeded.

Note: The following are likely to jeopardise this requirement:

(i) ferro-magnetic bolts, nuts and washers in instrument coaming and windscreen arches, set at a distance from the magnetic element less than their safe distances,

(ii) single pole DC wiring or loops in the cockpit area or near the detector unit,

(iii) single pole DC wiring from engine driven generators or ground supply, especially if the ground supply point is positioned at a distance from and in a different horizontal plane from the engine driven generator,

(iv) unscreened AC cable in the vicinity of the detector unit,

(v) unscreened signal leads between detector and amplifier.

6.2 In the case of components subjected to shock, such as undercarriage units, the limits specified shall refer to the component in the shock magnetised condition (See para 9.1).

6.3 A vertical corrector shall be fitted if the change in deviation due to movement of the Rotorcraft from ground to normal cruising attitude is 2° or more.

7 DIRECTIVE FORCE

7.1 When it is anticipated that the directive force at the compass position is likely to be impaired due to the presence of soft iron, tests shall be carried out to determine the co-efficient λ (lambda). This is especially important when it is likely that compass swinging techniques will be used where the whole swing will be carried out with the rotorcraft on one heading. (Lambda is the ratio of the mean value of the horizontal force at the compass position, as deduced from readings with the rotorcraft on a number of headings, to the horizontal force which would be experienced in the Earth's magnetic field alone at the same point). If a low lambda is found, the acceptability or otherwise of the value should be established in conjunction with the Admiralty Compass Observatory.

8 DEMAGNETISATION

8.1 Components which have been subjected to strong magnetic fields, such as magnetic flaw detection tests or production operations involving the use of magnetic chucks, and which are liable to affect the compass shall, prior to assembly, be demagnetised (See Leaflet 715/3 for the demagnetising methods that can be used).

All drawings of such components shall be marked as follows:

"De-magnetise, Compass test at "x" metres".

Where "x" equals the distance from the compass the component will be when assembled in the Rotorcraft.

9 TESTED EQUIPMENT

9.1 All equipment shall be positioned outside its specified safe distance. The safe distance shall be determined in accordance with BS 3G.100: Part 2: Section 2.

9.2 If it is not possible to comply with the specific safe distances, the Rotorcraft Project Director shall be consulted.

10 COMPASS CALIBRATION

10.1 Provision shall be made for the accurate alignment of the Rotorcraft to:

(i) one tenth of a degree on Rotorcraft fitted with "Doppler" navigation equipment, and

(ii) one quarter of a degree on all other Rotorcraft.

10.2 Where possible, sighting rods shall be so arranged that sighting may be made from either ahead or astern of the Rotorcraft. It shall be possible to determine the verticality of the rods (in the athwartships plane) and also possible to lock the rods in the true vertical position. The fixed portion of the rods (ie, the part fixed to the Rotorcraft) shall be parallel to the Rotorcraft vertical datum in the athwartships plane. The points of suspension (eg, the line through the joints or pivots) of the movable parts of the rods shall be parallel to the horizontal flight datum (ie, that datum which is horizontal when the Rotorcraft is in a normal flying attitude). The rods shall be so designed that the one farther from the sighting device can be readily aligned to and is visible below the nearer rod when viewed from a height of 1.5 metres at a distance of 25 metres from the nearer rod. (Rods of triangular cross-section, the further one pointed and the nearer one Vee-notched, will achieve this).

10.3 The use of protruding equipment provided for other purposes (eg, wireless mast) or the use of paint marks on the structure is not acceptable for sighting purposes.

10.4 Where removable sighting rods are used, they shall be in a form suitable for transportation by the Rotorcraft and shall be considered as ground servicing equipment.

11 COMPASS ACCEPTANCE TESTS

11.1 A complete magnetic and electrical investigation of the compass installation on the general lines given in Leaflet 713/4 shall be made, in consultation with the Admiralty Compass Observatory, on the first prototype and the first fully equipped production rotorcraft.

11.2 The Rotorcraft supplied for navigation assessment purposes shall be checked on 2 occasions, separated by an interval of at least 20 hours flying time, to ensure that the compass deviations remain within the limits in Table 2.

TABLE 1

MINIMUM DISTANCES BETWEEN COMPASSES

Compass Type	**Minimum Distances between the centres of the instruments in metres (inches)**			
	P11 with No 3 or 4 Corrector	**P12 with No 3 or 4 Corrector**	**E2**	**Remote Indicating Detector Units**
P11 with No 3 or No 4 Corrector …	0.51 (20)			
P12 with No 3 or No 4 Corrector …	0.51 (20)	0.51 (20)	0.31 (12)	
E2 … … … …	0.51 (20)	0.51 (20)		
Remote Indicating Detector for Units …	1.02 (40)	1.02 (40)	0.61 (24)	0.41 (16)

TABLE 2

PERMISSIBLE LIMITS OF COEFFICIENTS AND CHANGE IN DEVIATION OF COMPASSES

		Main Compass		
		Remote Indicating	**Direct Indicating**	**Stand-by**
(i)	Permanent magnetism of structure and equipments as given by uncorrected coefficients B and C (in a field of 18 microtesla 'h' and 43 microtesla 'z')	2°	10°	20°
(ii)	Induced magnetism as given by coefficients D and E	0.25°	1°	2°
(iii)	Operation of electrical equipment. Maximum deviation from:-			
	(a) Individual circuits	) 0.25°	(1°	2°
	(b) Any combination of circuits	)	(3°	5°
(iv)	Changes due to modification of the rotorcraft or equipment provided conditions (i) and (ii) are not exceeded:-			
	(a) Individual	) 0.25°	(1°	1°
	(b) Any combination of mods.	)	(3°	3°
(v)	Moving components when moving between limiting conditions. Change of deviation (see para 6.2)	0.25°	1°	2°
(vi)	Changes in magnetism of fixed structure subjected to intense shock. Change of deviation.	0.25°	1°	2°
(vii)	Change in deviation between ground and normal cruising position (see para 6.3)	0°	5°	5°
(viii)	Changes due to changing from engine running to engine off states. Change of deviation	0°	1°	2°
(ix)	Changes due to changing from external ground electrical supply to 'in flight' supply. Change of deviation	0°	1°	2°
(x)	Changes due to installation of role equipment eg ECM/RECCE pods (See Note 1)			
	(a) Individual items	) 0.25°	(1°	1°
	(b) Any combination of items	)	(3°	3°
(xi)	Changes due to fitting Armament and/or	) 0.25°	(1°	2°
	Guided Weapons (See Note 1)	)	(3°	5°
(xii)	Total Changes (sub-para. (iii) to (xi))	0.5°	5°	8°

Notes: 1 Equipments that are normally non-disposable shall meet the requirements of (x) and those that are normally disposable shall meet the requirements of (xi).

2 Any secondary compass which will be used for the completion of a mission in the event of the failure of the primary compass shall be subject to the 'Main Compass' requirements.

LEAFLET 713/0

MAGNETIC COMPASS INSTALLATIONS

REFERENCE PAGE

A & AEE Reports

Res/175/1	..	..	Lightning: Its effect upon compass deviation in aircraft
Res/175/3	..	..	Changes during flight of deviation due to fuselages, etc

RAE Technical Memoranda

LAP363	..	..	The magnetising effects of field strength variation on Gyro Magnetic Compass Mk 4 detector units

Air Publications

AP 3456D, Part 2	..	..	Heading and Alignment Instruments
AP 112B.	..	..	This series describes specific compass types

British Standards

2G 100	..	..	General Requirements for Aircraft Electrical Equipment and Indicating Instruments
3G 100	..	..	General Requirements for Equipment in Aircraft
			(2G.100 is progressively being superseded by 3G 100)

LEAFLET 713/1

MAGNETIC COMPASS INSTALLATIONS

EFFECT OF MAGNETIC STRUCTURE

1 INTRODUCTION

1.1 This Leaflet gives information on the effect of magnetic structure on magnetic compasses and makes recommendations on the positioning of such structures.

2 GENERAL

2.1 Magnetic poles will generally occur at discontinuities and joins in the structure, and the interfering field on the compass is due to the cumulative effect of those free poles. The magnitude of a free pole is dependent on the cross sectional area of the member at which the pole occurs, while the effect of such a pole on the compass varies as the inverse square of the distance from the compass. An assessment of the effect of components of the rotorcraft structure which have become magnetised through shock can be obtained by means of the following relationship:

$$F = 4\,\frac{\ell^{\frac{3}{2}}\,A^{\frac{1}{4}}}{d^2}$$

Where F	….	microtesla	….	field
ℓ	….	metres	….	length of the member
A	….	square metres cross sectional area		
d	….	metres	….	distance from the pole to the compass

It should be noted that this is an empirical equation which takes into account the demagnetising factor, the earth's field, and the mean value of effective permeability when subjected to shock. The equation is only valid for values of:

$\frac{\ell^2}{A}$ less than 6,500.

2.2 Assuming that the maximum permissible interference field F, for any one pole is 1 microtesla for the direct indicating compass and 0.1 microtesla for the remote indicating compass, then the minimum distance of the pole from the compass is given by:

(i) direct indicating compass, $d = 2\ell^{\frac{3}{4}}\,A^{\frac{1}{8}}$ and

(ii) remote indicating compass, $d = 6.3\ell^{\frac{3}{4}}\,A^{\frac{1}{8}}$

2.3 Magnetic materials should therefore not be used adjacent to the compass position. For remote indicating compasses, the minimum distances of these materials from the detector unit are laid down in Chapter 713. For direct indicating compasses, these materials should preferably not be used closer than 3/4 metre (2½ ft) to the compass. If it is not possible to comply with the above limitations the Contractor should discuss the position with the Rotorcraft Project Director at an early stage in the design.

3 FASTENERS

3.1 Where fasteners have to be used in the vicinity of the compass, an assessment of the effect of magnetic material can be obtained using the equation given above. Since the length of the fastener is generally small, much smaller distances d will be permissible. Where a number 'n' are being used, the total field can be assumed to be nF and hence the mean distance d will be given by:

(i) direct indicating compass, $d = 2n^{\frac{1}{2}} \ell^{\frac{3}{4}} A^{\frac{1}{8}}$ and

(ii) remote indicating compass $d = 6.3n^{\frac{1}{2}} \ell^{\frac{3}{4}} A^{\frac{1}{8}}$

4 PARTS SUBJECTED TO SEVERE SHOCK

4.1 In general the magnetisation of structure will be random and should be reasonably stable. Parts, however, which are subjected to severe shock, such as undercarriage units, gun blast tubes, etc., will become magnetised to an extent depending on the orientation of the item in the earth's magnetic field at the moment it experienced the shocks.

LEAFLET 713/2

MAGNETIC COMPASS INSTALLATIONS

EFFECT OF WIRING

1 INTRODUCTION

1.1 This Leaflet gives the precautions which should be taken when ground return systems are used in rotorcraft, to ensure that stray magnetic fields set up by current loops do not deviate the compass.

1.2 It is difficult to prescribe any set method of treatment, and each problem has to be treated very much on its merits. The following broad classification can, however, be made, but further information and advice on wiring layout may be obtained from the RAFLO, Admiralty Compass Observatory, Slough.

2 RE-ROUTEING OF WIRING FURTHER AWAY FROM THE COMPASS

2.1 Re-routeing of wiring, further away from the Compass can be applied particularly to light current leads installed close to the compass.

3 ALTERATION OF THE GROUNDING POINT

3.1 Often the deviation is due to the supply lead to equipment which is grounded locally. The interference can be eliminated by running a grounded lead from the equipment adjacent to the supply lead to a fresh grounding point some distance away. In this way, the part of the circuit adjacent to the compass can be made double pole. Where this kind of modification is used, the question of effective radio noise suppression should be considered and it may be necessary to install an additional suppressor in the grounding lead. In all cases advice should be obtained from the appropriate authority to prevent the cause of EMC problems.

4 MOVING CABLES INTO THE SAME HORIZONTAL PLANE AS THE COMPASS

4.1 If the lead producing the interference is moved to the same horizontal plane as the compass, the magnetic field produced by the lead at the compass will be vertical and will not cause deviations of the compass except when the rotorcraft is pitched or rolled. This method of correction is acceptable providing a pitch or roll of 10° does not result in more than 1° deviation of the compass.

5 INTRODUCTION OF CABLE LOOPS

5.1 This method involves the introduction of a loop of cable carrying the main current of the circuit to be corrected, this loop being arranged to produce a component of field which just cancels the component of field due to the lead. It is a method which is rarely used and should only be employed in an emergency when other methods have failed.

LEAFLET 713/3

MAGNETIC COMPASS INSTALLATIONS

DEMAGNETISATION

1 INTRODUCTION

1.1 This Leaflet gives the methods which can be used when demagnetisation of a part is necessary to comply with the requirements of Chapter 713. The method used will depend on the structure or components to be demagnetised. Further information may be obtained from the RAFLO, Admiralty Compass Observatory, Slough.

2 GENERAL

2.1 In general, the AC coil method is satisfactory for the routine demagnetisation of components before assembly, except where the component has a larger cross-sectional area, eg undercarriage units. In these cases, the formation of eddy currents in the metal reduces the effective intensity of the demagnetisation field. To overcome this a DC coil method is used. If it should be necessary to demagnetise fixed members of the rotorcraft structure after assembly then an AC wiper magnet may be used. This method may also be used for components which by reason of their size or shape cannot readily be passed through a demagnetising, coil. It is not suitable for components having a large cross-sectional area.

2.2 In all cases care should be taken in selection of the site for demagnetisation since the presence of masses of iron may adversely affect the operation.

3 AC COIL METHOD

3.1 With this method the component to be demagnetised is passed through a coil supplied with alternating current. The coil is mounted with its axis parallel to the E-W line so that the component is not subject to the earth's field. The field intensity within the coil necessary for satisfactory demagnetisation depends on the nature of the steel being demagnetised, but will be proportional to the coercive force of the steel. In most cases a field intensity of approximately 1000 amperes/metre will be found sufficient although it may occasionally be necessary to use a field intensity as high as 2000 amperes/metre.

4 DC COIL METHOD

4.1 The component to be demagnetised is placed within a coil large enough to enclose it completely, the axis of the coil being E-W. The direct current through the coil is reversed at approximately 2 second intervals and gradually reduced from a high value giving a field intensity of at least 800 amperes/metre to zero.

5 WIPER MAGNET METHOD

5.1 This method makes use of an AC magnet having U-shaped iron core. The member of the structure to be demagnetised is placed as nearly as possible at right angles to the earth's field. The magnet is then placed in contact with and moved slowly along the member. This operation is then repeated three or four times and before switching off the current the magnet should be removed away from the member.

Note: The DC coil technique, the AC wiper magnet technique and equipments are also described in CSDE Report 232/75. (Central Servicing Development Establishment, RAF Swanton Morley).

LEAFLET 713/4

MAGNETIC COMPASS INSTALLATIONS

ACCEPTANCE OF TYPE TESTS

1 INTRODUCTION

1.1 This Leaflet gives information on the Admiralty Compass Observatory (ACO) acceptance of type tests called for in Chapter 713 para 11.

1.2 The tests given in this Leaflet should be considered as a guide and the actual tests, which will depend on the various equipments fitted in the rotorcraft, should be agreed with the Admiralty Compass Observatory. (The ACO should be consulted at an early stage when it is proposed to design a compass system into a new rotorcraft, or into an existing rotorcraft where extensive modifications are required).

1.3 The definitions and methods of determining and correcting the coefficients referred to in this Leaflet and in Chapter 713 are given in AP 3456D, Part 1, Section 6, Chapter 2.

2 OBJECT OF TESTS

2.1 The object of the tests is to check:

(i) the functioning of the compass, including its power supply/supplies, and

(ii) the effects of the rotorcraft's magnetism and electrical equipment and wiring.

The compass installation should also be examined for ease of viewing, manipulation and servicing.

3 FUNCTIONING OF COMPASS, INCLUDING ITS POWER SUPPLY/SUPPLIES

3.1 Type testing of the proposed compass systems should include performance tests at the ACO.

4 EFFECT OF AIRFRAME

4.1 The magnetic sensors of the compass systems in the rotorcraft should be removed and replaced by ACO calibrated Survey compasses. A 16 point swing will then be carried out to determine the coefficients B,C,D and E at the magnetic sensor positions. These coefficients should not exceed the values given in Chapter 713, Table 2 (i) and (ii).

5 EFFECT OF EQUIPMENT AND COMPONENTS

5.1 Tests should be made on 4 headings approximately 45° apart to determine the effects of the following (the tests should be carried out using the rotorcraft compasses, if suitable, otherwise ACO survey compasses will be used):

(i) Switching on and off all electrical and radio equipment (ignoring surges) with power supplied:

(a) external, and

(b) from the internal power supplies in all modes of operation, including engine running.

(ii) (a) Detachable equipment and components, and

(b) movable equipment and components (controls, flaps, hood, seat, drop tanks, control locks etc).

(iii) Undercarriage.

5.2 When checking the effect of raising the undercarriage, if steel jacks are required, care should be taken to ensure that their effect on the detector unit is taken into account. It is sometimes possible to do this test in a hanger if the magnetic field inside the hanger is uniform and not greatly different from that outside.

5.3 The changes of deviation caused as a result of the above should not exceed the values given in Chapter 713 Table 2, sub paras (iii) to (xii).

6 DETERMINATION OF TILT ERROR

6.1 In order to enable a rotorcraft with a tail wheel type undercarriage to be swung in the tail down position, the tilt error should be determined by observing the change of compass heading between tail down and tail up attitudes on both East and West heading. The tilt error is the mean of the two values of change observed (regardless of algebraic sign).

6.2 If a crane is used for lifting the rotorcraft to the tail up attitude, the crane may be left in position during the test provided it is not electrically operated and provided it is not close to the sensitive elements of the compasses.

7 TEST FACILITIES

7.1 Depending on the type of rotorcraft and the nature of the tests required, some or all of the following will be necessary:

(i) Compass base cleared of other rotorcraft, steel trestles and ladders, etc.

(ii) Adequate stable ground power supply (the type with a charger incorporated and fitted with a voltmeter is preferable).

(iii) Rotorcraft fully equipped, and serviceable.

(iv) The necessary ground handling personnel and equipment, including non-magnetic ladders for access to detector units, etc.

(v) Any test and alignment equipment that is normally required for calibrating the compasses of the rotorcraft e.g., PHTS, sighting rods and test sets etc.

CHAPTER 714

TRANSPORT ROTORCRAFT - ROLE EQUIPMENT INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this chapter cover the general design and strength aspects of rotorcraft arising from the following roles:

(i) Passenger communications.

(ii) Carriage of internal freight, including wheeled or tracked vehicles.

(ii) Carriage of external freight.

(iv) Troop transport.

(v) Paratroop dropping.

(vi) Aeromedical evacuation.

(vii) Search and rescue.

The Rotorcraft Specification will state which roles apply to the particular type.

2 BASIC OPERATIONAL REQUIREMENTS

2.1 GENERAL

2.1.1 The equipment to be used for the various roles will be stated in the Rotorcraft Specification. (It will generally be that already in Service use and details can be obtained from the Rotorcraft Project Director). The change from one role to another shall be able to be made using the minimum of special equipment in the shortest possible time, and the fixed fittings for one role shall not interfere with the proper performance of another.

2.1.2 Doors shall comply with the requirements of Chapter 109 and wherever possible shall also be designed to act as emergency exits. For doors used as exits for jumping by paratroops, see para 2.6.4.

2.1.3 The floor and at least six inches up the adjacent fuselage walls shall be permanently sealed as far as possible to protect the rotorcraft structure against contamination by the seepage of spilled liquids. All rotorcraft and engine controls, and equipment in underfloor compartments shall be adequately protected from jamming and corrosion arising from the carriage of any loose stores. Any interior finish used on the fuselage walls and floor shall be tough, durable, flame-resistant and easily cleaned.

2.1.4 Inspection or access panels and hatches which it may be necessary to use in flight, and emergency exits, shall be so arranged that it is impossible for them to be obstructed by seats or other role equipment.

2.1.5 Adequate normal and emergency lighting shall be provided in passenger and freight compartments. In addition, illumination shall be provided near entrances and hatches to enable loading to be carried out at night. Lights shall be protected against accidental damage. Windows shall be provided with sunblinds and means for blacking out in accordance with Chapter 105, para 16.1, and with grilles for protection against freight where necessary. These arrangements shall not prejudice the use of such windows as emergency exists where they are so designated.

2.1.6 Suitable signs, which can be illuminated when necessary, shall be fitted as required for the various roles, e.g., "FASTEN SEAT BELTS" (see also paras 2.2.2 (iv) and 2.6.7). When fitted, such signs shall be operable by both the pilot and the cabin crew member, and at each station an indication of the signs from either shall be provided.

2.1.7 Provision shall be made for the stowage of hand-operated fire extinguishers in accordance with DEF STAN 42-13 to a scale agreed with the Rotorcraft Project Director.

2.2 PASSENGER COMMUNICATIONS ROLE

2.2.1 Toilet compartments shall be provided on the scale laid down in the Rotorcraft Specification and shall contain chemical closets, urinals and sick basins. Handholds shall be provided for use in adverse flying conditions. The lighting of the compartment shall be adequate with switches of easy access. Urinals shall be of ample dimensions and shall drain into a collector tank and not discharge to atmosphere. The sick basin shall be of the flushing type and discharge into the urinal collector tank. Unless specifically agreed otherwise, both chemical closets and urine tanks shall be capable of being serviced without entering the rotorcraft.

2.2.2 Where smoking is permitted on the flight deck and/or passenger compartments:

(i) seat upholstery and other furnishings shall be of flame- resisting materials,

(ii) non-fuming ashtrays shall be fitted in armrests or as required,

(iii) fuel and oil lines shall be routed outside crew and/or passenger compartments, or where this is impossible, shall be enclosed in a duct and vented to atmosphere, and

(iv) signs, which can be illuminated when required, shall be fitted in the cabin, to indicate "NO SMOKING".

2.3 INTERNAL FREIGHT ROLE

2.3.1 The freight floor shall be divided into a grid 508 mm (20 in) between the grid lines, one grid line coinciding with the centre line of the rotorcraft. At each intersection of grid lines, a cargo tie-down point (see para 3.2.1) shall be installed. In regions of reduced floor width, deviations may be permitted, to utilize to the best advantage the available floor space, subject to the agreement of the Rotorcraft

Project Director. Additional tie-down points in other positions on the floor may also be required.[1]

2.3.2 Freight lashing points shall be provided at heights and intervals on the fuselage side walls as stated in the Rotorcraft Specification.

2.3.3 The designated cargo compartment shall be free from bulkheads throughout its entire length. The sides of the compartment shall have no protrusions which will interfere with the loading and lashing of freight and equipment. Compartment limits shall be clearly marked by lines on the floor, bulkhead and fuselage walls.[2] Rows of tie-down points shall be identified by marking from port to starboard in letters and from front to rear in numbers. Stowage for lashing equipment, to a scale agreed with the Rotorcraft Project Director, shall be provided.

2.3.4 With the rotorcraft stationary on the ground, the freight floor shall be substantially level. It shall be possible to drive or winch in the maximum single load which is likely to be carried without upsetting the balance of the rotorcraft on its undercarriage. If necessary, provision shall be made to support the fuselage during this operation. The sills of all doors giving access to the freight compartment shall be flush with the floor and their height shall be as stated in the Rotorcraft Specification.

2.3.5 Suitable arrangements shall be provided for the fitment of a winch or hoist for the loading of heavy cargo.

2.4 EXTERNAL FREIGHT (CRANE) ROLE

2.4.1 Provision shall be made for the carriage of external loads up to the maximum load stated in the Rotorcraft Specification, by means of a controllable-release hook.

2.4.2 It shall be possible for the crew member to observe the hook and load safely from the freight compartment. Intercommunication shall be provided between the crew member and the pilot(s).

2.4.3 The normal hook release shall be operable by both the pilot and the crew member. An emergency mechanical release shall be provided, operable from the pilot's station.

2.4.4 The carriage of external freight shall not produce unacceptable handling characteristics in the rotorcraft.

2.5 TROOP TRANSPORT ROLE

2.5.1 On large rotorcraft where such an arrangement is economic of space, backward-facing seats, harnesses and anchorages shall be provided to the requirements of Chapter 307. They shall either pick up the cargo tie-down fittings (see para 2.3.1) or special seat rails shall be provided. Seat rails shall be at constant track throughout the fuselage. Gangways shall have a minimum floor width of 381 mm (15 in) and shall be not less than 508 mm (20 in) wide above elbow level to allow freedom of movement.

2.5.2 Where this arrangement is impracticable, sideways facing seats to the requirements of para 2.6.3 and/or forward facing seats to the requirements of Chapter 307 are acceptable.

2.5.3 Adequate provision shall be made for the safe stowage of troops' personal kit and equipment against the accelerations of para 3.1.1.

2.6 PARATROOPING ROLE

2.6.1 The rotorcraft shall be capable of straight and level flight with the paratroop exit doors open, up to an airspeed to be agreed with the Rotorcraft Project Director after consideration of both the paratrooping and search and rescue roles.

2.6.2 An aimer's station for the control of paratroop dropping, which may be the pilot's or co-pilot's station shall be provided in the nose of the rotorcraft. It shall possess adequate forward and downward view through an optically flat transparent panel, or its equivalent, and shall be equipped with suitable release, intercommunication and signalling lights systems (see also para 2.6.7). The instrument installation shall include an altimeter, gyro compass repeater and airspeed indicator.

2.6.3 Sideways facing seats which can be folded against the fuselage walls shall be provided. They shall be retained in the stowed position by a simple catch which can be operated by a gloved hand. It is desirable that stretchers should be capable of being fitted without removal of sideways-facing seats. Seats shall be equipped with safety nets or lap straps to restrain the body against acceleration in the event of a crash landing. Nets, straps and seats shall meet the requirements of Chapter 307. Means shall be provided for lashing his kit as close to him as possible. In order that extra space in large rotorcraft may be utilized to the full, it is acceptable to provide centre-line outward-facing seats, in addition to those mounted on the fuselage walls, provided that such an arrangement allows sufficient movement within the rotorcraft and permits access to the exit door(s).

2.6.4 Doors used for jumping shall be not less that 914 mm (36 in) wide with corners rounded to approximately 100 mm (4 in) radius. The sides of the exits shall be designed to avoid injury to paratroops or damage to equipment or strops. The floor covering near the door shall be of non-slip material and shall extend up the cabin far enough to enable the paratroops to turn to the door without slipping; the distance shall be agreed with the Rotorcraft Project Director. The sills shall be flush with the floor and free from steps or obstructions.

2.6.5 A position for a Despatcher shall be provided adjacent to each exit door such that he does not obstruct the paratroops' access to it. His safety harness shall be capable of attachment to the anchor system referred to in para 2.6.6 should he be required to move up the cabin, and to an alternative strong point, which shall be provided near the exit door, for use when actually engaged in despatching.

2.6.6 Anchor cables or an equivalent system for the deployment of parachutes shall be provided. Means shall be provided to haul in strops and parachute packs after the drop, and for strop stowage.

2.6.7 Provision shall be made for intercommunication between the Despatcher and the pilot, and also for a Drill Indicator Board positioned so as to be within reach of the Despatcher at his station and clearly visible to all the paratroops. Two
AL/4 signal lights, one green and the other red, shall be installed above or near each exit door so as to be clearly visible to the Despatcher and to the No. 1 of each stick when he is ready to jump. These lights shall be operable from the pilot's station and also from that of the aimer where such separate station is provided and at these stations an indication of operation of the signal lights from either shall be provided.

2. 7 AEROMEDICAL EVACUATION ROLE

2.7.1 Special role equipment for the installation of standard stretchers shall be provided. The associated fittings shall be such that when the rotorcraft is in its normal cruising attitude the stretchers are substantially horizontal. Where space permits, the stretchers shall be carried in tiers. The installation shall meet the crash landing requirements of Chapter 307. Stowage for the carriage of the stretcher support gear and of the associated restraining harnesses shall be provided.

2.7.2 It shall be possible to remove the seats referred to in para 2.5.1 and fit stretchers quickly and easily.

2.7.3 Where, for example, on small rotorcraft or on those of unusual design, it is impracticable to meet the requirements of para 2.7.1, suitable alternative arrangements shall be agreed with the Rotorcraft Project Director. These may include athwartships carriage of stretchers, non-standard stretchers and external carriage of stretchers.

2.7.4 Where stretchers are carried externally, provision shall be made for the protection of the occupant from his environment, i. e. temperature, rain, hail or snow, airflow (including rotor downwash), etc.

2.7.5 Arrangements for external carriage of stretchers shall not affect adversely the handling qualities of the rotorcraft.

2. 8 SEARCH AND RESCUE ROLE

2.8.1 Provision shall be made for the installation of a Search and Rescue (SAR) winch of an approved type. Means of controlling its operation shall be provided adjacent to the winch itself and also at the pilot's station.

2.8.2 Adequate stowage shall be provided for the standard item of SAR equipment (e.g. resuscitator, signalling lamp). Power supplies for the operation of any such equipment shall be provided as required.

2.8.3 Provision shall be made for the fitment of protective material (additional to that mentioned in para 2.1.3), to the floor and sides of the fuselage, to prevent contamination by sea water draining from the clothing of rescued persons or crew members.

2.8.4 Suitable handgrips, one above and one to the right of the entrance door, shall be provided.

2.9 WINCHES - SAFETY REQUIREMENTS

2.9.1 For loads up to the maximum dead weight for which the winch is designed, no single failure shall allow the load to run out of control. This requirement can be met by use of an independent brake applied automatically or by hand. Automatic operation is preferred in order to minimise the time delay.

2.9.2 The efficiency of the brake shall be demonstrated by simulating failure under maximum load conditions while lofting and lowering the load. For hand operated brakes, a delay time, to be agreed with the Rotorcraft Project Director, shall be assumed between the simulated failure and the application of the brake.

2.9.3 No single failure shall permit the cable to run past the limits of its design travel.

3 STRENGTH REQUIREMENTS

3.1 FUSELAGE

3.1.1 With the total payload restrained by any approved lashing scheme, the structure of the fuselage to which cargo tie-down fittings are attached, including any special equipment or installation required in the freight role, shall achieve:

(i) the proof and ultimate factors required for the rotorcraft as a whole in flight, take-off and landing, and

(ii) an ultimate factor of not less than 1.0 on the loads produced, during an emergency controlled landing, by the following rotorcraft accelerations:[4]

Upwards	4g
Backwards	4g
Forwards	2g
Sideways	1.5g
Downwards	2g

Notes: 1 The above factors apply only to non-occupied cargo areas or when personnel are protected in the direction of acceleration by rotorcraft structure or barriers. They should be assumed to act uni-directionally (i.e., not in combination).

2 Where cargo and passengers are mixed, without the protection quoted in Note 1 above, additional protection (e.g., barriers or nets), may be necessary to meet the crash conditions of Chapter 307. In this case, advice shall be obtained from the Rotorcraft Project Director.

3.2 CARGO TIE-DOWN FITTINGS

3.2.1 Cargo tie-down fittings for floor attachments shall be in accordance with the requirements of DEF STAN 00-3 and floor apertures with those of DEF STAN 16-23. Attachment fittings on fuselage side walls shall have strengths as specified in any direction inwards from the wall.

3.3 FREIGHT FLOORS

3.3.1 The unit loading kg/m^2(lb/ft^2), which the floor is required to carry, will be given in the Rotorcraft Specification together with axle loads and wheel spacings. Special combinations of freight or vehicles that shall be catered for will be stated by the Rotorcraft Project Director.

3.3.2 The floors of freight compartments, the rotorcraft structure as a whole, and any ramp or staging which forms part of the rotorcraft equipment (as opposed to ground equipment) shall have an ultimate factor not less than 3.0 under the combined loads arising from any unit of freight in any possible position during, loading and unloading, including winching loads, together with the loads due to the rotorcraft standing on the ground.

3.3.3 The floor shall be designed to distribute concentrated local loads (e.g., those arising from the boots of troops or loading crews), so that local failures will not occur. This requirement may be met by means of an exchangeable covering of rubber or other suitable non-inflammable material, the design of which shall permit the fitting of all appropriate role equipment. Protection against damage by the tracks of tracked vehicles need not be provided unless specified.

3.4 ANCHOR CABLE INSTALLATIONS FOR PARATROOPING

3.4.1 The anchor cable installation or equivalent system shall have a proof factor of not less than 1.25 and an ultimate factor not less than 1.5 under the load arising from consideration of the two following design cases whichever is the less:

<u>Case 1 Hang-up</u>

The drag of (N-2) static lines and bags at the maximum drop speed plus the impact or acceleration load, whichever is the greater, caused by the snagged (N-1)st paratrooper plus the maximum load of the normal deployment of the Nth paratrooper.

<u>Case 2 Failure</u>

The pull of the strongest strop and static line up to the maximum breaking load of its weakest element provided that this has been reliably established by ultimate strength tests plus the drag of N-2 static lines and bags at the maximum drop speed plus the maximum load of normal deployment of one paratrooper.

Note: N is the number of paratroops carried on one cable.

3.5 EXTERNAL FREIGHT (see Chapter 205 and Leaflets)

3.6 CARGO AND PERSONNEL LIFTING EQUIPMENT

3.6.1 Role equipment for lifting or winching cargo or personnel (including the SAR hoist), its installation and attachment to the rotorcraft, shall have proof and ultimate factors not less than 3.0 and 4.0 respectively on:

(i) the maximum dead-weight load which can be lifted, or

(ii) the maximum steady tensile load which can be produced in the cable by winching, whichever is appropriate.

3.7 STRETCHERS

3.7.1 Stretchers, their harnesses, attachments and back-up structure, shall have the same standard of strength as the rotorcraft as a whole. In addition the requirements of para 3.1 above shall be met.

3.7.2 Consideration shall be given to meeting the requirements of Chapter 307 as far as possible.

3.8 MOBILE CREW

3.8.1 Suitable attachment points for crew required to stand near an open door shall be provided. The number and location shall be agreed with the Rotorcraft Project Director.

3.8.2 Each attachment point shall have an ultimate factor of not less than 1.0 on a force of 11000N (2500 lbs) acting in any direction within a cone of 60^{o} included angle having its apex at the attachment point and its axis parallel to the vertical datum of the rotorcraft.

3.9 TROOP SEATS

3.9.1 Troop seats, their harnesses, attachments and back-up structure shall meet the relevant requirements of Chapter 307. In particular, static strength, static tests, dynamic analyses, and dynamic tests shall show compliance with Chapter 307 para 1.7.

4 TEST EQUIPMENT

4.1 A test shall be made, as agreed by, and to the satisfaction of the Rotorcraft Project Director, to demonstrate the ease and rapidity of change of role, using the appropriate equipment.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	44/17	-
2	-	3462
AL/4 3	-	-
4	44/32	3542

CHAPTER 715

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

1 INTRODUCTION

1.1 This chapter states the requirements for the design, installation and testing of optical transparencies.

1.2 The optical transparent components shall be designed to provide clear vision for both the pilot and crew, and to protect the cabin, passengers and crew as defined both in this chapter and the Rotorcraft Specification.

1.3 The chapter also states the requirements for satisfactory vision through optical transparencies in the course of normal operations. Additional special requirements may be necessary for transparencies associated with particular functions, for example camera windows for survey or reconnaissance.

1.4 When preparing the Rotorcraft Specification consideration must be given to certain special requirements which may need to be defined. The paragraphs within this chapter which need to be considered are:

Chapter 715 paras 2.1.1, 2.1.2(x), 2.3.6, 2.3.7, 2.3.9, 2.3.10, 2.3.11, 2.3.12, 2.3.13, 3.2.2, 3.2.8 and 7.1.
Chapter 715 Annex A paras 1.4 and 1.6.
Leaflet 715/2 paras 5.1 and 5.3.2.
Leaflet 715/4 para 2.3.

2 DESIGN REQUIREMENTS

2.1 GENERAL

2.1.1 Transparencies shall be designed to meet all possible combinations of steady and transient conditions that can occur within the specified flight envelope of the Rotorcraft or on the ground. Steady conditions are defined in those flight or ground conditions enduring for five minutes or more. All others are transient.

Note: Transient conditions can arise from speed and other effects or from combinations of them. Where speed effects are involved, flight at speeds above the maximum in level flight may be regarded as transient.

2.1.2 The following features shall also be considered in design and represented during approval tests:

(i) outside air temperatures in accordance with the requirements of Chapter 101 and the Rotorcraft Specification,

(ii) solar radiation,

(iii) cabin heating and cooling,

(iv) de-icing and demisting systems,

(v) runaway of windscreen heating system due to thermostat failure,

(vi) rain dispersal systems including windscreen wipers,

(vii) edge effects from mounting,

(viii) altitude effects,

(ix) rain, hail, sand and dust erosion,

(x) those special requirements detailed in Annex A para 1.2 which are called up in the Rotorcraft Specification.

2.1.3 To cater for ditching and crash landing, all transparencies and their supporting structure shall meet the requirements of Chapter 307 as far as they are relevant.

2.1.4 Complete dimensional tolerances shall be specified for the finished transparency. These shall be stated on the relevant drawings, specifying where necessary the appropriate temperature and humidity conditions at which compliance should be verified. Particular attention shall be paid to transparencies which are vulnerable to thinning or distortion due to the manufacturing process and to basic material tolerance which may accentuate thinning or cause lack of fit.

2.1.5 Where a specific angle of vision and/or areas with critical optical requirements are demanded, the relevant details and pilot's eye position shall be specified on the component drawings.

2.1.6 Consideration shall be given to the effect of storage conditions including plastic ageing and humidity.

2.2 MATERIALS AND CONSTRUCTION

2.2.1 Special consideration shall be given in the transparency design and choice of materials to minimise the breakage of any panel resulting in fragmented particles, which shall neither injure any occupant of the Rotorcraft nor create any hazard.

2.2.2 A complete process control document shall be prepared for the fabrication and installation of each complete transparency into the Rotorcraft.

2.3 DESIGN FEATURES

2.3.1 In the design of transparencies and their mountings, consideration shall be given to the following points:

(i) the choice of interlayer with respect to temperatures likely to be experienced in service and the effects of altitude,

(ii) the chemical compatibility of sealants, interlayers and organic materials when laminated to each other,

(iii) sealing of the assembly against possible attack by water, chemicals or other harmful substances such as sulphur dioxide and ozone, eg cleaning agents and adhesives,

(iv) the effect of ultra violet exposure,

(v) the ability of the transparency to resist crazing,

(vi) the ability of the transparency to resist scratches and abrasion,

(vii) all dimensional changes in the transparency and all deflections of the structure in normal and extreme operating conditions to ensure that they can be accommodated without giving rise to stresses in excess of those allowed for in the design,

(viii) possible damage from pressure pulses caused by firing of adjacent armaments, See Leaflet 715/1 para 3.5,

(ix) outside air temperature as detailed in para 2.1.2 together with all the other points detailed in para 2.1.2. Due consideration shall be given to:

a) material tolerances,

b) manufacturing tolerances,

c) deflections of the structure in flight.

(x) the provision of expansion gaps to accommodate dimensional changes which will not be rendered inoperative by sealing compounds or other fillers. (Statements to be included on the relevant drawings). Due consideration shall be given to:

a) material tolerances,

b) manufacturing tolerances,

c) deflections of the structure in flight.

(xi) the avoidance of steps at transparency edges or mountings that would be unacceptable from bird impact, rain, hail, ice accretion or aerodynamic considerations,

(xii) ease of replacement in service.

2.3.2 The transparency shall be designed to resist foreign object damage, such as stones thrown up by rotor action.

2.3.3 The transparency shall be designed to resist bird impact and shed ice in accordance with Chapter 206 and Chapter 711 respectively.

2.3.4 The transparency shall be designed to comply with the relevant parts of Chapter 105. When analysing the pilot clearance envelope the use of helmet-mounted night vision goggles shall be taken into account where the use of helmet mounted night vision goggles is a requirement of the Rotorcraft specification.

2.3.5 The transparency shall be designed to resist the effects of lightning strike in accordance with Chapter 708.

2.3.6 The transparency shall be designed to minimise the radar cross-section if required by the Rotorcraft Specification.

2.3.7 The transparency shall be designed to minimise infra-red emissions if required by the Rotorcraft Specification.

2.3.8 The magnetic emissions of the transparency shall be of a level that will not affect nearby instrumentation in accordance with DEF STAN 59-41.

2.3.9 Reduction of canopy glint in the visible and infra-red bands may be a requirement in the Rotorcraft Specification. In such a case the Rotorcraft Design Authority shall incorporate the requirement for reduction of canopy glint in the design and test programme for the rotorcraft. See also Leaflet 715/4.

2.3.10 The Rotorcraft Design Authority shall prepare a statement on the penalty for other design features of the Rotorcraft, if a reduction of canopy glint beyond that existing in the design proposed for delivery to the Services is considered necessary by the MoD. This statement shall also state the requirements for the in-service cleaning of the canopy.

2.3.11 The transparency shall defeat threats from lasers and particle beam weapons to the extent specified in the Rotorcraft Specification and in accordance with Chapter 717.

2.3.12 The transparency shall survive loads resulting from transient phenomena associated with nuclear detonation, in accordance with Chapter 717 and DEF STAN 00-35, if defined as a requirement of the Rotorcraft Specification.

2.3.13 The transparency shall resist the effects of nuclear, biological and chemical contaminations, and the detrimental decontamination solutions/cleaners, in accordance with Chapter 717 and DEF STAN 00-35, if defined as a requirement of the Rotorcraft Specification.

2.4 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

2.4.1 Means shall be provided, as appropriate, to reduce the effects of Specified and Defined Threats (see Chapter 112) on optical transparencies and canopies.

3 REQUIREMENTS FOR SATISFACTORY VISION

3.1 CATEGORIES FOR CLEAR VISION

3.1.1 The function of any transparency shall be clearly defined and area or areas thereof shall be identified according to the required quality of vision. The categories of vision areas are as follows:

Category I Areas of forward facing windscreens of the highest optical quality suitable, for example, for weapon aiming.

Category II Areas of panels used for critical reconnaissance and search purposes.

Category III Main vision areas of forward facing panels other than those in Categories I and II.

Category IV Side panels or other non-forward facing transparencies for all rotorcraft other than reconnaissance and search, selected areas of canopies.

Category V Cabin windows, defined areas of canopies.

3.2 REQUIREMENTS FOR CLEAR VISION

3.2.1 Transparencies shall be designed to allow adequate vision for the user under all possible combinations of steady and transient conditions that can occur within the specified flight envelope of the Rotorcraft or on the ground.

3.2.2 The optical qualities of categorised vision areas shall satisfy the requirements of Table 1, unless specified otherwise in the Rotorcraft Specification.

3.2.3 The optical qualities of a transparency shall be demonstrated as installed in correct relationship to the pilot's or user's normal eye positions. In the case of transparencies having replaceable components, for example certain air interspace transparencies, the optical qualities are to be demonstrated for the complete assembly.

3.2.4 The requirements of Chapter 104 shall be met with regard to frame of vision and maintenance of clear vision.

3.2.5 The optical qualities of a transparency shall be designed to meet the requirements with anti-icing, de-icing or de-misting equipment (when fitted) operating. Due attention should be paid to the possible deterioration of optical quality as a result of in-flight loading.

3.2.6 Attention shall be paid to the luminous transmittance and spectral distribution of adjacent panels in order to minimise marked contrast in the properties of such panels. Likewise abrupt differences in deviation between adjacent panels shall be avoided.

3.2.7 Where certain areas of a transparency are allocated special requirements, for example, those associated with weapon aiming, head-up display systems, or reconnaissance needs, careful attention shall be paid to the avoidance of disturbing effects at the junction of such areas with the remainder of the transparency.

3.2.8 When required by the Rotorcraft Specification, the transparency shall be night vision goggle compatible, see Chapter 115.

3.2.9 Ultra-violet transmission in the region of 280-315 nm shall not be greater than 5%.

3.2.10 Less stringent requirements are permitted in less critical areas of the transparency, eg around the edges where such items as busbars for electrical de-icing elements may be fitted. In such cases intrusion into the frame of vision (referred to in para 3.2.4) shall be considered. Small areas within the vision area needed for temperature sensors may suffer deterioration of optical properties; the position and extent of such areas shall be defined as required by para 3.2.2.

3.2.11 Every effort shall be made to avoid the presence of inhomogeneities which would disturb vision.

3.2.12 Unless the Rotorcraft Specification requires otherwise, all transparencies in the Rotorcraft shall be the same colour and where practical should be colourless. Transparencies should not degrade colour perception.

3.2.13 The stability of properties of materials used shall be such that marked discolouration of panels does not occur during the intended life of the transparency.

3.2.14 The choice of materials shall be such as to limit deterioration in use, for example, obscuration due to abrasion by windscreen wipers.

4 LABORATORY TESTS

4.1 Laboratory tests shall be in accordance with Annex A, and the test schedule in Leaflet 715/1.

5 GROUND TESTS

5.1 Transparencies shall be tested as part of a complete Rotorcraft or on a test specimen, consisting of at least a sufficient portion of the fuselage to be representative of support to the transparencies.

5.2 Ground tests shall include the following:

(i) Crashlanding/Ditching (CHAPTER 307),

(ii) Fatigue Tests (CHAPTER 201),

(iii) Resonance/Stiffness (CHAPTER 500),

(iv) Vibration Testing (CHAPTER 501),

(v) Bird Impact (CHAPTER 206).

6 FLIGHT TESTS

6.1 At the earliest practical stage in the flight trials, the magnitude and distribution of the air pressure loads and both the steady and transient temperatures on both inner and outer surfaces of the transparencies shall be measured at all speeds and acceleration up to the design diving speed and maximum acceleration of the Rotorcraft. All likely positions of openable canopies and panels shall be considered.

6.2 The measured loads and temperatures shall be compared with the estimated values used in the design and laboratory tests and the design reviewed accordingly.

7 DEMONSTRATION OF COMPLIANCE

7.1 Compliance with the requirements of this chapter shall be demonstrated by either Design Records, Analysis, Component Test, Ground Test, Flight Test or a combination of these means, in accordance with Table 2, unless otherwise indicated in the Rotorcraft Specification.

Note: Where test data is required it will normally relate to production standard components of the Rotorcraft. However some requirements may be satisfied by data obtained from tests with other representative components subject to the Rotorcraft Design Authority ensuring that this will not result in a deviation from the requirements of the Rotorcraft Specification.

TABLE 1

ACCEPTABLE LIMITS OF THE PARAMETERS ASSOCIATED WITH VISION THROUGH OPTICAL TRANSPARENCIES

PARAMETER	CATEGORY I	CATEGORY II	CATEGORY III	CATEGORY IV	CATEGORY V
IN LINE LUMINOUS TRANSMITTANCE IN HORIZONTAL PLANE IN AREA OF LOWEST TRANSMISSION	 NOT LESS THAN 60% NOT LESS THAN 40%	 NOT LESS THAN 70% NOT LESS THAN 50%	 NOT LESS THAN 55% NOT LESS THAN 40%	 AS CATEGORY III	 AS CATEGORY III
HAZE	NOT MORE THAN 3%	AS CATEGORY 1	AS CATEGORY I	AS CATEGORY I	AS CATEGORY I
ABSOLUTE DEVIATION	5 MINUTES VARIATION FROM AGREED VALUE	NOT MORE THAN 10 MINUTES	NOT MORE THAN 15 MINUTES	NOT MORE THAN 25 MINUTES	NOT SPECIFIED
OPTICAL RESOLUTION	ABILITY TO RESOLVE 1 MINUTE LINES WITH 1 MINUTE SEPARATION	ABILITY TO RESOLVE 10 SECONDS LINES WITH 10 SECONDS SEPARATION	AS CATEGORY I	AS CATEGORY I	NOT SPECIFIED
VISUAL DISTORTION - AS ASSESSED BY DIVERGENCE OF ADJACENT GRID LINES	REQUIREMENT COVERED BY OTHER PARAMETERS	LINE SLOPE NOT GREATER THAN 1 IN 25	LINE SLOPE NOT GREATER THAN 1 IN 20	LINE SLOPE NOT GREATER THAN 1 IN 10	LINE SLOPE NOT GREATER THAN 1 IN 5
BINOCULAR DEVIATION	NOT MORE THAN 10 MINUTES	AS CATEGORY 1. ALSO NOT TO EXCEED 2.5 MINUTES IN VERTICAL DIRECTION	AS CATEGORY I	AS CATEGORY I	NOT SPECIFIED
SECONDARY IMAGE SEPARATION	NOT SPECIFIED	NOT SPECIFIED	NOT SPECIFIED	NOT SPECIFIED	NOT SPECIFIED
VISIBLE INCLUSIONS, SEEDS, HAIRS, FIBRES AND SCRATCHES SEE NOTES	ALLOW 1 TYPE A DEFECT ONLY WITHIN ANY CIRCULAR AREA OF 100 mm RADIUS NO TYPE B DEFECTS NO TYPE C DEFECTS	A) ALLOW 1 TYPE B DEFECT AND 4 TYPE A DEFECTS WITHIN ANY AREA OF 150 mm RADIUS B) ALLOW 8 TYPE A DEFECTS ONLY WITHIN THE SAME AREA NO TYPE C DEFECTS	AS CATEGORY II	A) ALLOW 1 TYPE C DEFECT AND 4 TYPE A DEFECTS WITHIN ANY AREA OF 150 mm RADIUS B) ALLOW 2 TYPE B AND 8 TYPE A DEFECTS IN THE SAME AREA	AS CATEGORY IV

FOR TEST METHODS REFER TO LEAFLET 715/3

FOR DEFINITIONS REFER TO LEAFLET 715/2

NOTES TO TABLE 1

Type A defects: having a diameter in the range 0.2 - 0.5mm or equivalent area (0.03 - 0.2mm^2); this includes hairs, fibres or hair scratches of width not exceeding 0.1mm and equivalent area 0.2mm^2.

Type B defects: having a diameter 0.5 - 1.0mm or equivalent area (0.2 - 0.8mm^2) including hairs etc of width not exceeding 0.02mm and equivalent area 0.8mm^2.

Type C defects: having a diameter 1.0 - 1.5mm or equivalent area (0.8 - 1.8mm^2) including hairs etc of width not exceeding 0.2mm and equivalent area 1.8mm.

Defects larger than Type C not admissible.

The following overriding conditions are to be observed:

Defects which are dense black or of intense colour, and strongly reflecting defects (known as "glint") are not admissible in panels of Category I, areas for weapon aiming, but are admissible in other Category I areas, and in Category II and III areas. Similar black Type B defects are not admissible in category II and II areas.

A local accumulation of defects of dimensions smaller than Type A is admissible provided the haze requirement is met. The haze measurement should then be confined to the area of accumulation.

Permitted defects shall not be within 10mm of each other.

For test methods refer to Leaflet 715/3.

TABLE 2

COMPLIANCE DEMONSTRATION REQUIREMENTS

CH 715 PARA	DESCRIPTION	METHOD OF COMPLIANCE DEMONSTRATION
2.1.1	GENERAL DESIGN REQUIREMENTS	DESIGN RECORDS, ANALYSIS, (COMPONENT TEST OR GROUND TEST), FLIGHT TEST
2.1.2	DESIGN REQUIREMENTS/CONSIDERATIONS	DESIGN RECORDS, ANALYSIS, (COMPONENT TEST OR GROUND TEST), FLIGHT TEST
2.1.3	DESIGN REQUIREMENTS - DITCHING/ CRASHLANDING	(COMPONENT TEST OR GROUND TEST)
2.1.4	DESIGN REQUIREMENTS - TOLERANCES	DESIGN RECORDS, COMPONENT TEST
2.1.5	DESIGN REQUIREMENTS - EYE POSITION	DESIGN RECORDS, ANALYSIS
2.1.6	DESIGN REQUIREMENTS - STORAGE CONDITIONS	(DESIGN RECORDS OR COUPON TEST OR COMPONENT TEST)
2.2.1	MATERIALS - SELECTION	DESIGN RECORDS
2.2.2	MATERIALS - BREAKAGE/FRAGMENTATION	ANALYSIS. (COUPON TESTS OR COMPONENT TEST OR GROUND TEST)
2.2.3	MATERIALS - PROCESS CONTROL DOCUMENT	DESIGN RECORDS
2.3.1	DESIGN FEATURES - CONSIDERATIONS	DESIGN RECORDS, ANALYSIS, COUPON TESTS
2.3.2	DESIGN FEATURES - FOD	(DESIGN RECORDS OR ANALYSIS), (COMPONENT TEST OR GROUND TEST)
2.3.3	DESIGN FEATURES - BIRD IMPACT	(DESIGN RECORDS OR ANALYSIS), (COMPONENT TEST OR GROUND TEST)
2.3.4	DESIGN FEATURES - HUMAN ENGINEERING	ANALYSIS
2.3.5	DESIGN FEATURES - LIGHTNING STRIKE	(DESIGN RECORDS OR ANALYSIS), (COMPONENT TEST OR GROUND TEST)
2.3.6	DESIGN FEATURES - RADAR CROSS-SECTION	(DESIGN RECORDS OR ANALYSIS), COMPONENT TEST
2.3.7	DESIGN FEATURES - INFRA-RED EMISSIONS	(DESIGN RECORDS OR ANALYSIS), COMPONENT TEST
2.3.8	DESIGN FEATURES - MAGNETIC EMISSIONS	(DESIGN RECORDS OR ANALYSIS), COMPONENT TEST
2.3.9	DESIGN FEATURES - CANOPY GLINT	(DESIGN RECORDS OR ANALYSIS)
2.3.10	TEST PROGRAMME FOR CANOPY GLINT	DESIGN RECORDS, COUPON TESTS
2.3.11	LASERS AND PARTICLE BEAM WEAPONS	(DESIGN RECORDS OR ANALYSIS), (COMPONENT TEST OR GROUND TEST)
2.3.12	DESIGN FEATURES - NUCLEAR DETONATION	(DESIGN RECORDS OR ANALYSIS)
2.3.13	NUCLEAR BIOLOGICAL AND CHEMICAL CONTAMINATION & DECONTAMINATION CLEANERS	DESIGN RECORDS, (COUPON TESTS OR COMPONENT TEST OR GROUND TEST)
2.4.1	VULNERABILITY TO BATTLE DAMAGE	DESIGN RECORDS, ANALYSIS, (COUPON TESTS OR COMPONENT TEST OR GROUND TEST)
3.1.1	CATEGORIES FOR CLEAR VISION	DESIGN RECORDS
3.2.1	CLEAR VISION	DESIGN RECORDS, ANALYSIS, COMPONENT TEST, FLIGHT TEST
3.2.2	OPTICAL QUALITIES - DESIGN	DESIGN RECORDS, ANALYSIS, COMPONENT TEST, FLIGHT TEST
3.2.3	OPTICAL QUALITIES - DEMONSTRATION	COMPONENT TEST
3.2.4	OPTICAL QUALITIES - CLEAR VISION	DESIGN RECORDS, COMPONENT TEST
3.2.5	OPTICAL QUALITIES - DEICING AND DEMISTING	(COMPONENT TEST OR GROUND TEST) FLIGHT TEST
3.2.6	OPTICAL TRANS AND SPECTRAL DISTRIBUTION	DESIGN RECORDS, COMPONENT TEST
3.2.7	OPTICAL QUALITY - CONTINUITY	DESIGN RECORDS, COMPONENT TEST
3.2.8	NIGHT VISION GOGGLE COMPATIBILITY	DESIGN RECORDS, (COMPONENT TEST OR COUPON TESTS)
3.2.9	ULTRA-VIOLET TRANSMISSION	DESIGN RECORDS, (COMPONENT TEST OR COUPON TESTS)
3.2.10	OPTICAL QUALITY REQUIREMENTS - RELAXATION OF	DESIGN RECORDS
3.2.11	OPTICAL QUALITIES - INHOMOGENEITIES	COMPONENT TEST
3.2.12	OPTICAL QUALITIES - COLOUR	DESIGN RECORDS (COMPONENT TEST OR COUPON TESTS)
3.2.13	OPTICAL QUALITIES - DISCOLOURATION	DESIGN RECORDS (COMPONENT TEST OR COUPON TESTS)
3.2.14	OPTICAL QUALITIES - DETERIORATION	DESIGN RECORDS, (COMPONENT TEST OR COUPON TESTS)
4.	LABORATORY TESTS	AS DESCRIBED IN ANNEX A
5.	GROUND TESTS	GROUND TESTS
6.1	FLIGHT TESTS	FLIGHT TESTS

BRACKETS INDICATE ALTERNATIVES

CHAPTER 715 - ANNEX A

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

LABORATORY TESTS

1 GENERAL

1.1 Each new type of transparency used in Grade A applications shall be tested to demonstrate its strength and to indicate its probable reliability under the conditions of Chapter 715 paras 2.1.1 and 2.1.2 and the relevant parts of Chapter 201. Details of the test schedule are given in Leaflet 715/1.

1.2 When selecting the critical case(s) for testing, the following points shall be considered:

(i) the magnitude of the loads, thermal and otherwise,

(ii) the sequence in which loads are imposed,

(iii) the duration of the loads,

(iv) the relation of the load/time/temperature patterns to the physical characteristics of the material concerned,

(v) temperature effects upon mixed material laminates.

1.3 The following cases shall be considered when formulating the test programme and covered as far as practicable from both strength and deflection aspects:

(i) that which gives the most severe total stress,

(ii) that which gives the most severe steady stresses other than thermal stresses,

(iii) that giving the most severe transient stresses other than thermal stresses, and

(iv) that giving the most severe total thermal stress.

Note: For organic materials, a critical temperature exists above which the structural properties change significantly. The critical temperature for each different material shall be determined. Tests above and below this temperature shall be considered where applicable.

1.4 Additional tests. When the Rotorcraft Specification calls for a particular transparency to meet special requirements additional tests to demonstrate proof of compliance shall be agreed with the Rotorcraft Project Director during the project definition stage.

1.5 During testing, in all cases where the simulated conditions are not more severe than those for which the frame or mounting has been designed, the actual frame or mounting shall be used. If the testing involves the application of conditions of greater severity, suitable supports for the frame or mounting beyond proof loading or a special structure may be used. In either case the support or special structure shall be such that edge conditions, stiffness and heat transfer are closely represented, and that the stress distribution in the transparency at the relevant factor is as close as possible to that obtainable in the unfactored flight or ground condition, from strain gauge tests.

1.6 For cyclic testing, each specimen shall be mounted in a local structure fully representative of the Rotorcraft upon which appropriate Rotorcraft systems which might affect the component (eg demisting air jets), are represented. Where they may be significant, the effects of flight inertia loads and externally produced airframe deflections shall also be represented. The required number of cycles and the overall form of the programme shall be based upon the most severe operating conditions and replacement life for the transparency, and demonstrate compliance with the airworthiness requirements for the Rotorcraft type.

1.7 Testing shall be in accordance with:

(i) Para 1.2 for test cases,

(ii) Para 3 for sequence, applicability and minimum number of specimens for test,

(iii) Leaflet 715/1 for test schedules.

1.8 Except where otherwise specified in the schedules, all loads and temperature effects applied in the tests shall be those arising from compliance with Chapter 715 paras 2.1.1 and 2.1.2, they shall be applied at rates representative of those anticipated in service, and, where this may be significant, shall be applied in the sequence anticipated in service.

Notes: 1 In special circumstances, the severity of the test requirements and the number of test specimens may be relaxed. Examples of such are:

(i) individual research Rotorcraft,

(ii) simple acrylic unformed panels, and

(iii) lightly stressed panels.

2 If a Rotorcraft includes a number of similar transparencies of identical thickness and edge restraint, the thermal stresses in which do not exceed 10% of the total stress, testing may be confined to specimens of that item calculated to be subject to the highest stress.

2 NOMENCLATURE OF TESTS

Test No 1 Contraction and Expansion Test

Test No 2 Strength Tests:
Proof - Stage 2A
Ultimate - Stage 2B

Test No 3 Cyclic Reliability Test

Notes: 1 Stage 2A and Stage 2B tests will be done in the sequence defined in paras 2.2 and 2.3 of Leaflet 715/1.

2 (i) (LT) Denotes the critical design cases occurring below the critical temperature.

(ii) (HT) Denotes the critical design cases occurring above the critical temperature.

3 APPLICABILITY AND SEQUENCE OF TESTS

3.1 Where more than one critical design case occurs, further specimens may be required for Test No 2 (fully factored test) under each critical condition.

Note: Where it can be shown that there is no significant change of the material or component during testing of the first critical case, then that same specimen may be used for testing additional critical cases at the discretion of the Rotorcraft Design Authority.

3.2 Glass. Minimum number of specimens required - 5.

Specimen No.	Sequence of Tests
1, 2 and 5	1, 2A, 2B
3 and 4	1, 3

Note: In cases where it can be shown that the thermal stress is less than 10% and at the same time where it can be shown by calculation that there is a strength reserve factor of at least 10 on the limit load in the critical loading case, Specimen Nos 2 and 5 may be omitted.

3.3 Organic Materials. For transparencies where flight temperatures do not exceed the material critical temperatures, the minimum number of specimens required for test shall be 4.

Specimen No	Sequence of Test
1	1, 2A (LT) 2B (LT)
2	2A (LT) i) and ii) 2B (LT)
3	1, 3
4	1, 3

For transparencies where flight temperatures exceed the material critical temperature, the minimum number of specimens required for test shall be 5.

Specimen No	Sequence of Test
1	1, 2A (LT) 2B (LT)
2	1, 2A (LT) i) and ii) 2B (LT)
3	1, 3
4	1, 3
5	1, 2A (HT) 2B (HT)

Notes: 1 For organic materials in LT cases the additional Specimens (para 3.1) shall be subjected to the test sequences of specimens 1 and 2 and for HTcases the No 5.

2 Leaflet 715/1 Table 2 shows how a reduction of factor can be obtained by testing an increased number of Specimens to the full requirements of Test No 2.

3.4 Glass and plastic composite screens.

Test Procedure as for Organic Materials (para 3.3).

Note: This sequence of tests can be conducted in one of two ways:

Either the transparency is subjected to the loads consistent with the highest load factor of the materials employed.

or,

The transparency is subjected to a series of tests, each at loads commencing with the lowest and progressing to the highest.

The transparency shall be deemed to be acceptable if each material survives the test at its particular load factor.

4 FACTORS

4.1 The variability factors to be applied in Test No 2 shall be those stated in Tables 1 and 2.

4.2 The total factors of Table 3 include normal proof and ultimate factors.

4.3 The variability factors of Leaflet 715/1 Tables 1 and 2 are applicable to the minimum of the test results not the mean.

5 RE-TESTING

5.1 After clearance of a design under the test procedure outlined above, any change in the following:

(i) its design,

(ii) the design of any local effective structure,

(iii) the manufacturer, or

(iv) any process in the method of manufacture.

Consideration shall be given to re-testing the design.

6 FLIGHT TESTING BEFORE COMPLETION OF LABORATORY TESTS

6.1 Any flight testing done before completion of Tests Nos 1 and 2 shall be subject to flight limitations determined in accordance with the principles stated in Leaflet 900/1, para 3.2.

LEAFLET 715/0

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

REFERENCE PAGE

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DTD 218	Laminated safety glass
DTD 761	Safety glass windscreen, weapon sighting quality
DTD 869	Laminated safety glass, high light transmission
DTD 900/6017	Process for the toughening of structural glasses
DTD 900/4953	Triplex Ten Twenty glass
DTD 900/6016	Triplex Ten Twenty glass for non-structural applications
DTD 925	The fabrication of acrylic panels and shapings
DTD 1082	Dry air sandwich windscreens
DTD 1088	Sealed dry air sandwich windows
DTD 5548	Glazing compound
DTD 5576	Laminated safety glass heated by electrically conducting films
DTD 5592	Acrylic sheets for aircraft glazing
DTD 5625	Standard annealed float glass glazing
DTD 5626	High light transmission annealed plate glass for aircraft glazing

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EN 2155	Test methods for transparent materials for aircraft glazings

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171	The evaluation Ten Twenty glass
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R & M No 3654	Strength variability in structural materials

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AFML-TR-65-212	Transparent materials for aircraft enclosures
AFFDL-TR-73-103	Windscreen bird strike structure design criteria
AFML-TR-73-126	Conference on transparent aircraft enclosures
AFFDL-TR-74-75	Improved windshield and canopy protection development
AFFDL-TR-75-150	Bird impact forces in aircraft windshield design
AFFDL-TR-75-2	Vol I bird strike alleviation techniques
AFFDL-TR-75-115	Windshield/canopy/support structure life cycle cost and failure analysis
AFML-TR-76-54	Conference on aerospace transparent materials and enclosures
AFMDL-TR-76-75	Effects of laboratory simulated precipitation static electricity and swept-stroke lightning on aircraft windshield sub systems
AFFDL-TR-76-114	The determination of deflection and stress distribution for a transparent laminated beam
AFFDL-TR-77-1	Vol I and II. Windshield technology demonstrator programme. Detail design options study
AFFDL-TR-77-92	Evaluation of windshield materials subjected to simulated supersonic flight environments
AFFDL-TR-77-141	Precipitation static electricity and swept-stroke lightning effects on aircraft transparency coatings
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MIL-W-81752(A)	Windshield systems, fixed wing aircraft, general specification for
MIL-P-83310	Plastic sheet, polycarbonate, transparent

LEAFLET 715/1

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

LABORATORY TEST SCHEDULE

1 TEST No 1 - CONTRACTION AND EXPANSION TEST

1.1 Cool the specimen to 5°C below the lowest temperature expected in the air or on the ground but making no allowance for alleviation by:

(i) solar radiation, or

(ii) the effect of any Rotorcraft system.

1.2 Maintain the cooling until a uniform temperature distribution is achieved and hold this temperature for a minimum period of one hour to ensure stable conditions.

1.3 Maintain the cooling medium and switch the Rotorcraft de-ice/de-mist system to the appropriate level for the maximum period designated for continuous use or until steady conditions are reached.

1.4 Switch off de-ice/de-misting system and remove cooling medium; allow specimen to regain room temperature.

1.5 Heat the specimen to the highest temperature expected in the air or on the ground taking into account the effect of:

(i) solar radiation, and

(ii) any Rotorcraft system.

1.6 Maintain the heating until a stable uniform temperature is achieved. Maintain this uniform temperature for a minimum period of one hour if the design condition is derived from a ground soak or for a time equal to the maximum endurance of the Rotorcraft if the design condition is derived from a flight condition.

1.7 Remove the heating medium and allow the assembly to return to room temperature.

1.8 For specimens incorporating an electrical method for de-misting or de-icing purposes, check this feature for correct operation.

1.9 Repeat the cycle 1.1 to 1.8 twice more.

1.10 Inspect the specimen. The specimen shall exhibit no failure or optical defect of any kind due to the tests including delamination, peel, splintering or crazing.

2 TEST No 2 - FULLY FACTORED TEST

2.1 This test is designed to check the strength of the transparency in all operating conditions and comprises two stages. The loading conditions and total factors to be achieved in both test stages are summarised in Table 3.

Note: Throughout this test, strict safety precautions must be observed so as to avoid the possibility of injury to personnel in the event of failure of the transparency, particularly during inspection of a panel under load.

2.2 STAGE 2A

(i) Bring the specimen to the unfactored design thermal condition relevant to the test and stabilise the surface temperatures. If the critical case is a transient thermal condition, the unfactored design thermal condition relevant to the test will be the steady state thermal condition immediately preceding the transient thermal condition.

(ii) Apply a factored load to the specimen at a representative rate (or, if more convenient at 34 kPa/minute). The factor shall be 1.5 times the variability factor in Tables 1 or 2.

(iii) Maintain this condition for a period equal to three quarters of the maximum endurance of the Rotorcraft or for two hours whichever is the longer except when the non-thermal loading is transient in which case the conditions need only be maintained for five minutes.

(iv) Where feasible, inspect the specimen at the end of this period while maintaining steady conditions. (See Note above on the need for safety precautions). The criteria for rejections are the same as those specified in Test No 1 (x). If the state of the specimen is satisfactory on completion of the inspection do not relax the loading condition but continue into Stage 2B.

(v) Where it is not possible to carry out a visual inspection for reasons such as masking by the rig, the loading and thermal conditions may be completely relaxed to enable this to be achieved. If satisfactory, reload the specimen to the conditions of (i), (ii) and (iii) and continue into Stage 2B.

2.3 STAGE 2B

2.3.1 With the specimen still under the temperature and loading conditions of Stage 2A:

(i) (a) If the critical case is a steady state thermal condition, maintain the temperature conditions of Stage 2A and load the specimens at a rapid uniform rate until double the variability factor in Tables 1 or 2 is attained.

(b) If the critical case is a transient thermal condition, maintain the pressure loading at double the variability factor in Tables 1 or 2 and at the same time adjust the rate of heating and cooling of the specimen to obtain a factor of 1.2 on the thermal stress distribution.

The transparency temperature should not deviate excessively from the design temperature at the fully factored condition.

(ii) On attaining the above test condition, relax all loading and temperature conditions completely.

(iii) Inspect the specimen. All specimens subject to Test No 2 are required to sustain the conditions without any failure of the material designed to be load bearing.

3 TEST No 3 - CYCLIC RELIABILITY TEST

3.1 This test is intended to investigate the reliability of the assembly under representative environmental conditions. The test is not intended to establish the fatigue life of the transparency.

3.2 It consists of the repeated application of unfactored thermal stresses, inertia loadings and pressure loadings appropriate to the normal flight plan of the Rotorcraft and includes the application of a number of special conditions.

3.3 The content of the normal test cycle, the special conditions and the stages at which they are to be considered, shall be determined by the Rotorcraft Design Authority to meet the requirements of the Rotorcraft Specification.

3.4 NORMAL CYCLE

3.4.1 The conditions are to include:

(i) External aerodynamic pressure loads and all thermal effects.

(ii) Each cycle of loading is to represent, as closely as is practicable, the transient flight phases for ascent, descent, acceleration and deceleration as well as the "steady state" cruise. The cruise condition, in respect of time, is to be represented by:

(a) the normal cruise time, or

(b) the time needed for the temperature conditions in the assembly to reach equilibrium,

whichever is the shorter, with the proviso that the time is not to be less than 30 minutes.

(iii) Between each cycle, after the specimen has reached ground temperature, a "rest" period of at least 30 minutes is required. Ground temperatures here are those appropriate to the areas in which the Rotorcraft is designed to operate.

3.5 EMERGENCY AND SPECIAL CONDITIONS

3.5.1 The following should be considered:

(i) Maximum limits of the design flight envelope.

(ii) The operation of any relevant Rotorcraft system under extreme environmental conditions which are not represented in the normal cycle.

(iii) Thermal shock and any additional effect due to flight into rain, hail etc.

(iv) Shortened rest periods representative of "quick turn-round" time.

(v) Opening and closing of transparent panels.

(vi) Ground manoeuvring and wind loading sustained by the transparency in open position on the ground.

(vii) Transient loading from pressure pulses caused by firing of adjacent armaments, eg ship or aircraft armaments.

3.6 TEST RESULTS

3.6.1 An acceptable level of reliability will have been demonstrated if, after completion of the full cyclic test programme, the specimen is still considered serviceable. Failure of the specimen during this test does not necessarily mean rejection of the design providing the transparencies are fully functional as laid down by the Acceptance Test Schedule, ie structurally sound, electrically operable and with optical qualities which comply with the requirements of Chapter 715 para 3.2.2.

TABLE 1

VARIABILITY FACTORS - GLASS

Materials of Load Bearing Members	**Variability Factor** **Grade A Items**
Annealed soda-lime or annealed alumino-silicate glass	3.3
Fused silica	3.3
Thermally toughened aluminosilicate glass	1.4
Thermally toughened soda-lime glass	1.9
Chemically toughened glass	1.4

Note: Confirmation that the variability factor quoted in Table 1 applies to material provided by a special manufacturer should be established. Variability factors for materials not listed in this table should be established by the Rotorcraft Design Authority.

TABLE 2

VARIABILITY FACTORS FOR PERSPEX P TO DTD 5592

Description of Item	**Number of Specimens Tested to the Full Requirements of Test No. 2 See Chapter 715 Annex A Para 3.3)**		
	1	2	3
Grade A items which have undergone a forming operation	2.75	2.40	2.20
Grade A items which have not undergone a forming operation	2.35	2.05	1.95

TABLE 3

TOTAL FACTORS - ALL MATERIALS

Test Stage	Loading Condition		Total Factors Applicable
No. 2 Stage A	Steady	Non-thermal	1.5 x Variability Factor in Table 1 or Table 2
		Thermal	1.0
No. 2 Stage B	Steady	Non-thermal	2.0 x Variability Factor in Table 1 or Table 2
		Thermal	1.0
	Transient	Non-thermal	2.0 x Variability Factor in Table 1 or Table 2
		Thermal	1.2

LEAFLET 715/2

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION DEFINITIONS, CATEGORIES OF TRANSPARENCIES AND LIMITING ACCEPTABLE VALUES OF PARAMETERS ASSOCIATED WITH ACCURATE VISION

1 INTRODUCTION

1.1 Because of misunderstandings which have occurred in the past it is essential that there should be clear definitions of the qualities which can influence vision through a transparency . These are given in para 2 of this leaflet. It is furthermore expedient to distinguish the various transparencies used in Rotorcraft according to their function. Limiting values may then be placed upon the qualities associated with good vision, and the tolerances adjusted to suit the nature of each category of transparency thereby effecting economies in manufacture and inspection.

1.2 In para 4 the categorisation of transparencies is discussed, and in para 5 the limiting values which have been allocated are explained.

2 DEFINITIONS

2.1 The qualities which determine vision through a transparency are listed and, where necessary, defined below. It is recognised, as will be seen later (para 5.7), that depending on the function of a particular transparency, not all of these parameters need be specified for every class of transparency.

2.1.1 In-line luminous transmittance: this may be defined as the intensity of an emerging beam of light compared with that of the incident parallel beam falling upon the transparency under examination. Losses are due to absorption and scattering within the specimen and reflection occurring not only at the outer surfaces, but also at interfaces within which there are changes of refractive index.

2.1.2 Haze or halation: an obscuration of view or the spreading of an image beyond its proper limits caused by the scatter of light from tiny particles within the material or defects on the surface of the material, eg lack of polish.

2.1.3 Absolute deviation: the angular deviation of an emergent beam of light compared with the incident beam. A ray of light emerging at an angle other than normal from a transparent specimen with perfectly parallel faces suffers a linear displacement compared with the incident beam. If the faces are not parallel, the emergent beam also suffers an angular deviation compared with the incident beam.

2.1.4 Optical resolution: The ability to distinguish clearly between two objects which subtend a small angle at the eye. The ability to distinguish fineness of detail (acuity) is strongly influenced by the contrast in brightness of the object compared with the brightness of the background.

2.1.5 Binocular deviation: the difference in deviation of two parallel incident rays which, for this application, are considered to be 64mm apart in a horizontal plane, this being the average spacing of the human eyes. Visions through a panel having binocular deviation can result in eye strain and double vision.

2.1.6 Visual distortion: over the area of specimen, local variations of deviation (or 'wedge') can occur which result in distortion of the image; known straight lines become crooked or curved and the effects are magnified in the case of thick panels. Curved panels can be particularly susceptible to optical distortion in some areas.

2.1.7 Secondary image separation: in an oblique view through a panel, secondary image formation can appear because of internal reflections at interface where there are changes of refractive index. The effect is particularly disturbing when manifest in the appearance of multiple images of landing lights. Two factors contribute to the appearance of double images, namely the extent of separation of the secondary image from the primary, and the contrast between the secondary image and the background illumination. The separation is increased by increased thickness of the panel and by increased angle of obliquity.

2.1.8 Scratches and inclusions: panels may be accidentally scratched during manufacture and inclusions may be present in the materials used or may arise in lamination. Both types of defects can be disturbing under some conditions of illumination; the detection of such defects relies very much on the method of illumination and the eye of the inspector.

2.1.9 Angle of incidence: this is defined as the angle between the incident beam of light and the normal to the panel at the point of incidence.

3 CONDITIONS UNDER WHICH THE REQUIREMENTS APPLY

3.1 Many of the properties listed change markedly as the angle of incidence varies and satisfactory assessment of these can only be made from the correct viewing position. The optical qualities can suffer severe deterioration in the case of a highly raked windscreen where the angle of incidence can be 60° or more.

3.2 One further condition is that certain optical qualities be checked when de-misting and de-icing equipment associated with the transparencies is operating under normal control conditions thereby ensuring that no further deterioration of vision occurs when flying conditions are likely to be poor due to bad weather etc. Where flight loads are likely to affect the optical quality, this should be checked under the appropriate conditions.

4 CATEGORISATION OF TRANSPARENCIES

4.1 It is important to grade transparencies or areas thereof (see Chapter 715 para 3.1) according to the function which they perform; for example a pilot's transparencies or area thereof, associated with some weapon-aiming device must have higher quality and accuracy associated with it than, for example, a screen provided to give the pilot a wider angle of vision. On the other hand the cabin window of a Rotorcraft must not be so poor as to cause the passenger visual discomfort.

4.2 For reasons of economy designers should take care not to specify higher optical quality for transparencies than is required to satisfy the end use. To achieve this, the most critical requirements should be confined to limited areas of the panel. As an example the critical area of a weapon aiming transparency would be limited to the area covered by the weapon aiming system, but the surrounding areas could be reduced from Category I to Category II.

4.3 While significant improvements beyond the defined optical qualities are highly desirable for the improved performance and their attainment is to be the aim of the designer, due regard must be paid to any excessively high cost of manufacture or inspection likely to be incurred by their achievement.

5 ACCEPTABLE LIMITS FOR PARAMETERS ASSOCIATED WITH VISION

5.1 In Chapter 715 Table 1 acceptable limits of the various parameters (see paras 5.3 - 5.8 below) are stated, these being considered as limiting values consistent with safe operation. In many cases improvements in the quality of vision are possible thereby increasing the effectiveness of the operator using the transparency. The Rotorcraft Specification should therefore state the values agreed by designer and manufacturer and justification provided when improved requirements demand abnormally high quality materials or control in fabrication; should any degraded standard of optical quality be required in the Rotorcraft Specification justification for this must be provided.

5.2 The vision area normally covers the whole area of the panel to within 25mm of the edge except in areas as agreed necessary for the inclusion of sensors, busbars etc. Critical areas of panels having special requirements (eg for weapon aiming) shall be defined on the drawing or in the specification.

5.3 IN-LINE VISUAL LUMINOUS TRANSMITTANCE

5.3.1 For transparencies of Categories I, II and III it is necessary not only to stipulate minimum acceptable values of in-line luminous transmittance from the pilot's or user's normal eye position in the horizontal plane (when the Rotorcraft is in horizontal flight), but also in the area where there is the lowest light transmittance. The latter would normally occur at the lowest part of the panel (ie where angle of incidence from the user's eye position is at a maximum) so that specification of the transmittance in this area ensures the best vision under critical flight conditions (eg on landing).

5.3.2 The quoted minimum acceptable in-line luminous transmittance values for Category I and III transparencies (ie 60% and 55% respectively in the horizontal plane) are effectively controlled by the need for thick panels to achieve bird strike protection. The transmittance decreases with increasing angle of incidence and 40% transmittance is quoted as a just acceptable value where the angle of incidence to the pilot's or user's eye is at its maximum. This minimum value must be maintained at the aircrews eyes when wearing devices which may reduce the overall transmittance, eg visors, spectacles, etc. Therefore consideration must be given to all such devices for which there is a requirement in the Rotorcraft Specification. For Category II transparencies, as used in reconnaissance and search roles, the maximum possible transmittance is required and, for such panels, bird strike requirements do not normally exist and angles of incidence are usually low so that high values of luminous transmittance are achievable.

5.3.3. For Categories IV and V in-line luminous transmittance of 80% is achievable when heating films are not required. However the customer may choose designs of lower in-line luminous transmittance in which case the minimum acceptable value in the installed position should be 55%.

5.3.4 Transmittance measurements are normally made using a standard light source and every care must be taken to ensure that heating films do not significantly affect the spectral distribution of the light reaching the observer's eye.

5.4 HAZE OR HALATION

5.4.1. The effect of haze in a transparency is very similar to a deterioration in optical resolution. The maximum allowable value for haze of 3.0% of the total visible light transmission is measured normal to the panel since it is impracticable to measure haze at angles of incidence other than normal; the difference is in any case negligible. The value of 3.0% is normally readily obtainable with glass laminate but may cause problems with plastic materials due to their susceptibility to abrasion and scratching.

5.5 ABSOLUTE DEVIATION

5.5.1 In the extreme case, imprecise definition of the requirement for optical deviation could lead to inaccuracy in weapon aiming or in the positioning of a Rotorcraft approaching a landing site. The requirements are defined to minimise these effects and weapon aiming will always demand special requirements; for Category 1 a variation of g 5 minutes from a specified value is quoted as a guide, the real value would be determined by the weapon system tolerance.

5.5.2 For Category II the maximum acceptable value of 10 minutes is determined by the reconnaissance role. Relaxation to 15 minutes and 25 minutes is allowed for Categories III and V respectively in view of the less stringent need for accuracy of position in their associated roles.

5.6 OPTICAL RESOLUTION

5.6.1 The generally accepted practical limit of resolution of the human eye is about 30 seconds of arc under good conditions and it is considered that the transparencies used in operational flying of a Rotorcraft should not impair this resolution beyond 1 minute of arc. Conditions of test must be made under standardised conditions of contrast and illumination. For Category II transparencies which are frequently used in conjunction with binoculars the resolution afforded by the panel must not cause deterioration of vision. Since 7 x 50 binoculars are in common usage, a criterion of 10 seconds of arc imposed upon Category II vision areas ensures adequate performance when binoculars are used.

5.7 VISUAL DISTORTION, BINOCULAR DEVIATION AND SECONDARY IMAGE SEPARATION

5.7.1 It should be noted that if the absolute deviation is rigorously specified as is required for the highest quality screens, and the tolerance is achieved over a sufficiently fine lattice covering the transparency or area thereof, these dependent parameters (ie visual distortion, binocular deviation and secondary image separation) are automatically controlled and need not be further specified.

5.7.2 In normal circumstances however and especially in the case of curved panels, visual distortion needs to be specified and assessed in order to show that undistorted vision is provided. Thus it is a requirement that panels of Categories II, III, IV and V be assessed for visual distortion. It is usual to quantify distortion in terms of the slope and/or divergence of the image lines of a grid projected through the panel, or obtained by an equivalent method.

5.7.3 Similarly panels of Categories III and IV must be assessed for binocular deviation which must not exceed 10 minutes of arc. Thicker panels having areas of Category I quality may need to be checked for binocular deviation, particularly in vision areas where Categories I and II adjoin, For Category II a special additional requirement arises from the possible use of binoculars. Because the eyes have less ability to compensate for deviations in the vertical plane than in the horizontal plane, binocular deviation in the vertical direction must not exceed 2.5 minutes of arc, that in the horizontal direction remaining at not greater than 10 minutes of arc.

5.7.4 Separation of secondary images normally becomes serious only under certain circumstances and specification of acceptable limits is necessary only in such circumstances. Thus, if on the preliminary viewing of such a panel the images of bright lights (as on a landing site) become separated and move about on slight movement of the head, then specification and assessment of this parameter is required. No limits have been quoted as it is impossible to eliminate entirely double imaging except at the cost of very high optical quality. Should specification be required however, then 5 minutes separation would be a recommended value.

5.8 VISIBLE INCLUSION, SEEDS, HAIRS, FIBRES AND SCRATCHES

5.8.1 Considerable problems are associated with formulating requirements for inclusions etc as certain conditions of illumination can render a small defect intensely irritating whereas it might normally be invisible.

5.8.2 The eyes normally focus on the distant scene rather than upon the transparency unless this exhibits some disturbing (or interesting) feature. The guiding principle is therefore to avoid such disturbances by specifying their size to be less than the limit of resolution of the eye and their colour to be unattractive.

LEAFLET 715/3

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

RECOMMENDED METHODS FOR THE DETERMINATION OF OPTICAL QUALITIES

1 INTRODUCTION

1.1 Suitable methods for determining the various properties are described in the following sections. Alternative methods, or the combination of two or more of the inspection processes, may be acceptable provided that adequate inspection for each of the included parameters can be demonstrated.

1.2 List of Sections:

(i) In Line Visual Light Transmission,

(ii) Haze,

(iii) Absolute Deviation,

(iv) Optical Resolution

(a) Categories I, III, IV and V Transparencies,

(b) Category II Transparencies Only,

(v) Visual Distortion,

(vi) Binocular Deviation,

(vii) Secondary Image Separation,

(viii) Visible Defects.

SECTION (i) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF IN-LINE VISUAL LIGHT TRANSMISSIONS

1 INTRODUCTION

A photometric method is used employing a defined light source and a photocell matched to the response of the human eye. Because this method deals with materials which are basically colourless certain deviations from the ideal conditions are allowable and these are indicated.

2 APPARATUS (See Figure 1)

2.1 LIGHT SOURCE

2.1.1 For the purposes of this specification the light source is deemed to be illuminant A of the International Commission on Illumination, ie a tungsten filament lamp operated at a colour temperature of 2855.5K (See Note below). For practical purposes the colour temperature tolerance may be g 10% and this will be attained by employing a gas filled tungsten filament lamp operated at its rated voltage.

2.1.2 The light source is combined with an optical system to produce a parallel beam.

Note: Although source 'A' does not correspond to solar radiation temperature, it has been chosen for simplicity in use, and ease of maintaining a standard. Practical tests have shown that measurements made with a photometer corrected to approximate eye response did not reveal a change in transmission of more than 0.5% when the illuminant was changed from 'A' to 'C'. These tests were made on a selection of nearly colourless materials of 50-95% transmission.

2.2 POWER SUPPLY

2.2.1 The power supply to the lamp should be stabilised. The short term change in voltage output should be not more than ±0.1%.

2.3 PHOTOMETER - GEOMETRY

2.3.1 The photometric measurements may be made either with a photocell or alternatively with a photometric integrating sphere.

2.3.2 The photocell should be fitted with a diffusion screen and should be of sufficient dimension to cover the whole of the parallel light beam.

2.3.3. The integrating sphere should conform to the requirements of BS 354: Photometric Integrators. It should have an aperture of 70mm diameter for the light beam and an aperture for fitting the photometer. There should be baffles fitted so that direct light does not reach the photometer from the aperture or the area on the sphere where the light beam falls. The diameter of the sphere shall be approximately 500mm.

2.3.4 The distance of the photometer from the illuminated area of the transparency should not be less than 250mm.

2.4 PHOTOMETER - SPECTRAL RESPONSE

2.4.1 The spectral response of the photocell shall be corrected to approximate that of the photopic (cone) response of the human eye.

2.5 PHOTOMETER - ACCURACY

2.5.1 The output response of the photometer shall be linear within ±0.5%. The output shall be capable of being read to a discrimination of ±0.5%.

3 CONDITIONS OF TEST

3.1 The sample shall be cleaned on both surfaces before measurements are made.

3.2 The measurements shall be made with respect to the pilot's or user's normal eye position and with the windscreen at the installed angle.

3.3 The area of the parallel light beam shall be at least 1 cm^2.

3.4 Precautions shall be taken to ensure that no other light by reflection or other means except that from the source shall reach the photocell.

4 METHOD

4.1 The light source is set up at a convenient distance from the photometer with both items rigidly mounted.

4.2 The apparatus is allowed appropriate time to reach thermal equilibrium before measurements are made.

4.3 A measurement is made without the transparent specimen (a) and a second measurement (b) with the specimen interposed between the lamp and the photometer. The percentage light transmission is then given by $\frac{100b}{a}$.

4.4 When testing thick panels at angles of incidence other than normal, there is a substantial displacement of the light beam after passing through. This may be accommodated with the integrating sphere photometer if the aperture is large enough. When using a photocell alone, a lateral movement of the light source or photocell will be necessary to ensure that the exit beam is fully received by the photocell.

5 CALIBRATION

5.1 Each of the parameters specified in para 2 (ie (a) colour temperature of light source, (b) stability of power supply, (c) spectral response of photocell and (d) linearity of photometer), may be qualified by submission to a laboratory accredited for optical measurements by the National Measurement Accredition Service.

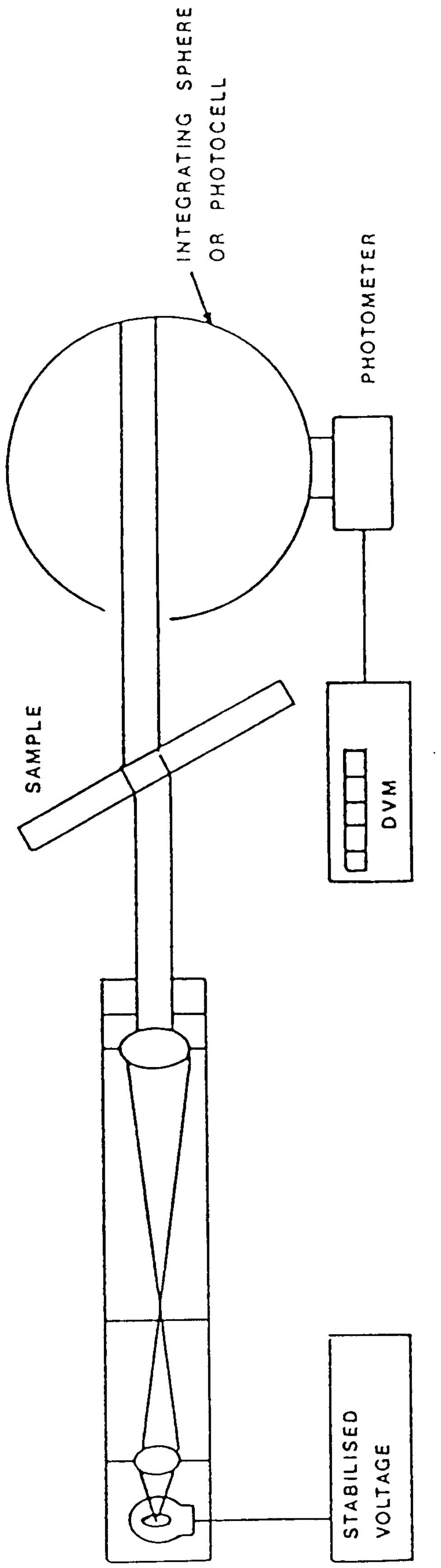

Figure 1 - Measurement of In-Line Visual Light Transmission

SECTION (ii) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF HAZE

1 INTRODUCTION

1.1 A photometric method is used employing a defined light source, an integrating sphere and a photometer matched to the response of the human eye. Because this method deals with materials which are basically colourless, certain deviations from the ideal conditions are allowable and these are indicated.

2 APPARATUS (See Figure 2)

2.1 LIGHT SOURCE

2.1.1 For the purpose of this specification the light source is deemed to be Illuminant A of the International Commission on Illumination, ie a tungsten filament lamp operated at a colour temperature of 2855.5K (See Note below). For practical purposes the colour temperature tolerance may be g 10% and this will be attained by employing a gas filled tungsten filament lamp operated at its rated voltage.

2.1.2 The light source is combined with an optical system to produce a parallel beam 50mm diameter.

Note: Although source 'A' does not correspond to solar radiation temperature, it has been chosen for simplicity in use, and ease of maintaining a standard. Practical tests have shown that measurements made with a photometer corrected to approximate eye response did not reveal a measurable change in haze measurement when the illuminant was changed from 'A' to 'C'. The tests were carried out on a selection of nearly colourless transparent materials.

2.2 POWER SUPPLY

2.2.1 The power supply to the lamp should be stabilised. The short term change in voltage output should be not more than ±0.01%.

2.3 INTEGRATING SPHERE

2.3.1 The integrating sphere should conform to the requirements of BS 354: Photometric Integrators. It should have an aperture of 70mm diameter for the light beam and an aperture for fitting the photometer. There should be baffles fitted so that direct light does not reach the photometer from the aperture or the area on the sphere where the light beam falls. The diameter of the sphere shall be approximately 500mm.

2.4 PHOTOMETER - SPECIAL RESPONSE

2.4.1 The spectral response of the photocell shall be corrected to approximately that of the human eye.

2.5 PHOTOMETER - ACCURACY

2.5.1 The output response of the photometer shall be linear within ±0.5%. The output shall be capable of being read to a discrimination ±0.5%.

2.6 WINDSCREEN HOLDER

2.6.1 There shall be provided a holder for the windscreen panel capable of holding the windscreen in a plane at right angles to the light beam and capable of moving the windscreen along the light beam a distance equal to the diameter of the sphere.

3 CONDITIONS OF TEST

3.1 The sample shall be cleaned on both surfaces before measurements are made.

3.2 The measurements shall be made with the light beam normal to the surface of the transparency.

3.3 The light source (ie parallel light beam), must be as free as possible from inherent light scatter and must be provided with adequate baffles and the lenses kept clean.

3.4 The haze is expressed as the percentage light scattered of the transmitted light and is an average of readings from three randomly chosen areas.

4 METHOD

4.1 The apparatus is set up as shown in the diagram. It is essential that all components are rigidly held in relation to one another.

4.2 The apparatus is allowed appropriate time to reach thermal equilibrium before measurements are made.

4.3 A measurement is made of the total transmission with the transparency in contact with the sphere aperture (a). A second measurement is made with the transparency moved away from the aperture a distance equal to the diameter of the sphere (b). The scattered light in the second measurement will fall outside the sphere aperture and will not be measured. The percentage haze will therefore be $\frac{100(a-b)}{a}$.

5 ALTERNATIVE METHOD

5.1 An alternative method is given in American Standard Test Methods ASTM D1003-52 and this method is acceptable but the general measurements of light beam diameter and sphere diameter outlined above should be maintained.

6 CALIBRATION

6.1 Each of the parameters specified in para 2 (ie (a) colour temperature of light source, (b) accuracy and stability of power supply, (c) spectral response of photocell and (d) linearity of photometer), may be qualified by submission to a laboratory accredited for optical measurements by the National Measurement Accredition Service.

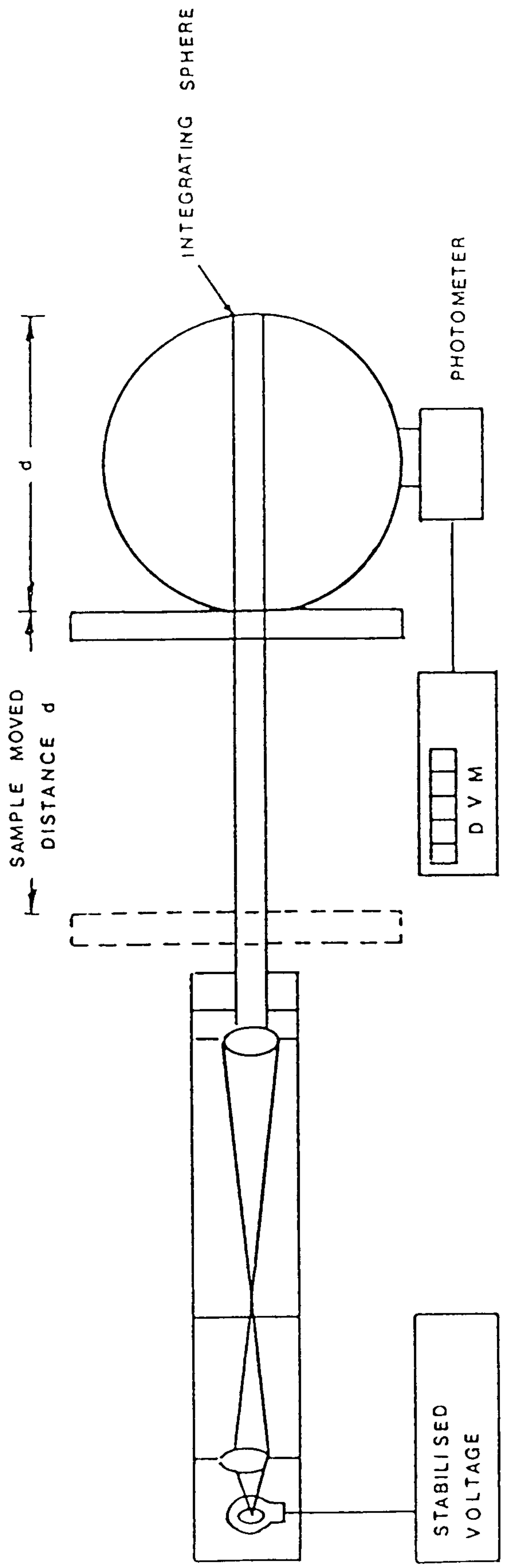

Figure 2 - Measurement of Haze

SECTION (iii) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF ABSOLUTE DEVIATION

1 INTRODUCTION

1.1 In this method a collimator and telescope are aligned and measurements of absolute deviation are made from the pilot's eye position.

2 APPARATUS (See Figure 2)

2.1 The collimator shall have an object glass of at least 50mm diameter and a focal length of not less than 500mm. In practice it is convenient to choose a much larger collimator (ie, with an object glass of 150mm diameters and a focal length of 1m), as this makes:

(i) the alignment much easier,

(ii) allows a binocular system with a single collimator, and

(iii) eliminates any need for any correction for beam displacement.

2.2 The collimator graticule shall be a black cross on a transparent ground; the angular dimension of the thickness of the graticule in the focal plane of the collimator objective shall be one minute.

2.3 The telescope shall have an object glass of 12mm diameter or greater and a focal length of approximately 500mm. The magnification shall be approximately 15.

2.4 The telescope graticule shall consist of a centre black cross with a black concentric circle on a transparent ground. The angular dimensions of the graticule in the focal plane of the telescope objective shall be: thickness of line ¼ minute, length of cross lines 5 minutes, diameter of circle equal to the limit of deviation being measured. Where more than one limit is specified for different areas of the transparency more than one circle can be marked on the graticule.

2.5 The collimator and telescope shall be mounted on solid supports and aligned so that the graticule of the collimator when illuminated appears in the telescope eyepiece superimposed on the telescope graticule.

2.6 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The whole vision area of each panel as defined by the drawing shall be examined.

3.2 The panel shall be supported at such an angle that the axis of the viewing telescope is identical with a line of sight from the designed eye position, as shown on the appropriate drawings, to a central area of the panel.

3.3 The area of the panel shall be examined by scanning along a series of traverses, the separation between adjacent traverses having been agreed to adequately scan the area of the panel for optical quality.

4 METHOD

4.1 The aperture of the telescope is limited to 12mm. With no panel in position, the centres of the two graticules are brought into coincidence.

4.2 The panel to be tested is mounted in the appropriate position on its holder and any change in position of the image of the collimator cross in the telescope noted.

4.3 The panel is then scanned by suitable movement of the holder and relationship of the two graticules observed. The centre of the image of the collimator graticule cross shall not extend beyond the concentric ring of the telescope.

4.4 When the panel is inserted between the telescope and the collimator, the image of the collimator graticule in the eyepiece of the telescope may become blurred or indistinct. This may also occur if the panel is scanned. Such areas should be identified and examined for optical resolution (Section (iv)).

Note: When testing thick panels there is a substantial displacement of the light beam after passing through but this can be accommodated if the diameter of the collimator is large enough. If, however, the collimator is not large enough a correction will have to be made. This can be achieved by mounting the collimator or telescope on a precision parallel movement or by fitting a compensating thick parallel glass plate in front of the telescope and correcting the light beam by setting the angle of the parallel plate.

5 CALIBRATION

5.1 The angular subtense of the graticule may be qualified, either by submission to a laboratory accredited for optical measurements by the National Measurement Accredition Service, or by measurement with a certified instrument.

SECTION (iv) - PART (a) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF OPTICAL RESOLUTION FOR CATEGORIES I, III, IV AND V TRANSPARENCIES

1 INTRODUCTION

1.1 In this method a collimator and telescope are aligned and the optical resolution assessed from the pilot's eye position.

2 APPARATUS

2.1 The collimator shall have an object glass of at least 50mm diameter and a focal length of not less than 500mm. In practice it is convenient to choose a much larger collimator (ie with an object glass of 150mm diameter and a focal length of 1m), as this makes the alignment much easier and eliminates the need for any correction for beam displacement.

2.2 The telescope shall have an object glass of 12mm diameter or greater and a focal length of approximately 500mm. The magnification shall be approximately 15.

2.3 The collimator graticule shall consist of a black double-lined cross with vertical and horizontal lines on a transparent background. The angular dimension of the thickness of the graticule line in the focal plane of the collimator objective shall be one minute and the space between the lines shall be one minute.

2.4 The collimator and telescope shall be mounted on solid supports and aligned so that the image of the collimator graticule, when illuminated, appears in the telescope eyepiece.

2.5 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The area to be examined will have been identified from observations during the measurement of Absolute Deviation Section (iii).

3.2 The panel shall be supported at such an angle that the axis of the viewing telescope is identical with a line of sight from the designed eye position, to a central area of the panel, as shown in the appropriate drawings.

3.3 The panel shall be rotated, vertically or horizontally, about the designed eye position until the area to be examined appears on the axis between the collimator and telescope.

4 METHOD

4.1 The aperture of the telescope is limited to 12mm. With no panel in position, a clear image of the collimator graticule should be visible in the telescope eyepiece.

4.2 The panel to be tested is mounted in its appropriate position and the area to be examined brought on the optical axis of the telescope. The quality of the image is examined.

4.3 The image of the collimator graticule is considered resolved if the image can be identified as being made up of two parallel lines, (ie the two lines should not have blended together to be indistinguishable from a single line).

4.4 An unresolved image which can be improved and made acceptable by refocussing the telescope may be considered resolved providing the refocussing is within the range g 0.1 dioptre.

Note: When testing thick panels there is a substantial displacement of the light beam after passing through, but this can be accommodated if the diameter of the collimator is large enough. If, however, the collimator is not large enough the collimator or telescope may have to be moved to compensate.

5 CALIBRATION

5.1 The angular subtense of the graticule lines may be qualified, either by submission to a laboratory accredited for optical measurements by the National Measurement Accredition Service, or by measurement with a certified instrument.

SECTION(iv) - PART (b) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF OPTICAL RESOLUTION FOR CATEGORY II WINDSCREENS ONLY

1 INTRODUCTION

1.1 In this method a target and a pair of 7 x 50 conventional prismatic (not roof prism type) binoculars are aligned and the resolution assessed from the user's eye position.

2 APPARATUS

2.1 The target shall consist of a black double-lined cross with vertical and horizontal lines on a white background. The angular dimension of the thickness of the lines and the space between the lines from the viewing position shall be 10 seconds of arc (1mm at 20m). The angular dimension of each arm of the cross shall be not more than 1 minute.

2.2 The binoculars shall have a magnification of seven times and an objective lens diameter of 50mm (7 x 50).

2.3 The target and binoculars shall be mounted on solid supports 20m apart (other distances may be used providing the binoculars focus satisfactorily and that the target makes the correct angular subtense).

2.4 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The whole vision area as defined by the drawing shall be examined.

3.2 The panel shall be supported at such an angle that the axis of the viewing system is identical with a line of sight from the designed eye position, to a central area of the panel, as shown in the appropriate drawings.

3.3 The area of the panel shall be examined by scanning along a series of horizontal traverses, and then a series of vertical traverses, the separation between adjacent traverses having been agreed to adequately scan the area to be examined.

4 METHOD

4.1 With no panel in position a clear image of the target shall be visible through the binoculars.

4.2 The panel to be tested is mounted in its appropriate position and the quality of the image examined.

4.3 The image of the target is considered resolved if the image can be identified as being made up of two parallel lines, (ie the two lines should not have blended together to be indistinguishable from a single line).

4.4 The panel is then scanned by suitable movement of the holder and the quality of the image observed.

4.5 An unresolved image which can be improved, and made acceptable, by refocussing the binoculars may be considered resolved. Alternatively, if the image is resolved by one eye, it may also be considered resolved.

SECTION (v) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF VISUAL DISTORTION

1 INTRODUCTION

1.1 In this method the image of a grid is projected on to a screen to assess the visual distortion through a transparency as seen from the normal viewing position.

2 APPARATUS

2.1 A graticule consisting of equally spaced parallel lines in two directions at 90° to one another on a clear ground shall be rigidly held in the slide carrier of a 50mm x 50mm lantern slide projector.

2.2 The spacing of the lines of the graticule shall be such that the sides of each square form an angular dimension of 30 minutes in the focal plane of the projector, the lines being one minute thick.

2.3 The projector, fitted with a 150mm focal length lens shall be firmly mounted 5m from a screen, (See Figure 3).

2.4 The screen approximately 1.5m wide and 1m high shall be constructed of a translucent material so that the image can be viewed from the opposite side of the screen to which it is projected. The screen shall be marked with a grid of lines which exactly superimposes upon the projected image. There shall be a facility for a parallel adjustment of the grid lines to make the superimposing possible.

2.5 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The whole vision area of each panel as defined by the drawing shall be examined.

3.2 The panel shall be supported at such an angle that the optical axis of the projector is identical with a line of sight from the designed eye position, as shown on the appropriate drawings, to a central area of the panel.

3.3 The transparency support shall be in a position between the projector and screen such that the pilot's eye position is 4m from the screen.

3.4 The area of the panel shall be examined by rotating the panel horizontally or vertically in its support to cover the area to be examined.

4 METHOD

4.1 The projector and screen are set up with no panel in position and the projected image and grid are brought into coincidence.

4.2 The panel to be tested is mounted in the appropriate position on its holder. A movement of the lines is noted and the position of the grid may be moved in any direction parallel to the grid lines to restore coincidence. Any regular or curved lines on the projected image may then be noted.

4.3 Any area showing distortion shall be brought to the middle of the screen for measurement. The slope shall be measured by laying a straight edge on the projected line if straight, or tangential to it if curved. The slope is expressed as the ratio of the vertical and horizontal intercepts made by the straight edge with the lines on the screen.

Note: In using this method a camera may be substituted for the projector and a gridboard substituted for the screen. The geometrical distances remain unchanged. Photographs may be taken and measured to give the slopes.

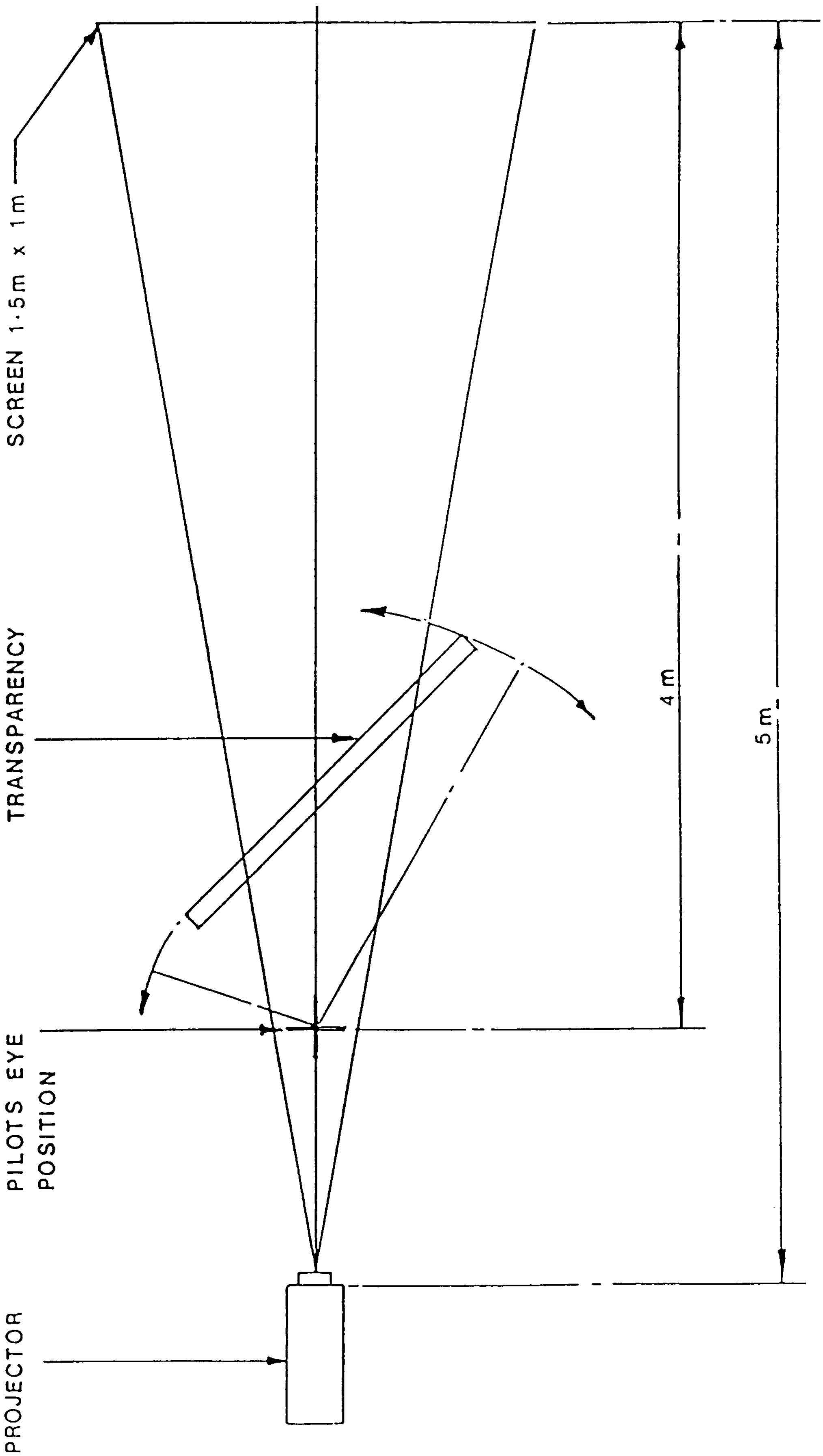

Figure 3 Measurement of Visual Distortion

SECTION (vi) SPECIFICATION OF A METHOD FOR THE MEASUREMENT OF BINOCULAR DEVIATION

1 INTRODUCTION

1.1 In this method a collimator and telescope are aligned and measurements of binocular deviation are made from the pilot's eye position.

2 APPARATUS

2.1 The collimator or collimator system may consist of one of the following alternatives:

(i) Collimator having an object glass of at least 150mm diameter and focal length of 1m.

(ii) Two collimators each having an object glass of about 50mm diameter and focal length of 500mm, placed with their optical axes parallel and 64mm apart in a horizontal plane.

(iii) One collimator having an object glass of about 50mm diameter and focal length of 500mm fitted with a beam splitter to convert the single optical axis into two parallel optical axes 64mm apart in a horizontal plane.

2.2 The collimator graticule shall consist of an opaque disc placed at the centre of a cross on a transparent ground. The graticule shall be placed in a plane through the principal focus of the collimator, perpendicular to its optical axis. The angular dimension of the disc in the focal plane of the collimator objective shall be 10 minutes diameter and the lines of the cross one minute thick. For Category II panels the opaque disc should be elliptical, 2.5 minutes vertical and 10 minutes horizontal.

2.3 The telescope or telescope system may consist of either of the following.

(i) A telescope having an object glass of at least 120mm diameter and a focal length of about 500mm. The magnification shall be approximately 15. The object glass shall be provided with a mask in which two apertures 12mm diameter and 64mm apart have been placed equally disposed horizontally about the centre.

(ii) A telescope having an object glass 12mm diameter or more, but stepped down to that dimension, and a focal length of about 300mm. The magnification shall be approximately 15. The telescope shall be fitted with a beam splitter to receive the two (nearly) parallel lines of sight, and then to superimpose them along a single line of sight.

2.4 The telescope graticule shall consist of a centre black cross on a transparent background. The angular dimensions of the graticule in the focal plane of the telescope objective shall be; thickness of line 1/4 minute; length of arms of cross lines 10 minutes.

2.5 The collimator and telescope shall be mounted on solid supports and aligned so that the images of the illuminated graticule or graticules of the collimator system appear superimposed upon each other and in line with the telescope graticule.

Note: Any combination of telescope and collimator system described can be used but when testing thick panels there is a substantial displacement of the light beam after passing through; this can be accommodated if the diameter of the collimator is large enough. If, however, the collimator is not large enough a correction will have to be made. This can be achieved by mounting the collimator or telescope on a precision parallel movement or by fitting a compensating thick parallel glass in front of the telescope and correcting the light beam by setting the angle of the parallel plate.

2.6 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The whole vision area of each panel as defined by the drawing shall be examined.

3.2 The panel shall be supported at such an angle that the axis of the viewing telescope is identical with a line of sight from the designed eye position, as shown on the appropriate drawings, to a central area of the panel.

3.3 The area of the panel shall be examined by scanning along a series of horizontal traverses, and then a series of vertical traverses, the separation between adjacent traverses having been agreed to adequately scan the area of the panel, for optical quality.

4 METHOD

4.1 The centres of the two graticules are brought into coincidence with no panel in position.

4.2 The panel to be tested is then mounted in the appropriate position on its holder and the images observed in the eyepiece of the telescope. The two images from binocular view through the transparency will superimpose on each other if there is no difference between the deviation in each view. Any difference will be shown as two images overlapping. When the specified tolerance is exceeded, separation of the two images occurs.

4.3 The panel is then scanned by suitable movement of the holder and any separation of the collimator image noted.

SECTION (vii) SPECIFICATION FOR THE MEASUREMENT OF SECONDARY IMAGE SEPARATION

1 INTRODUCTION

1.1 In this method a collimator and telescope are aligned and measurements of secondary image separation are made from the pilot's eye position.

2 APPARATUS (See Figure 4)

2.1 The collimator shall have an object glass of at least 50mm diameter and a focal length of not less than 500mm. In practice it is convenient to choose a much larger collimator (ie with an object glass of 150mm diameter and a focal length of 1m), as this makes:

(i) the alignment much easier,

(ii) allows the same collimator to be used for measuring absolute deviation and binocular deviation, and

(iii) eliminates the need for any correction for beam displacement.

2.2 The Collimator shall be provided with a beam splitter to allow two graticules to be superimposed upon one another.

2.3 One graticule shall consist of a clear spot on a dark background. The angular dimension of the diameter of the spot in the focal plane of the collimator shall be one minute. The second graticule shall consist of concentric circles producing angular dimensions in the focal plane of the collimator of 1, 2, 3, 4, 5, 10 and 20 minutes. The graticule shall consist of clear lines on a dark ground. The individual components of the collimator are shown in Figure 4.

2.4 The telescope shall have an object glass of 12mm diameter or greater, and a focal length of approximately 500mm. The magnification of the eyepiece shall be approximately 15.

2.5 The telescope graticule shall consist of a centre black spot on a transparent ground. The angular dimension of the spot in the focal plane of the telescope objective shall be 1 = minutes.

2.6 The collimator and telescope shall be mounted on solid supports and aligned so that the centre spot of the collimator when illuminated appears in the telescope eyepiece superimposed on the centre of the telescope graticule. The image of the second graticule shall also be superimposed and concentric with the other graticules. The visible contrast of the image may be enhanced by introducing a red filter behind the collimator spot graticule and a green filter behind the concentric ring. It is advisable to have a dimmer control on the concentric rings graticule to increase the contrast between the bright centre spot and the measuring graticule.

2.7 The transparency shall be held in a support which can rotate about vertical and horizontal axes, both axes passing through a point which will be the designed eye position.

3 CONDITIONS OF TEST

3.1 The area to be examined shall be defined by the drawing.

3.2 The panel shall be supported at such an angle that the axis of the viewing telescope is identical with a line of sight from the designed eye position as shown on the appropriate drawings.

3.3 The area of the panel shall be examined by scanning along a series of horizontal traverses, the separation between adjacent traverses having been agreed to adequately scan the area of the panel for optical quality.

4 METHOD

4.1 The panel to be tested is mounted in the appropriate position on its holder.

4.2 The panel is then scanned by suitable movement of the holder. The position of the primary and secondary images are observed.

4.3 When a definite measurement is required the scanning is stopped at the appropriate position and the position of the telescope adjusted until the collimator and telescope graticules are brought into coincidence. The angular difference between the primary and secondary images may then be observed.

Note: When testing thick panels there is a substantial displacement of the light beam after passing through but this can be accommodated if the diameter of the collimator is large enough. If however, the collimator is not large enough a correction will have to be made. This can be achieved by mounting the collimator or telescope on a precision parallel movement or by fitting a compensating thick parallel glass plate in front of the telescope and correcting the light beam by setting the angle of the parallel plate.

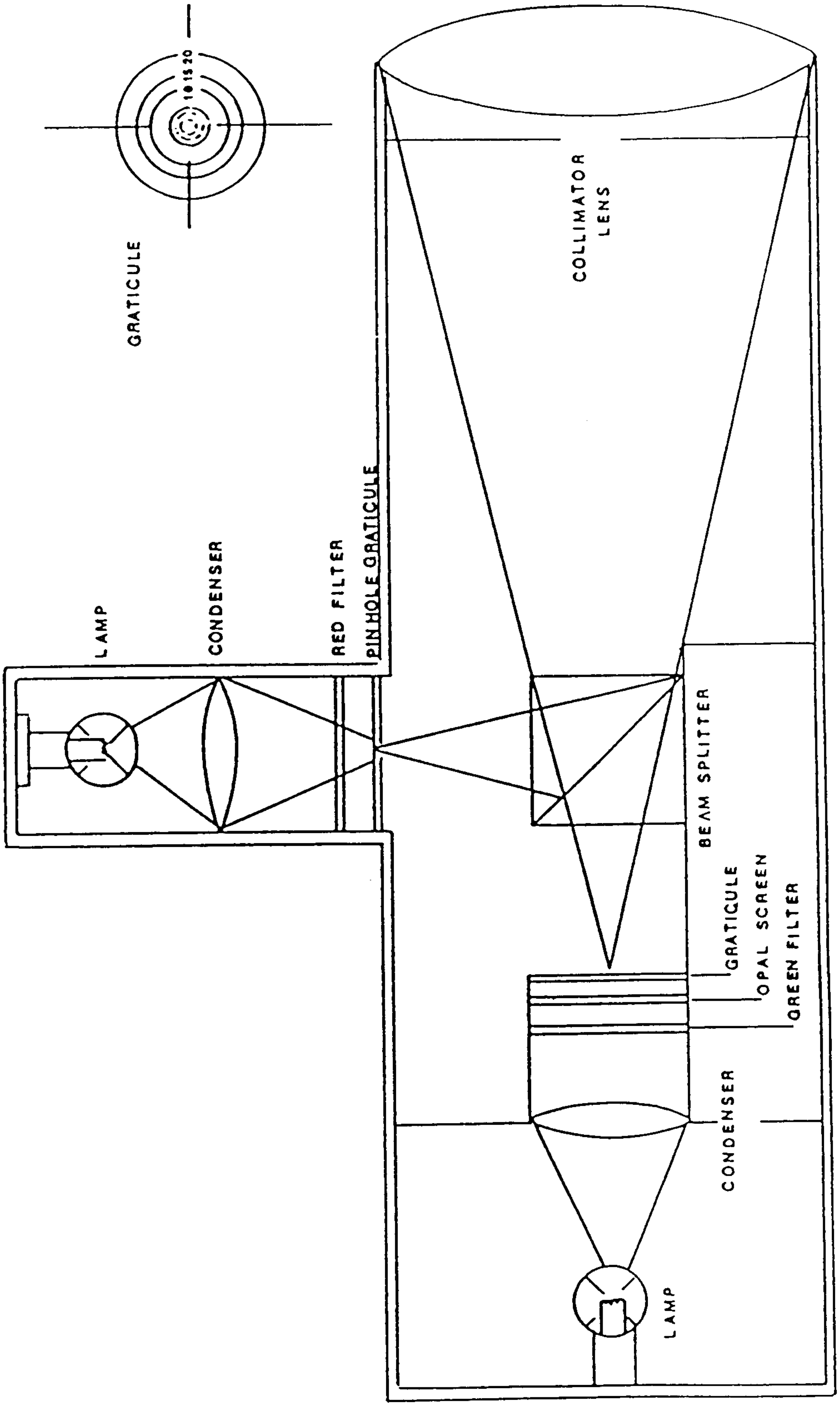

Figure 4 Collimator for Measurement of Secondary Image Separation

SECTION (viii) SPECIFICATION FOR THE MEASUREMENT OF VISIBLE DEFECTS IN AIRCRAFT TRANSPARENCIES

1 INTRODUCTION

1.1 This specification shall apply to the visible defects in the vision area of any optical transparency for which this needs to be defined as stated in Chapter 715. Panels are initially examined for the presence of defects by viewing against an evenly illuminated background. Identification and measurement of the defects is subsequently carried out under strong oblique illumination.

2 APPARATUS

2.1 A suitable apparatus (See Figure 5) consists of a horizontal matt white screen large enough to accommodate the area to be examined and provided with even illumination. A convenient arrangement is afforded by two 40W fluorescent lights set just below the panel and on each side of it and shaded so that the screen is illuminated without direct line of vision between the viewer and these lights. Provision is made for the panel under examination to be supported parallel to the screen and at a convenient distance from it.

2.2 The white screen is replaced by a matt black screen and the shading of the light is adjusted so that the viewer can examine the panel against the strong oblique illumination provided by the lights but again without direct line of vision into them. A method of measurement is provided by a microscope of magnification approximately 10 fitted with a graticule graduated in 0.1mm. Where necessary the area of each defect shall be assessed by approximating to the nearest simple geometrical shape.

3 METHOD

3.1 The panel under examination shall be cleaned on both surfaces before examination and measurement.

3.2 The panel shall be examined against the white background and the defects marked.

3.3 The panel shall be examined against the black background with oblique illumination and only additional visible defects marked.

3.4 The defects shall be measured and counted against which ever background should be most revealing. Suitable masks delineating the circles of specified radius are helpful in determining the density of defects.

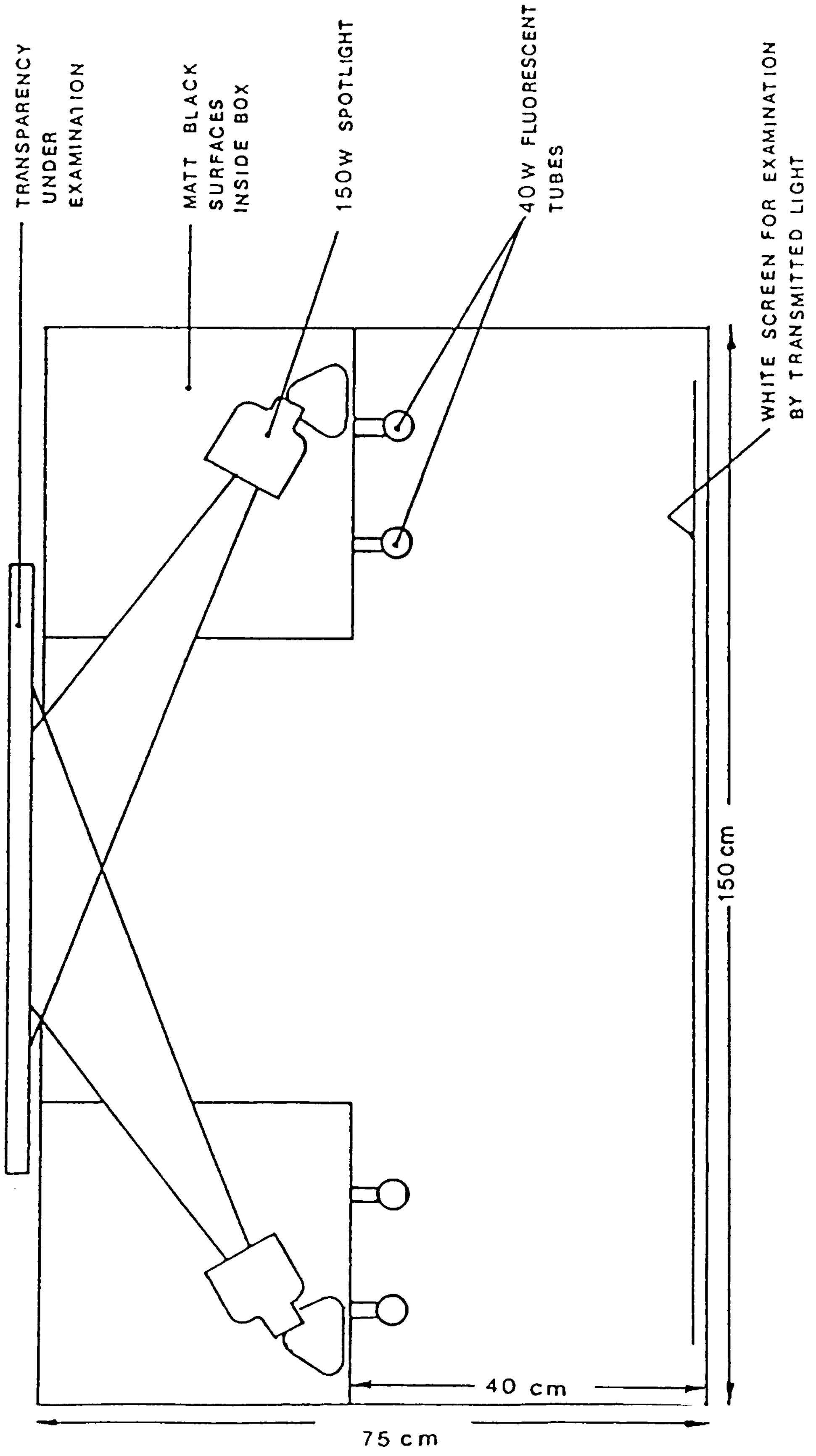

Figure 5 Light Box for Examination of Transparencies for Scratches etc

LEAFLET 715/4

OPTICAL TRANSPARENT COMPONENTS

DESIGN, INSTALLATION, TESTING AND REQUIREMENTS FOR CLEAR VISION

REDUCTION OF CANOPY GLINT

1 INTRODUCTION

1.1 This leaflet advises on means of compliance with the requirement of Chapter 715 paras 2.3.9 and 2.3.10.

2 MEANS OF COMPLIANCE

2.1 Several potentially successful techniques for the reduction of canopy glint are known. These include anti-reflective coatings, non-reflective canopy materials, etching the canopy outer surface, shading the canopy and the use of plane canopy surfaces such that directional glint is not sustained in a mobile operational situation.

2.2 Any coating which is applied should be resistant to abrasion, normal cleaning and erosion experienced during flight. It should be resistant to aircraft fluids (eg fuel, hydraulic fluids etc) and, if appropriate, wear caused by windscreen wiper action.

2.3 It is essential to achieve a balance between improvements in vulnerability and penalties incurred to obtain those improvements.

CHAPTER 716

STATIC AND PITOT PRESSURE SYSTEMS

1 INTRODUCTION

1.1 The requirements of this Chapter govern the design of static and pitot pressure systems (see Chapter 711 para 2.5 for the protection of Air Data Sensors).

2 SYSTEM CONFIGURATION

2.1 Each rotorcraft shall have a minimum of either one pitot-static tube or a combination of pitot tube(s) and flush static vents, mounted such as to provide unambiguous indications of altitude and airspeed within the pressure error tolerances defined in para 3.3.

2.2 The static and pitot pressure system shall be designed to minimise moisture or dirt ingression into the system.

2.3 A duplicated static and pitot pressure system shall be provided for two independent Air Speed Indicator (ASI) Systems unless the Rotorcraft Specification calls for a single system. A duplex system shall not use a single data source.

2.4 In order to reduce lag, consideration shall be given to conversion of pressure to an electrical signal as close as possible to the pressure sensor.

2.5 Where an altitude or airspeed indicating system requires an electrical supply for their operation, a secondary electrical source shall be available to permit continued operation in the event of failure of the primary or normal supply.

3 PRESSURE ERRORS

3.1 GENERAL

3.1.1 In all flight conditions errors in the static and pitot system shall be kept to a minimum, but in any event they shall not exceed the tolerances specified at paras 3.2 and 3.3.

3.1.2 The errors quoted below are exclusive of instrumentation errors which are given in the appropriate instrumentation specification.

3.2 LAG ERRORS

3.2.1 The time interval, or lag, for 90% of a pressure change at the source to appear at the pilots primary ASI and altimeter shall not exceed 0.5 seconds at ground level.

3.3 PRESSURE ERROR TOLERANCE

3.3.1 When the rotorcraft is in level flight or descending on a flight path inclined at less than 10° to the horizontal, or in any other steady state flight condition (including turns and yawed flight, sidewards and rearwards flight), the pressure error shall not lead to an altitude reading differing from the correct altitude by more than 30 ft.

3.3.2 When the rotorcraft is in level flight or descending on a flight path inclined at less than 10° to the horizontal, at an airspeed exceeding 30kt, the pressure error shall not lead to an airspeed reading differing from the correct speed by more than 5kt or 3%, whichever is the greater.

3.3.3 The Vertical Speed Indicator (VSI) shall provide a constant indication of rate of climb and descent at all Equivalent Airspeeds (EAS) above 40kt. Small changes in rotorcraft pitch attitude or airspeed shall not affect the indication during an otherwise steady climb or descent. In transient flight conditions, including rapid entry to climbs or descents, the VSI shall indicate in the correct sense of the change without excessive hesitation.

3.3.4 The accuracy of the VSI shall be within ± 10% of the instrument reading at all rates of climb and descent below 4000 ft/min, and at airspeed in excess of 40kt EAS. Ideally the error should be no worse than ± 5% of the true rate of change of pressure height.

3.3.5 Changes in configuration that are permitted in normal flight (e.g., release of external stores, opening/closing of doors) shall not cause the airspeed to vary by more than ± 2kt or the altitude to vary by more than ± 15 ft. Changes in configuration that may occur in emergency situations (e.g., deployment of Flotation Gear) shall not generate airspeed and altitude pressure errors that prejudice rotorcraft safety.

3.4 AIRSPEED INDICATION

3.4.1 The ASI system shall provide consistent indicators of relative airspeed along the flight path during all normal manoeuvres required by the rotorcraft. These indicators should be available at above 20kt EAS and rates of climb/descent (ROC/D) of less than 200 ft/min; and above 40kt EAS at ROC/D above 1000 ft/min. These requirements should be met at sideslip angles up to 20°.

3.4.2 The minimum airspeed above which consistent indications are produced during acceleration from hover, and the minimum speed down to which consistent indications are produced during decelerations, shall be repeatable for all rotorcraft loadings and configurations.

3.4.3 In all transient manoeuvres the airspeed indication should be available at airspeeds above 40kt EAS and shall be available at airspeeds in excess of 70kt EAS. Entry to autorotative flight and recovery from steep pitch attitudes are defined as important transient manoeuvres in this context.

4 AERODYNAMIC REPEATABILITY

4.1 PITOT AND PITOT-STATIC HEADS

4.1.1 The static pressure measurement when referenced against a known standard shall not differ by more than ± $0.002q_c$, where q_c is true dynamic pressure (impact pressure). The pitot pressure when referenced against a known standard shall not differ by more than ± $0.005q_c$ (See Leaflet 716/2).

5 PITOT-STATIC SYSTEMS IN POWERED FLYING CONTROLS

5.1 Where the pitot-static pressure is monitored and signalled into the powered flying controls, the pitot-static system shall be regarded as part of the powered flying controls and shall meet the requirements of Chapter 605, para 2.

6 PITOT-STATIC SOURCES UTILISED IN AUTOMATIC FLIGHT CONTROL SYSTEMS AND FOR MANUAL BLIND FLYING

6.1 The characteristics of the static and/or pitot sources, such as lag of the complete pressure system or its response to changes of altitude, shall not result in destabilising signals to any mode of automatic flight control system.

7 INSTALLATION OF PIPES

7.1 GENERAL

7.1.1 The aim shall be to ensure that the installation of pitot and static pipes, together with their components and supports, will be such that they are capable of withstanding throughout the life of the rotorcraft the worst effects of vibration, structural distortion and temperature likely to occur. All conditions within the design flight envelope of the rotorcraft shall be taken into account.

7.2 PRECAUTIONS AGAINST INCORRECT ASSEMBLY (Ref Chapter 100, para 7).

7.2.1 Where possible, joints in pipe lines shall be staggered.

7.2.2 All pipes shall be marked in accordance with the requirements of Chapter 806.

7.2.3 All pipes shall carry a supplementary identification marking P or S as appropriate complying with the requirements of BS M23.

7.2.4 Pipe to pipe couplings and pipe connections to instruments and associated equipment shall be in accordance with DEF STAN 47-25.

7.3 PROTECTION FROM WATER

7.3.1 The installation of the pipe lines for the static and pitot systems shall be such as to minimise the entry or the accumulation of water. In this connection pipes having an inside diameter less than 6.25mm (0.25 in) will not normally be acceptable.

7.3.2 When a static vent is fitted, there shall be an immediate vertical rise of at least 150mm in the pipe at the static vent inlet, wherever this is possible.

7.3.3 When it is impracticable to have a continuous rise in the pipe lines from the static or pitot pressure source to the instruments, accessible drain traps which are proof against leaks shall be fitted at the lowest points between each fall and rise in the pipe lines. The lowest points shall be determined when the rotorcraft is in the normal attitude on the ground.

7.3.4 The capacity of the drain traps shall be kept to a minimum to limit the lag in the system but shall be sufficient to cater for the condensation arising as a result of one flight in the most critical case (for condensation) of the roles specified under the maximum humidity conditions of Chapter 101.

7.4 PROTECTION FROM ENEMY ACTION

7.4.1 When two static vent systems and/or two pressure heads are fitted, the run of the pipe lines shall be as far apart as possible, preferably on opposite sides of the fuselage.

8 INSTALLATION OF STATIC AND PITOT SOURCES

8.1 PITOT-STATIC BOOM MOUNTING

8.1.1 Pitot and pitot-static probes shall be equipped with a positive means of alignment, to prevent incorrect installation. Consideration shall be given to marking probes, where appropriate, to indicate intended alignment. The probe(s) shall be securely mounted, with particular regard to birdstrike requirements.

8.2 PITOT-STATIC SYSTEM ANTI-ICING

8.2.1 The Pitot-Static system shall be capable of continuous operation in the icing environment defined in Leaflet 711/1 such that it will still meet the requirements of para 3 in respect of pressure errors, and the system calibration shall not change by the equivalent of more than 0.5 kt IAS in airspeed or 5 ft in indicated altitude. Icing effects shall be considered at various angles of attack in the choice of location of flush static vents.

8.2.2 Where a pitot or a pitot-static probe is mast mounted, any icing protection supplied to the mast shall be so connected that it will operate concurrently with the pitot or pitot-static probe anti-icing system.

8.2.3 All electrically de-iced pitot and pitot-static systems shall be equipped with a monitor to detect heater failure. The monitor shall not degrade the reliability of the heater circuitry.

8.3 HOSES

8.3.1 Where flexible hoses are required (e.g., for connecting to pitot-static instruments or associated equipment fitted with adaptors conforming to DEF STAN 47-17, Table 1) they shall be in accordance with AGS 3914 for pitot lines and AGS 3915 for static lines.

9 STATIC VENTS

9.1 When the rotorcraft is in any steady flight condition, at zero sideslip, and the rotorcraft is subsequently yawed to a sideslip angle of 10^{o}, the airspeed indicator shall not change by more than 2kt, and the altitude indication shall not change by more than 15 ft.

9.2 When the static pressure is derived from static vents, these shall be fitted at corresponding positions on each side of the fuselage each being interconnected with the corresponding vent on the opposite side of the fuselage, and the instrument connection being taken from the mid-point of this interconnection.

10 PRESSURE SOURCES FOR EQUIPMENT

10.1 The pitot and static systems supplying the pilot's primary instruments shall not be used for instruments or equipment which may be removed for servicing.

11 CALIBRATION FACILITIES

11.1 To facilitate calibration checks on instruments in situ, two-way test valves shall not be fitted in the pipe lines. These tests can instead, be carried out using the external static and pitot sources.

12 GROUND TESTS

12.1 LEAKAGE TESTS

12.1.1 Tests shall be made on the piping system alone, by the method detailed in Leaflet 716/1, para 2, to detect any pressure drop. The test must show no pressure drop from 69kPa (10 psi) after 10 minutes.

12.1.2 Tests shall be made on the complete installation, by the method detailed in Leaflet 716/1, para 3, to measure the time for the altimeter reading to fall from 5,000 ft to 4,800 ft and the ASI reading to fall from 130kt to 125kt. In each case the time shall not be less than 3 minutes.

LEAFLET 716/0

STATIC AND PITOT PRESSURE SYSTEMS

REFERENCE PAGE

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RAE Library Bibliographies

293	..	..	Fuselage static vents

ARC Reports

CP 475		..	.. Free flight experiments on the measurement of free stream static pressure at transonic speeds with particular reference to the MK.9 pitot-static head

Defence Standards

47-21	..	..	Flexible hose assemblies for pitot and static systems in aircraft (Metric)
47-25	..	..	Pipelines and pipe couplings for aircraft fluid systems (metric)

British Standards

M23	..	..	Specification for an identification scheme for pipelines

LEAFLET 716/1

STATIC AND PITOT PRESSURE SYSTEMS

TESTS

1 INTRODUCTION

1.1 This Leaflet gives the methods of carrying out the leakage tests required by Chapter 716, para 12.

2 LEAKAGE TEST OF PIPING SYSTEM

2.1 This test should be carried out before coupling the pressure head, instruments, and other related equipment to the system.

Note: When making this test, no additional capacity, other than that of the pressure gauge, should be included in the system, as the capacity of the system affects the leakage rate.

2.2 After blanking off as necessary, a pressure of 69 kPa (gauge) should be applied to the pitot and static pipe lines in turn. After allowing about 3 minutes for adiabatic temperature change effects to settle out, measure again at 69 kPa then hold this pressure without any measurable fall for a period of 10 minutes.

3 LEAKAGE TEST OF COMPLETE INSTALLATION

3.1 This test should be carried out with the pressure head (and/or static vent), instruments, and other related equipment properly connected to the pressure lines (see also Note to para 2.1).

3.2 The pressure in the static system should be reduced until the rotorcraft altimeter (or a special test altimeter coupled into the system) reads just over 5,000 ft, the rate of pressure change not exceeding approximately 10,000 ft/min. After allowing about 3 minutes for adiabatic temperature change effects to settle out, measure the time for the altitude to drop from 5,000 ft to 4,800 ft, lightly tapping the altimeter to overcome breakout friction (stiction).

3.3 With the static system open to the atmosphere, the pressure in the pitot system should be increased until the rotorcraft ASI (or a special test ASI coupled into the system) reads just over 130 kt. After allowing about 3 minutes for adiabatic temperature change effects to settle out, measure the time for the ASI reading to fall from 130 kt to 125 kt lightly tapping the ASI to overcome stiction.

LEAFLET 716/2

STATIC AND PITOT PRESSURE SYSTEMS

AERODYNAMIC REPEATABILITY

1 INTRODUCTION

1.1 This Leaflet defines the acceptable methods of demonstrating aerodynamic repeatability of pitot-static heads and outlines the recommended procedure for the manufacture of static plates.

2 WIND TUNNEL TEST

2.1 The pitot/pitot-static head should be mounted in a wind tunnel or in front of a free jet with the tube aligned along the axis of the airflow. The airflow should have a minimum velocity of 85 kt. Both pitot and static readings (where applicable) should be taken at this airspeed. The data should be compared with that of a calibrated standard. There should be at least one such calibrated standard. Acceptance/Rejection criteria is stated in Chapter 716, para 4.

3 DESIGN GUIDELINES FOR STATIC PLATES

3.1 The static plate should consist of a plate with integral fittings to connect to the static pressure line. The plate should be manufactured to have the same surface profile as that of the rotorcraft, to within 0.01% over the linear dimensions of the plate. It should be located on an area of the rotorcraft where distortion of the plate will not occur as a result of stressing during operation, service or maintenance, in order to ensure static pressure repeatability.

3.2 Any gap between the static plate and the rotorcraft surfaces should not exceed 0.25mm. The surface finish should be no worse than 0.1μm R_a.

4 ALTERNATIVE TEST METHODS

4.1 The use of a calibrated wind tunnel for the purposes of checking repeatability may be impractical or too expensive. Alternative methods such as laser measurement or silhouette profiles should be considered, but only adopted with the agreement of the Rotorcraft Project Director.

CHAPTER 717

PROTECTION FROM THE EFFECTS OF NUCLEAR EXPLOSIONS, LASER WEAPONS, CHEMICAL AND BIOLOGICAL WARFARE AGENTS

1 INTRODUCTION

1.1 This chapter specifies the design requirements which will enable the Rotorcraft to survive in Nuclear, Biological, Chemical (NBC) and/or laser environments and their associated decontamination environments where relevant.

1.2 When there is a requirement for NBC and/or laser hardening; this chapter specifies the requirements to be applied for the protection of the Rotorcraft and crew both in the air and on the ground in an NBC/laser environment which is considered survivable.

1.3 Leaflet 717/0 provides general references associated with NBC and laser threats.

1.4 Leaflet 717/1 contains all definitions applicable to this Chapter 717 and other associated leaflets.

1.5 717/2 describes the effects of Nuclear Weapon Explosions on equipments with special reference to vulnerability reduction measures and kill categories.

1.6 Leaflet 717/3 provides information on Biological Warfare (BW) and Chemical Warfare (CW) agents, and recommendations for the reduction of vulnerability of the Rotorcraft and personnel to these threats.

1.7 Table 1 gives the Defined Threat Effects and also contains some data which may be used in a Vulnerability Analysis if more specific information is not available. The DEF STAN 00-970 Vol 1, Chapter 600, para 3 Aircraft Classes are directly applicable to aeroplanes and may also be applied to Rotorcraft to obtain probabilities of occurrence if more relevant data is not given in the Rotorcraft Specification. The directional qualifiers are applicable only to aeroplanes. For Rotorcraft the full 360° in elevation and azimuth should be considered and an appropriate set adopted for the project under consideration.

1.8 Several of the reference documents quoted in this chapter are classified and are only available on a need-to-know basis.

1.9 The attention of the author of the Rotorcraft Specification is drawn to the following:

Paras 3.2.2, 3.2.3, 3.2.4, 3.3.1, 3.3.2, 4.1.1, 4.1.3, 4.2.1, 4.2.2, 4.2.5 and Table 1.

2 GENERAL REQUIREMENTS

2.1 The Rotorcraft and its installed equipment shall be designed to be operated by personnel wearing full specified NBC and laser protective clothing. If full NBC protective clothing is not worn by the aircrew then the Rotorcraft shall be equipped with a system capable of supplying suitably pressurised and filtered air to the crew.

2.2 The design of the Rotorcraft and equipment shall be such that maintenance, replenishment and rearming tasks can be performed by personnel wearing full NBC protective equipment. (See Leaflet 717/3).

2.3 When defining the form of hardening to be adopted consideration shall be given to the following for all materials and design features incorporated in the rotorcraft and its equipment:

(i) immediate effects,

(ii) prolonged contamination,

(iii) the long-term consequences of short term contamination,

(iv) decontaminants.

3 NUCLEAR ENVIRONMENT REQUIREMENTS

3.1 GENERAL

3.1.1 The basic nuclear survivability aim for military rotorcraft weapon and ground systems, and installations shall be to comply with the application environment given in Def-Stan 08-4 (Issue 2) Part 4, Chap 4-05, Table 2.

3.1.2 The factors which shall be considered during the design phase to reduce vulnerability to the effect of nuclear weapons explosions are described in Def Stan 08-4 Part 4 and Leaflet 717/2. The effects of contamination from nuclear fallout shall be considered in the design process.

3.2 DESIGN

3.2.1 The design objective for the nuclear hardening of the Rotorcraft, its installed equipment and installed weapons, shall be compliance with the environmental criteria laid down in Def-Stan 08-4 (Part 4/2) Chap 4-05 and that associated with the nuclear threat by friendly weapons. Table 2 is typical of the parameters involved in the Nuclear Hardening design.

3.2.2 The principal design aim shall be to maximise the probability that the threat effects defined in Table 1 or in the Rotorcraft Specification will not degrade the controllability of the Rotorcraft though the mission effectiveness would be inadequate or the total workload of the pilot would approach the limit of his capacity.

3.2.3 During the Project Definition stage the Rotorcraft Designer shall undertake initial feasibility studies of the design to quantify its Nuclear survivability level in relation to the Rotorcraft Specification.

3.2.4 The results of the initial Nuclear Survivability Feasibility Studies shall be submitted to the Rotorcraft Project Director as evidence of the estimated hardness levels of the proposed design in relation to the requirements of paragraph 3.2.1, specifying any anticipated deviations from the requirements in the Rotorcraft Specification and the design changes proposed to overcome the deviations.

3.2.5 Consideration shall be given to damage to the Rotorcraft caused by its own or friendly forces weapons. It shall be assumed that essential aircrew are protected against flash-blindness.

3.3 OPERATIONAL CONDITIONS

3.3.1 The flight conditions and configuration of the Rotorcraft eg: speed, altitude, attitude, autopilot engagement, and stores configuration, immediately prior to exposure to the nuclear burst will be defined in the Rotorcraft Specification.

3.3.2 Similarly, when the rotorcraft is on the ground eg: parked in the open, sheltered, or in a Hardened Aircraft Shelter (HAS) the condition will be defined in the Rotorcraft Specification.

4 CHEMICAL AND BIOLOGICAL ENVIRONMENT REQUIREMENT

4.1 GENERAL

4.1.1 The level of chemical hardening to be embodied in the Rotorcraft, its installed equipment, associated armament and ground systems shall be determined by the environment given in the Rotorcraft Specification.

4.1.2 The properties of chemical warfare agents and decontaminants together with the factors which shall be considered during the design phase to reduce vulnerability of equipments to the effects of such agents and decontaminants are contained in CDE Technical Memorandum No 79.

4.1.3 The Rotorcraft Specification will state the requirements for survivability and operability in a Chemical and Biological warfare environment.

4.2 DESIGN

4.2.1 The design objective for the Chemical and Biological hardening of the Rotorcraft its installed equipment and weapons shall be compliance with the criteria detailed in Rotorcraft Specification. (See also Leaflet 717/3).

4.2.2 The principal design aim shall be to maximise the probability that the threat effects defined in Table 1 or in the Rotorcraft Specification shall not degrade the flying qualities of the Rotorcraft below that required to accomplish the mission flight phase, but with some degradation in mission effectiveness or increase in the workload of the pilot, or both.

4.2.3 Rotorcraft components and equipment shall be categorised as follows: (i) those items which are likely to be subjected to direct chemical attack, (ii) those items not subject to direct chemical attack (see Leaflet 713/3 para 2.2.3) but which may become contaminated due to ingestion, or the internal carriage of contaminated passengers, troops, or stores. (See Leaflet 717/3 para 2.2.4).

4.2.4 Rotorcraft components and equipment in the interior of the Rotorcraft shall not be considered to be subject to direct attack unless they can be contaminated due to ingestion of C/W agents via external apertures, air systems, etc, or due to the carriage of contaminated personnel or stores.

4.2.5 The Rotorcraft shall be capable of safe return to base after chemical contamination, and without decontamination measures being applied it shall be possible to utilise the rotorcraft for a specific number of missions involving repeated contamination or for a period defined in the Rotorcraft Specification.

4.2.6 All components and equipment subjected to direct attack shall be designed to resist the damaging effects of the specified CW agents and their decontaminants and to facilitate the rapid reduction or elimination of the hazard from any chemical contamination which may be found in the Rotorcraft or equipment.

4.2.7 All components and equipment subjected to direct attack shall be designed to allow Service personnel dressed in full Individual Protective Equipment to handle, decontaminate, or service the equipment.

4.2.8 During the Project Definition stage the Rotorcraft Project Director shall undertake initial Chemical and Biological survivability feasibility studies of the design of the structures, systems and components in relation to technical achievability and the penalties incurred in performance, reliability, maintainability, cost limitations and to the disposition of component/system within the Rotorcraft based on the specified criteria.

4.2.9 The results of the feasibility study shall be submitted to the Rotorcraft Project Director as evidence of the acceptability of the proposed design with the criteria referred to in para 4.2.8.

4.2.10 The design, structure and equipment shall prevent as far as practical the ingress of liquid chemical agents or their penetration into crevices, joints, slots. etc. Where a high degree of hardening is required hermetic sealing of equipment is preferable and a positive internal pressure is also desirable.

4.2.11 Design criteria for the environmental control system shall consider the installation of a filter or a method to completely shut off ambient air and allow any contaminated air trapped in the system to be purged prior to the subsequent use of the cockpit/cabin system. The filtration requirements for environmental control systems shall include the rejection of particles greater than 0.2 microns diameter.

4.2.12 All weather seals shall be designed to also preclude the entrance of CW or BW contaminants into not only the cabin but also other equipments. Contamination of equipment compartments would be a threat to ground maintenance personnel.

4.2.13 Suitable filtration shall be used in air cooled equipment. The filtration system shall remove liquid droplets and particulate matter down to about 1 micrometre.

4.2.14 Materials used in positions liable to contaminations shall:

(i) minimise the penetration/absorption of contaminants and thereby reducing the residual vapour hazard following decontamination.

(ii) not be damaged by liquid contaminants or solid or liquid decontamination agents to an extent that will impair the performance of the aircraft or its equipment within the limitations specified under para 4.2.6.

4.2.15 The Rotorcraft Design Authority shall demonstrate the ability of materials which are subjected to direct attack to withstand CW agents and decontaminants (see leaflet 723/3).

4.3 CHEMICAL AND BIOLOGICAL TESTING

4.3.1 The Rotorcraft Design Authority shall supply a representative sample of material proposed for use on Transparencies, Radome, Canopies and Aerials etc, measuring no more than 0.1 sq m in its finished condition to the CDE, or other Authority designated by CDE and agreed with the Rotorcraft Project Director.

5 LASER REQUIREMENTS

5.1 GENERAL

5.1.1 This section is being prepared, but in the meantime advice can be sought from DRA Farnborough.

TABLE 1

DEFINED THREAT EFFECTS

THREAT	SOURCE	FORM; ie density/dose, area exposed, power density. etc	PROBABILITY OF OCCURRENCE BY AIRCRAFT CLASS (NOTE 1)				DIRECTIONAL QUALIFIERS (NOTE 3)		GENERAL EFFECT
			I	II	III	IV	ELEVATION (°)	AZIMUTH (°)	
Directed Electro-Magnetic Energy	Laser	See Rotorcraft Specification	0.0	0.2	0.3	0.2	+0 -90	+180 -180	Degradation
Thermal	Nuclear	" " ")							Degradation
Neutron Flare	Nuclear	" " ")							Disruption
Gamma Radiation	Nuclear	" " ")	0.8	0.4	0.5	0.3			Degradation
Electromagnetic	Nuclear	" " ")							Disruption
Blast	Nuclear	" " ")							Disruption
Biological	-	Liquid, Aerosol and Vapour	0.1	0.2	0.1	0.25		Note 2 (	Incapacitation
Chemical	-	See NGASR No 562 for crew and personnel protection.	0.1	0.2	0.1	0.25		(	Incapacitation
TOTAL FOR GROUP 2 THREATS			1.0	1.0	1.0	1.0			

NOTES:
1. See Volume 1 Chapter 600 for definition.
2. See AP3395.
3. Not applicable to Rotorcraft for which the full 360° circle in elevation and azimuth should be considered.

TABLE 2 - ROTORCRAFT NUCLEAR HARDENING - TYPICAL PARAMETERS

AIR BLAST

REF	PARAMETER	SYMBOL	VALUE
1a	Incident Peak Static Overpressure	Δp	
1b	Peak Dynamic Pressure	q	
1c	Dynamic Pressure Impulse	Iq	
1d	Arrival Time	ta	
1e	Static Overpressure Duration	tp	
1f	Static Overpressure Impulse	Ip	
1g	Peak Underpressure	Δp neg	

THERMAL RADIATION

REF	PARAMETER	SYMBOL	VALUE
2a	Total Thermal Fluence	Q	
2b	Time to maximum irradiance	t max	
2c	Maximum irradiance	Q max	
2d	80% energy delivery time	S	
2e	Pulse width at half Q max	S	

INITIAL NUCLEAR RADIATION

REF	PARAMETER	SYMBOL	VALUE
3a	Total dose (tissue)	Dt	
3b	Maximum neutron contribution	Dn	
3c	Maximum gamma contribution	Dy	
3d	Maximum combined ionizing dose	Di	
3e	Maximum neutron Fluence	Fn	
3f	Peak gamma dose rate		

EMP

ENDOATMOSPHERIC EMP				EXOATMOSPHERIC EMP			
REF	PARAMETER	ELECTRIC FIELD	MAGENTIC FIELD	REF	PARAMETER	ELECTRIC FIELD	MAGENTIC FIELD
4a	Peak Amplitude A	Kv/m	A/m	5a	Peak Amplitude A	Kv/m	A/m
4b	t_1(0.9A)	ns	ns		t_1(0.9A)	ns	ns
4c	t_2(A)	ns	ns	5c	t_1(0.9A)	ns	ns
4d	t_3(0.83A)	ns	ns	5d	t_3(0.5A)	ns	ns
4e	t_4(0.65A)	ns	ns	5e	t_4(0.1A)	ns	ns
4f	t_5(0.1A)	ns	ns				

Notes:

1. The numerical values associated with Air blast, Thermal Radiation, Initial Nuclear Radiation and EMP parameters will be specified and agreed with the Rotorcraft Project Director (see paras 3.2.1 and 3.2.3).

2. The number of exposures to the above parameters will b defined by the Rotorcraft Project Director.

3. The above parameters are to be considered separately except where combinations are considered significant to the design. In this case the combinations to be considered together with relevant separation timescales will be defined by the Rotorcraft Project Director.

4. The Kill Levels to be associated with the threat values given in the above table will be that of Mission Completion (see Leaflet 717/1 para 4.21).

LEAFLET 717/0

PROTECTION FROM THE EFFECTS OF NUCLEAR EXPLOSIONS,

LASER WEAPONS, CHEMICAL AND BIOLOGICAL WARFARE AGENTS

REFERENCE PAGE

Note: See also Leaflets 717/1, 717/2, and 717/3.

NWE General

HTI-R-78-109 Horizons Technology Inc San Diego, California.	Nuclear Weapons Effects Programs; User's Manual
MSDS T(F)M805 Pilkington FB MSDS(NSG) Frimley * (Restricted)	MSDS/NS Guide 5: Designers' Guide to the Nuclear Survivability of Electro-Optical Components
RMCS Shrivenham Course Notes	Nuclear Weapon Effects Course.
NH 9006 Jan 1983 BAe Dynamics Group	Nuclear Vulnerability Handbook- A Handbook and Guide for Designers of Electronic Equipment.
WADC-TR-58-301 Wright Air Development Centre Wright-Patterson AFB Ohio Witner EA	The Effects of Atomic explosions on the Main Rotors of Helicopters in Flight

NWE Blast

MSDS/NSI/R32 MSDS(HSG)Frimley (Restricted)	The Effects of Thermal Radiation and MSDS Blast from a Nuclear Explosion on some* Glass Reinforced Plastic Components.
RMCS TN Struct, 2 RMCS Shrivenham Smith PD, Pennelegion L.	Report on the Sixth International Symposium on Military Applications of Blast Simulation.
SID 63-656 North American Aviation Inc Downey, California, USA. Hodden DT, Fowler RA.	A Study of High Altitude Nuclear Burst Data.
AFSWP-877(NAVORD 4486) Naval Ordnance Lab, White Oak, Maryland, USA. Willet JE.	The Effect of Altitude of Detonation in the First Thermal Pulse and Early Shock Pressures from an Atomic Bomb.

NWEF Report 1116 Friedberg R, Hughes PS	Intensive Vulnerability of the Heavy Lift Helicopter to Gust Environments from a Nuclear Explosion.
DNA 2910 F-1, 2, 3 Karman Avidyne Zartarian G, Thomsen J	Analysis of Helicopters Response to Nuclear Blast. Vols 1, 2 and 3.
Ballistic Research Labs APG Aberdeen USA Coyle T, Harris W	The Effect of Blast against Helicopter Rotor Blades.
Def Atomic Support Agency Washington USA WT-1430 Walls JJ, Heslin N	In-flight Structural Response of an HSS-1 Helicopter to a Nuclear Detonation.
USAAV Labs-TR-65-51 Fort Gustus, Va USA Carlson R, Helzinger K	Analysis and Correlation of Helicopter Rotor Blade Response in a Variable Flow Environment.

<u>NWE Thermal Radiation</u>

SR1 Proj IMU-4021 Stanford Research Institute, Mento Park, California, Rogers JC, Miller T.	Survey of the Thermal Threat of Nuclear weapons.
WADC TN-54-103 Wright Air Development Centre, Wright-Patterson AFB, Ohio. Feizen M Ambros.. A	Behaviour of Magnesium and Fibreglass Panels subjected to Thermal Radiation
AWRE SLF N 20/81 AWRE Aldermaston Balderston J	Thermal Tests on Military Aircraft Transparencies.
MOD(PE) MR(Nuc)2c Building 10C35 The Mearings, Burghfield Berks, RG3 3RP.	Thermal Data Book
AFWL-TR-67-85(AD383261) *(Classified)	Aircraft Thermal Vulnerability to Large High Altitude Detonations
Naval Applied Science Lab Lab Project 9400-45 Brooklyn, New York, USA Bracciaventi J	The Effects of Thermal Radiation on Coated Aircraft Skin
NOLTR 71-135 Naval Ordnance Lab White Oak, Md, USA, Christianson C	Effects of Thermal Radiation on Radome Materials

ASD-TOR-62-823 Wright-Patterson AFB Ohio, USA Aeronautical Systems Division S Gran WN	Thermal Radiation from Air Burst Nuclear Weapons Incident on Low Altitude Aircraft

<u>NWE Radiation</u>

Shell P-602 Shell Development Company Emeryville, California, USA Nixon AC, Thorpe RE, Monor HB.	Effects of Nuclear Radiation on Jet Fuels.
RMCS TN PD/28/82 and 83 RMCS Shrivenham	Effects of Radiation on Optical Fibres: CVD Contract No RV 26-4
ASD-TDR-63-893 Aeronautical Systems Div, Eglin Air Force Base, USA.	The Effects of Nuclear Radiation on Explosive Solids.
STL-GM-TR-Q165-00358 Space Technology Labs, Los Angeles, USA	Effects of Nuclear Radiation on some Materials and Electronic Components.
BAe NH 9008 iss 2 BAe Dynamics Group Bracknell Div *(Restricted)	Transient Radiation Effects on Electronic Components and Circuits - Guidelines for Avionic Engineers.
AWRE SSWL 2/76 AWRE Aldermaston * (Restricted)	A Guide to Radiation Effects on Electronic Equipment at the Tactical Level.
AFSWP-1100 Nuclear Development Corp of America.	The Nuclear Radiation Handbook.
REIC Report No 5 Radiation Effects Info Centre Ballelle Memorial Institute Columbus, Ohio USA.	The effects of Nuclear Radiation in Structural Metals.
MOD(PE) MR(Nuc)2c Building 10C35 The Mearings, Burghfield Berks RG3 3RP.	SIRE(Semiconductor Index to Radiation Effects) Data Book.

<u>NWE EMP</u>

A&AEE Note 3205 Aeroplane and Armament Experimental Establishment Boscombe Down, Salisbury,	The Procurement of Aircraft for Survival in an Electromagnetic Pulse Environment.

AFWL IN 315 KF Casey, Emtec Inc, Los Angeles California, USA.	EMP Penetration through Advanced Composite Skin Panels.
DNA 2114 H-1, H-2 and H-3 General Electric-TMPO, DASIAC, Santa Barbara, California, USA	DNA EMP (Electromagnetic Pulse) Vol 1, Vol 2, and Vol 3.

<u>Chemical and Biological Weapon Effects (C&BE)</u>
<u>General</u>

Stanag 2133 Nuclear, Biological and Chemical Operational Interservice Working Party(NBCOIWP)	Vulnerability assessment of Chemical and Biological Hazards.
AC/225(Panel VII/ASP)D/24 NAAG	Chemical Agents Removal Using Sacrificial Coatings.
AC/225(Panel VII/ASP)WP/56 NAAG	Resistance of Canopy Materials to Chemicals.
AEP 7 NATO Panel VII * (NATO Restricted)	Chemical Defence Factors in the Design of Military Equipment.
AC225(Panel VII/ASP)WP/16 NAAG	Probability of Contamination of Aircraft Cockpits During Flying or Taxying in a Chemical Environment.
AC225(panel VII/ASP)WP/28 NAAG *(NATO Restricted)	Use of Dilution Pressure Demand Oxygen Systems-Protection against CW Agents.
Chemical Research & Dev Centre US Army Armament, Munitions and Chemical Command, Aberdeen Proving Ground Maryland, USA.	NBC Contamination Survivability. A Handbook for Development/Management of Material Programs.
CRDEC-CR-87033, December 1986 Battelle Columbus Laboratories, USA Bailey P, Hill T E, McNeeley	Nuclear, Biological and Chemical Contamination Survivability Methodology. A Manual for Equipment Developers, Contractors, and Government Combat and Material Developers.

Documents marked thus * are classified and may be available only on a need-to-know basis.

LEAFLET 717/1

PROTECTION FROM THE EFFECTS OF NUCLEAR EXPLOSIONS, LASER WEAPONS, CHEMICAL AND BIOLOGICAL WARFARE AGENTS

DEFINITIONS

1 INTRODUCTION

1.1 The following definitions apply to the terms used in this Chapter 717 and in the associated Leaflets concerning the Nuclear, Laser and Chemical and Biological threats respectively.

2 DEFINITIONS - GENERAL

2.1 DEFINED THREAT EFFECTS. Those Threat Effects defined in Chapter 717 Table 1 para 1.7.

2.2 MISSION. The Task to be performed during a sortie.

2.3 MISSION - ESSENTIAL SYSTEM. A System that is essential to the successful completion of an air mission.

2.4 PROBABILITY OF OCCURRENCE (SEE CHAPTER 717 TABLE 1). A function of three factors:

(i) Probability of encountering a particular threat,

(ii) Threat lethality,

(iii) The susceptibility of the Rotorcraft to the threat.

These are combined in Chapter 717 Table 1 to give a single probability for each Rotorcraft class that indicates the importance of each type of threat.

2.5 SORTIE. An operational flight by one or more aircraft.

2.6 SPECIFIED THREAT EFFECTS. Those Threat Effects specified in the Rotorcraft Specification or by the Rotorcraft Project Director.

2.7 SYSTEM. An aggregate of hardware, software, and man that satisfies a specific end-use function.

2.8 THREAT EFFECT. The definition of a threat in terms of those physical characteristics which affect rotorcraft design.

2.9 THREATS. Those hostile elements of an environment which could reduce the ability of a rotorcraft, its systems and the crew to perform its mission.

3 DEFINITIONS - CHEMICAL AND BIOLOGICAL THREATS

3.1 BALANCED CHEMICAL HARDENING. A concept of system chemical hardening in which all sub-systems have been brought to a commensurate hardness level in the sense that no one link (sub system) in the system chain is overhardened, nor is any link in the chain of significantly greater vulnerability than the rest of the sub-systems.

3.2 BIOLOGICAL AGENT. A micro-organism which causes disease in man, plants, or animals or causes the deterioration of material.

3.3 BLISTER AGENT. A chemical agent which injures the eyes and lungs, and burns or blisters the skin.

3.4 BLOOD AGENT. A chemical compound including the cyanide group, that affects bodily functions by preventing the normal transfer of oxygen from the blood to body tissues.

3.5 CHEMICAL AGENT. A chemical substance which is intended for use in military operations to kill, seriously injure, incapacitate man through its physiological effects. Excluded from consideration are riot control agents, herbicides, smoke and flame.

3.6 CHEMICAL HARDENING. Measures taken during the design and construction of military equipments to avoid damage to materials caused by CW agents and to reduce or eliminate the hazard to personnel arising from the presence of chemically contaminated surfaces.

3.7 CHEMICAL OPERATIONS. Employment of chemical agents to kill injure, or incapacitate for a significant period of time, man or animals, and deny or hinder the use of areas, facilities or material, or defence against such employment.

3.8 CHEMICAL SURVEY. The directed effort to determine the nature and degree of chemical hazard in an area and to delineate the perimeter of the hazard area.

3.9 CHEMICAL SURVIVABILITY. The capability of a system to withstand a severe level of hostile chemical warfare environment(s) without suffering loss of its ability to accomplish the designated mission(s).

3.10 CHEMICAL VULNERABILITY. The characteristics of a system which cause it to suffer degradation (ie the capability to perform its designed mission) as a result of having been subjected to a given level of a hostile chemical warfare environment.

3.11 CONTAMINATION. The deposit and/or absorption of radioactive material, biological or chemical agents on and by structures, areas, personnel, or objects. See also induced radiation; residual radiation.

3.12 CONTAMINATION CONTROL. Procedures to avoid, reduce, remove, render harmless, temporarily or permanently, nuclear, biological, and chemical contamination for the purpose of maintaining or enhancing the efficient conduct of military operations.

3.13 CONTAMINATION DENSITY. The amount liquid or solid chemical contamination per unit area of surface; usually expressed in g m^{-2}.

3.14 DECONTAMINANTS. Chemicals employed for the purpose of promoting the absorption, dissolution, dilution or destruction of chemical or biological warfare agents.

3.15 DECONTAMINATION. The process of making any person, object, or area safe by absorbing, destroying, neutralising, making harmless, or removing, chemical or biological agents, or by removing radioactive material clinging to or around it.

3.16 DESORPTION TIME. The time taken for the concentration of chemical agent vapour evolved from a contaminated surface to decrease to a safe level.

3.17 FILTRATION. The process of actively reducing chemical agent solid particulate, liquid, aerosol or vapour concentrations in the atmosphere by the passage of the contaminated air through a suitably absorbent material.

3.18 HERBICIDE. A chemical compound which will kill or damage plants.

3.19 NERVE AGENT. A lethal agent which causes paralysis by interfering with the transmission of nerve impulses.

3.20 PENETRATION TIME. The time taken for a CW agent vapour or liquid to penetrate through a specified thickness of a sheet material to produce a hazardous condition usually expressed as a vapour concentration, on the other side.

3.21 PERMEABILITY. A measure of the susceptibility of solid materials to penetration by CW agent liquids.

3.22 PERSISTENCY. An expression of the duration of effectiveness of a chemical agent which is dependent upon the physical properties of the agent, the method of dissemination, the weather conditions and the characteristics of the contaminated surfaces.

3.23 POLYMERIC MATERIAL. The general description of all types of polymers including paints, thermoplastics, thermosets, elastomers, rubbers and transparent coatings.

3.24 VESICANT AGENT. See Blister Agent.

4 DEFINITIONS - NUCLEAR THREAT

4.1 AIR BURST. The explosion of a nuclear weapon in the air, at a height greater than the maximum radius of the fireball.

4.2 BALANCED NUCLEAR HARDENING. A concept of a system nuclear hardening in which all sub-systems have been brought to a commensurate hardness level in the sense that no one link (sub-system) in the system chain is overhardened, nor is any link in the chain of significantly greater vulnerability than the rest of the sub-system.

4.3 BALANCED SURVIVABILITY. Survivability is balanced when the following conditions are satisfied:

(i) all sub-systems and components have approximately equal survivability for each specified nuclear environment.

(ii) the entire system is not vulnerable to one or more environmental effects, while having adequate survivability for all other associated effects.

4.4 ELECTRICAL ISOLATION. Separation of electrical circuits, signals, or data to preclude ambiguity, interference, or information perversion. This may be achieved through physical isolation or by any property which distinguishes one electrical signal from all others (for example, time, phase, amplitude, or frequency).

4.5 ELECTROEXPLOSIVE DEVICE (EED). Any electrically initiated explosive device within an electroexplosive sub-system having an explosive or pyrotechnic output, and actuated by an electroexplosive initiator.

4.6 ELECTROMAGNETIC COMPATIBILITY (EMC). The ability of electronic equipment sub-systems and systems to operate in their intended operational environments without suffering or causing unacceptable degradation because of unintentional electromagnetic radiation or response.

4.7 ELECTROMAGNETIC INTERFERENCE (EMI). Any electromagnetic disturbance which interrupts, obstructs, or otherwise degrades or limits the effective performance of electronics/electrical equipment. It can be induced unintentionally, as a result of spurious emissions and responses, intermodulation produces, and the like.

4.8 ELECTROMAGNETIC PULSE (EMP). A sharp pulse of radio frequency (EMP) electromagnetic radiation produced when an explosion occurs in an unsymmetrical environment especially at or near the earth's surface or at high altitude.

4.9 ELECTROMAGNETIC RADIATION (EMR). Radiation made up of oscillating electric and magnetic fields and propagated with the speed of light. Includes gamma radiation, X rays, ultraviolet, visible and infra red radiation, and radar and radio waves.

4.10 ENDO ATMOSPHERIC. Within the atmosphere, altitude less than 35km.

4.11 EXO ATMOSPHERIC. Outside the atmosphere, altitude greater than 35km.

4.12 FAIL-SAFE. A design feature of a nuclear weapon system/component which ensures that, under failure, no critical function, damage to equipment, or injury to personnel will occur.

4.13 FIREBALL. The luminous sphere of hot gases which forms a few millionths of a second after a nuclear explosion. The fireball is a result of the absorption by the surrounding medium of thermal X-Rays emitted by the extremely hot weapon residues which are at several tens of million degrees.

4.14 FISSION. The process whereby the nucleus of a particular heavy element splits into (generally) two nuclei of lighter elements, with the release of substantial amounts of energy. The most important fissionable materials are uranium-235 and plutonium-239; fission is caused by the absorption of neutrons.

4.15 FLUENCE. Fluence is the time integrated flux, ie the number of particles per unit area.

4.16 FLUX. Flux is the number of particles (photons) per unit area per second.

4.17 FUSION. The process whereby the nuclei of light elements, especially those of the isotopes of hydrogen, namely deuterium and tritium, combine to form the nucleus of a heavier element, with the release of substantial amounts of energy.

4.18 GAMMA RADIATION (SYMBOL γ). High frequency electromagnetic radiation with a very short wavelength (10^{-11} to 10^{-14}m) emitted from atomic nuclei, and accompanying many nuclear reactions. Physically gamma rays are identical with X-Rays of high energy, the essential difference being that X-Rays do not originate from atomic nuclei, but are produced in other ways, like slowing down fast electrons of light energy.

4.19 GROUND BURST. The explosion of nuclear weapon at the surface of the earth or at a height above the surface less than the radius of the fireball at maximum luminosity (in the second thermal pulse).

4.20 INITIAL RADIATION. Radiation produced by a nuclear explosion within 1 minute following the burst. It includes neutrons and gamma rays given off at the instant of the explosion, gamma rays produced by the interaction of neutrons with weapon components and the surrounding medium, and the alpha, beta, and gamma rays emitted in the fission products and other weapon debris during the first minute following the burst.

4.21 MISSION COMPLETION. Mission Completion kill category is the level of response of an aircraft corresponding to a damage level between incipient and catastrophic damage. The rotorcraft should be just able to accomplish its assignment satisfactorily.

4.22 NEUTRON (SYMBOL η). A neutral particle with no electrical charge. Its mass is 1.00867 atomic mass units or 1.67492×10^{-27}Kg. It is present in all atomic nuclei except those of ordinary (light) hydrogen. Neutrons are required to initiate the fission process, and large numbers of neutrons are produced by both fission and fusion reactions in nuclear (or atomic) explosions.

4.23 NUCLEAR FALL-OUT. The settling of the radioactive debris resulting from an atomic explosion.

4.24 NUCLEAR HARDENING. Nuclear hardening is the employment of any technique which circumvents or mitigates the effects of an adverse nuclear environment, that is, which improves nuclear survivability.

4.25 NUCLEAR RADIATION. Particulate and electromagnetic radiation from atomic nuclei in various nuclear processes. The important nuclear radiations, from the weapons standpoint, are alpha and beta particles, gamma rays and neutrons. All nuclear radiations are ionizing radiations, but the reverse is not true; X-rays for example, are included among ionizing radiations, but they are not nuclear radiations since they do not originate from atomic nuclei.

4.26 NUCLEAR SURVIVABILITY. The capability of a system to withstand a nuclear environment without suffering loss of its ability to accomplish the designated mission.

4.27 NUCLEAR SURVIVABILITY CRITERIA. The specified nuclear environmental level used to define the nuclear survivability required of a given system.

4.28 NUCLEAR VULNERABILITY. The characteristics of systems which cause it to suffer degradation (ie in the capability to perform successfully its designated mission) as a result of having been subjected to a given level of a hostile nuclear environment.

4.29 RADIATION DOSE. The total amount of ionising radiation absorbed by material or tissue, commonly expressed in Grays (or sub-multiples thereof) and indicating the absorbing material, e.g. cGy(si).

4.30 RESIDUAL RADIATION. Nuclear radiation, chiefly beta particles and gamma rays, which persists for some time following a nuclear explosion.

4.31 SLANT RANGE. The direct distance between an explosion and a target, as opposed to the horizontal distance between ground zero and a target which is the ground or surface range.

4.32 SURE-KILL. Sure-kill is the level of response corresponding to a catastrophic damage condition which results in essentially immediate loss of the aircraft.

4.33 SURE-SAFE. Sure-safe is that level of response which in no way affects mission completion and allows the rotorcraft to return home or to an alternate base in an essentially undamaged condition.

5 DEFINITIONS - LASER THREAT

TBA

LEAFLET 717/2

PROTECTION FROM THE EFFECTS OF NUCLEAR EXPLOSIONS, LASER WEAPONS, CHEMICAL AND BIOLOGICAL WARFARE AGENTS

NUCLEAR WEAPON EFFECTS ON ROTORCRAFT

1 INTRODUCTION

1.1 This leaflet gives information on the effects of a nuclear explosion on rotorcraft and provides methods which may be used to evaluate the hardness of the Rotorcraft and avionic systems and guidance on how this hardness may be improved. For a detailed description of the characteristics of a nuclear explosion see Reference 1 ('Handbook for Analysis of Nuclear Weapons Effects on Aircraft', Volume 1, Section II), and/or Def-Stan 08-04 Part 1, Nuclear Weapons Explosions. Definition of the terms used are given in Leaflet 717/1.

1.2 The limitations on the capabilities of rotorcraft to deliver conventional stores are imposed by such factors as range and payload and enemy defensive actions. With the emergence of nuclear weapons the enhanced nuclear yield has placed the Rotorcraft in danger of being damaged by the weapon it has delivered. Additionally consideration of the effects of the nuclear explosion on parked rotorcraft and also the conditions under which rotorcraft may be shot down or killed by the effects of nuclear weapons needs to be given. It is necessary therefore to construct sure-safe, mission completion, and sure-kill envelopes for the Rotorcraft under consideration.

1.3 The effects of the weapon characteristics on the Rotorcraft or its crew include velocity (gust) effects, overpressure effects, thermal radiation effects, initial radiation effects, residual radiation effects and combinations of these features.

2 NUCLEAR WEAPON EFFECTS ON AIRCRAFT

2.1 GUST EFFECTS

2.1.1 As a Rotorcraft is engulfed by a blast wave it encounters changes in material velocity, pressure, temperature and density. Hence angle of attack, sideslip, dynamic pressure and Mach number. These changes cause changes in the aerodynamic loads acting on the Rotorcraft.

2.1.2 If the Rotorcraft is intercepted by a blast wave from below, then eventually a translational velocity will ensue as the Rotorcraft responds to the gust. Ultimately the increment in loading will reduce below the initial value caused by the encounter. The highest loadings are likely to occur soon after the interception in a very dynamic manner that depends strongly on the characteristics of the Rotorcraft and flight conditions; "riding with the gust" may limit the duration of the loading.

2.1.3 In general the change in aerodynamic conditions during an encounter with a blast wave will cause moments on the Rotorcraft which will result in angular accelerations. A rotorcraft will rotate into a gust once it is fully immersed, but the initial response may be different and will depend, for example, on the direction of the blast. This initial response may be particularly marked where a rotorcraft is overtaken by a blast wave, where there may be a significant time difference between for example the change of lift on the rotor and on the pitch control surfaces.

2.1.4 Blast-induced gust analysis of a helicopter fitted with a hinged or articulated rotor indicates that flapping motions and flapwise bending stresses are mechanisms which define the critical gust envelopes for sure-safe conditions. Extreme flapping could result in the blades hitting the flapping limits or stops, and in combination with elastic deformations resulting from flapwise bending, could result in a collision between a blade and the fuselage. In the case of the Rotorcraft using rigid or hingeless blades bending is important such that the maximum total tip deflection in the absence of flapping can be estimated from the results of computational steps for the maximum bending moments. A teetering rotor experiences rapid body flapping under an asymmetric loading and acts like a rigid rotor and bends rather than flaps under symmetric loading. Insomuch as the loadings on such rotors are generally asymmetric, flapping and flapwise bending are present.

2.1.5 Severe loadings are required in order to produce damage corresponding to a sure-kill condition and these loadings can generally occur only if the gust induced angle of attack or sideslip is large and often well beyond the angle for which linearity can reasonably be assessed. However, if the Rotorcraft is manoeuvring when it encounters a blast wave, the additional load needed to produce damage will be reduced.

2.1.6 The change of aerodynamic loading will in general cause dynamic structural deformations of the fuselage and aerodynamic surfaces, which will modify the loading and response of the Rotorcraft because of the associated local velocities and displacements.

2.1.7 As the deformation of a structure increases beyond the onset of buckling its load carrying capability decreases and may become sufficiently low that a sure-kill condition exists.

2.1.8 Although it is unlikely that the autopilot would be in engagement under 'battlefield' conditions, it should be noted that an autopilot which is maintaining constant barometric altitude could react violently to the change in pressure accompanying the blast wave. An autopilot would normally disengage automatically under such circumstances but the response of active control systems and automatic terrain following systems should be considered. Consideration should also be given to the gust effects on engaged stability augmentation systems (SAS) with due regard to the height hold facility, should this be selected under "battlefield" conditions.

2.1.9 Parked rotorcraft may also be damaged by bending of the fuselage or tail pylon due to aerodynamic loading of the tail pylon. For sure-safe conditions, this bending will be elastic; for sure-kill conditions, inelastic response may also be important.

2.2 OVERPRESSURE EFFECTS

2.2.1 Overpressure influences smaller elements of the structure such as the skin, the stringers, and the frames, particularly on the fuselage. When a rotorcraft is struck by a blast wave, the pressure on the side of the fuselage facing the burst point is increased above the incident value by reflection, and a local loading of short duration is generated. As the blast wave continues to engulf the Rotorcraft, the pressure on the side of the fuselage facing the burst point decays to the pressure behind the blast wave. The characteristic loading, then, is a high reflected pressure (from two to eight times the overpressure associated with the blast wave) which decays very rapidly, in a few milliseconds, to the value of the pressure behind the blast wave. This high pressure short-duration pulse is then followed essentially by the much longer duration, but lower pressure, pulse characteristics of the blast wave.

2.2.2 It is primarily the high reflected pressure, short-duration pulse which is responsible for damage to skin panels, stringers and frames. These structural elements are vulnerable to such short-duration loadings because of their high frequencies. For the converse reason, the much lower frequency major components are influenced very little by the short-duration loading.

2.2.3 The short duration pulse produces dishing-in of skin panels and buckling of stringers and frames or portions of stringers and frames. As in the case of analysis of gust effects, analysis of overpressure response need consider only elastic response for the sure-safe case, but should properly include inelastic response for sure-kill conditions.

2.2.4 The distinguishing feature of the Rotorcraft, the rotor blade, is virtually invulnerable to overpressure effects hence, the Rotorcraft is no different from an aeroplane in this regard.

2.2.5 Overpressure damage is generally the predominant effect for parked rotorcraft. For inflight rotorcraft however, overpressure damage is usually of equal importance in comparison with gust effects.

2.3 THERMAL RADIATION EFFECTS

2.3.1 The radiant exposure of a rotorcraft in-flight varies widely with atmospheric conditions, orientation of the Rotorcraft with respect to the burst, rotorcraft velocity, the ground reflecting surfaces, and clouds. Reflection adds and scatter may add to the direct radiation, and under some circumstances the thermal energy incident on a rotorcraft in space may be two to three times the direct thermal energy computed at a given slant range. Conversely, when a heavy cloud layer is between the burst and the Rotorcraft, the radiant exposure may be only a fraction of the predicted value for a given range. In other situations, reflected radiation from clouds may contribute significant thermal energy to areas of the aeroplane shaded from direct radiation. During weapon effects tests of an aeroplane flying in a cloud above the burst, the radiant exposure at the top of the aeroplane and its cockpit area was observed to be as much as one-fourth of the direct radiation on the lower surfaces. This experiment demonstrated the need for protection of weapon delivery rotorcraft from radiant exposure from any direction.

2.3.2 The motion of the Rotorcraft during the time in which significant thermal radiation is being emitted by the fireball can exert a very important influence on the thermal radiation incident upon the rotorcraft.

2.3.3 The absorptivity of the Rotorcraft metal skin and the angle of incidence of the thermal radiation affect the amount of energy absorbed by the structure; the boundary layer in the air flow adjacent to the structure leads to convective cooling. Very thin skins are rapidly heated in damaging temperatures, because the energy is absorbed by the skin much more rapidly than it can be dissipated by conduction and convective cooling. The reduction of rotorcraft vulnerability to thermal radiation may be achieved by coating materials with low absorptivity paints, by eliminating ignitable materials from exposed surfaces, and by substitution of thicker skins for very thin skins.

2.3.4 An irradiated rotorcraft thin metal skin panel, supported by internal structure which is usually much cooler, may be heated to a point where it may be badly buckled by thermal stresses or melted. For a thin skin in either case, there will be essentially no temperature variation through the thickness. A step higher in complexity is the thick skin case which involves a temperature distribution across the thickness of the skin. A still more complex temperature distribution occurs in built-up structures, with air gaps acting as insulators between spars, stringers, and skin.

2.3.5 Thin skins of Carbon Fibre Composite (CFC) structure may be subject to surface damage and delamination because of the high thermal gradient produced through the skin due to the low thermal conductivity of CFC. Thermal stressing may not be a problem due to the low thermal expansion of CFC.

2.3.6 Analyses of thermal radiation effects on rotorcraft may be based upon temperature rises, thermal stresses, or melting of exposed structural material. In general, the sure-safe envelopes are based upon thermal stress for single skin panels and upon temperature rise for sandwich panels. For sandwich panels and composites the temperature of the bonding agent is critical; hence, the use of a temperature rise as the criterion is reasonable. For sure-kill, skin melting is required, any lesser response being considered to be sub-lethal.

2.3.7 Biological injury to the crew from intense thermal radiation and damage to nonstructural elements which would adversely affect mission performance should also be considered when dealing with thermal criteria. In many cases, these problems can be minimized by adequate protective measures such as reflective coating applied to transparencies.

2.3.8 Rotorcraft inadequately hardened may sustain permanent damage at very low thermal levels as a result of ignition of items such as rubber, plastic items used with the cockpit and/or fabric seals, fixed landing gear rotorcraft tyres, flotation equipment, cushion and headrest covers.

2.3.9 Rotorcraft painted with dark paint are especially vulnerable to thermal radiation damage because the dark painted surfaces absorb three to four times the thermal energy that is absorbed by polished aluminium surfaces or surfaces protected with reflective paint.

2.3.10 For small yields, thermal radiation is generally of secondary importance for both parked and inflight rotorcraft; thermal radiation may be dominant for both for high-yield weapons.

2.3.11 Sections 4, 5, and 6 of Ref 3 covers Material Properties relevant to the Thermal Radiation Environment, Temperature Rise and Distribution, and Thermal Radiation Effects respectively, and constitute a 'designers guide' relative to these aspects.

2.3.12 The smoke and toxic emission characteristics should be assessed to minimise their effects on cockpit/cabin occupants.

2.4 NUCLEAR RADIATION EFFECTS

2.4.1 The vulnerability of rotorcraft to nuclear radiation effects depends upon the effect of nuclear radiation on the crew, electronic gear, special weapons, or instruments which may be carried by the rotorcraft. The important radiation consists of gamma rays and neutrons emitted during a brief period after the nuclear explosion; both forms of radiation travelling significant distances through air capable of producing harmful effects in living organisms and electronic components.

2.4.2 These forms of initials gamma and neutron radiation (ie radiation which occurs within one minute after the burst) can have deleterious effects on the electrical and electronic system of the Rotorcraft such that a large enough dose of initial nuclear radiation can incapacitate a rotorcraft. Additionally, initial radiation can considerably reduce the effectiveness of a nuclear weapon carried by the Rotorcraft, and can have injurious effects on rotorcraft components such as plastics, rubber, fuels and lubricants.

2.4.3 The effect of the gamma radiation is to cause ionisation, energy being transferred to the electrons of an atom by photon scattering and the electrons then losing energy by further ionisation. The excess carriers produced can lead to charge transfer, increased conductivity, photocurrents and trapped charges. The primary effect of neutron irradiation occurs in some conductor materials, defects being formed by movement of atoms. Gamma and neutron radiation effects are generally considered under the heading TREE (Transient Radiation Effects on Electronics).

2.4.4 Transient malfunctions caused by gamma radiation are less likely to be serious in analog systems than in digital systems, where data in a memory may be corrupted and one-shot devices prematurely operated. The reverse is true for neutron irradiation, analog circuits being more susceptible than digital circuits due to reductions in transistor gain.

2.4.5 EMP can damage or cause malfunction in rotorcraft electric circuits, cables and electronic components. EMP fields couple to aircraft by direct penetration through electrically poor joints in the Rotorcraft skin, diffusion through CFC, aerials, and through ports such as transparencies and hatches. The impact of EMP upon a rotorcraft is a system dependent phenomenon and each rotorcraft electrical and electronic system must be assessed independently.

2.5 RESIDUAL RADIATION EFFECTS

2.5.1 Residual radiation can be a problem to both equipment and personnel. Rotorcraft which will traverse a nuclear dust cloud will receive an immersion dose due to the decay of the radioactive material in the cloud. In addition, if air outside the Rotorcraft is taken into the Rotorcraft to cool electronics (or for other purposes) then the equipment inside the Rotorcraft may receive a cockpit/cabin dose as well.

2.5.2 Fallout which has been deposited on an airbase will contribute to the radiation dose received by the Rotorcraft while they are parked. The response of the equipment to the residual gamma radiation will be the same as for the initial radiation and dependent upon the total dose received.

2.5.3 As with equipment, residual radiation can be a concern to aircrew members during flight or ground operations, however it is unlikely that dose to the aircrew will be sufficient to have an impact on the success of the mission. The crew will require an air filtration unit to remove particulate radioactive matter when flying through nuclear debris clouds or ground manoeuvring in a fallout region.

3 NUCLEAR HARDNESS EVALUATION OF AIRFRAME AND EQUIPMENT

3.1 The nuclear weapon effects considered in this paragraph are airblast, thermal pulse, initial nuclear radiation, electromagnetic pulse (EMP). The various weapon effects are those immediately external to the Rotorcraft, equipment, or personnel. The degree of hardening to reduce nuclear vulnerability with respect to specific weapon yields will be a compromise between possible cost and performance penalties on the one hand and improved serviceability on the other. In calculating the hardness of a rotorcraft structure it is necessary to determine and compare the hardness for each nuclear weapon effect ie airblast, initial radiation, thermal radiation and EMP effects.

3.2 EVALUATION OF AIRBLAST EFFECTS

3.2.1 When a rotorcraft is parked or flying in the vicinity of a nuclear blast it may be subjected to both dynamic and static overpressure effects. The dynamic effects result in gusts which in turn apply loads to rotors, stabilisers, and vertical tail pylons.

3.2.2 The gust effects on the Rotorcraft should be considered for each of the following categories:

(i) the rotor system.
(ii) major components other than the rotor system.
(iii) effects on the Rotorcraft as a whole.

3.2.3 The gust effects analysis for rotor systems should assess:

Blade Flapping - to determine whether the blades deflect sufficiently to make contact with the flapping stops.

Blade Elastic Deformations - to determine whether (flapwise bending) deformations, together with the rigid body flapping motion, if relevant, result in sufficient total deflections so as to cause the blades to contact the fuselage thus jeopardizing the safety of the rotorcraft and

Blade Flapwise Bending Moment Distribution of each blade in the system to establish whether allowable bending moments are exceeded at points along the span.

3.2.4 Although the gust effects have been categorised in Para 3.2.2. above, the possibility of coupling between the effects should not be overlooked, eg the vertical translation of the Rotorcraft affects the main rotor blade loadings, the flexibility of the tail boom interacts with the loadings on the tail rotor etc.

3.2.5 GUST ANALYSIS - IN-FLIGHT Ref 1 Appendix B, Sections B3.1 and B3.2 illustrates two methods of analysis:

(i) The analysis given in B3.1 is based upon determining the load factor produced on the Rotorcraft during the blast encounter, accounting roughly for the fact that this incremental load factor is dynamically applied, and comparing the resultant effective load factor with the critical load factor. The parameters associated with this analysis are flight altitude, weapon yield, AUW at time of interest, pre-blast rotorcraft velocity, main rotor blade angular velocity, main rotor blade radius and chord, number of blades in main rotor, distance from centre of rotation to station beyond which the blade is aerodynamically effective, Rotorcraft pre-blast load factor, Rotorcraft positive and negative load factors, wing platform (if relevant), and ground attitude.

(ii) An alternative method which is more involved is given in B3.2 and comprises Part A covering the gust analysis of rotorcraft blades and Part B associated with major components other than blades. Part A - The analysis is based upon determining the maximum flapwise bending moments subsequent to blast encounter at selected blade spanwise stations and comparing these bending moments with the corresponding critical values. The incremental static bending moments induced by the blast are computed assuming a steady gust velocity equal to maximum velocity appropriate to the weapon yield and range. These incremental static bending moments are then multiplied by a dynamic factor to estimate the maximum incremental bending moments that occur as a result of the dynamic applied loads. For sure-kill the critical bending moments are the products of the design ultimate bending moments and a lethal ratio.

Part B - The analysis is based upon determining the bending moments produced by the blast encounter at each of several positions of the Rotorcraft and comparing each of these bending moments with the corresponding critical value. Each bending moment is determined by first calculating the static bending moment, ie the bending moment produced by the combination of the airloads and the resulting inertia forces due to rigid-body translation, if applicable. The static bending moment is then multiplied by an approximate dynamic factor in order to estimate the maximum bending moment that will occur. The dynamic factor is a composite factor which represents the alleviation of the loads due to rigid-body translation (riding with the gust) and the overstress due to structural dynamic effects.

For sure-safe, the critical bending moments are based upon design limit conditions. For sure-kill, the critical bending moments are based upon design ultimate conditions and a lethal ratio. The lethal ratio is determined from a single representation of post-failure response by a single degree of freedom system.

Apart from the specific geometry of the Rotorcraft under consideration, the following parameters are required for the analysis; Rotorcraft altitude, weapon yield, Rotorcraft gross weight at time of encounter, pre-blast Rotorcraft velocity, main rotor angular velocity, main rotor blade radius and chord, number of blades in main rotor, tail rotor angular velocity, tail rotor blade radius, chord and number of blades, rotorcraft pre-blast-load factor, pre-blast lift on each main rotor blade, pre-blast lift on wings, fundamental wing bending frequency, fundamental fuselage vertical and horizontal bending frequency, fundamental tail vertical and horizontal bending frequency, and ground altitude.

3.2.6 GUST ANALYSIS - PARKED ROTORCRAFT

(i) Reference 1 - Appendix B, Section B4 illustrates possible methods of analysis applicable to aeroplanes but the same procedures will apply for rotorcraft. In the parked mode the Rotorcraft will be subjected to gust effects on the vertical tail and fuselage, induced lift-off, and in the case of tail-on gusts, crushing of the landing gear. In the absence of large lifting surfaces, the lift-off, and the landing gear crushing problems are much less severe than for aeroplanes.

Part A of Reference 1 Appendix B, Section B4 illustrates an analysis method for estimating the gust effects on the Vertical Tail and Rear Fuselage for Parked Aeroplanes and Rotorcraft. The analysis is based upon determining the bending moment produced by the blast encounter at two positions on the Rotorcraft and comparing each of the bending moments with the corresponding critical value.

Each bending moment produced by the static application of the airloads is multiplied by an approximate dynamic factor in order to estimate the maximum bending moment actually occurring. Critical bending moments are required for sure-safe and sure-kill conditions.

3.2.7 OVERPRESSURE ANALYSIS Overpressure effects primarily influence secondary structural elements such as, panels, stiffness, frames, canopies, radomes, etc., in contrast to gust effects which influence deformations of primary structural components such as wings, fuselage, and horizontal/vertical tail units. Skin panels may yield or rupture; longerons, stringers and frames may fail by comprehensive yielding or local buckling. The fuselage is generally the most susceptible to these types of damage and hence the overpressure analysis is more relevant to this area of the rotorcraft. Ref 1, Chapter C1 gives methods of analysis for assessing the overpressure effects in flight or parked on fuselages and a more sophisticated method is also referenced with no restriction to fuselage structures.

3.3 EVALUATION OF THERMAL RADIATION EFFECTS

3.3.1 The response of the Rotorcraft to the incident thermal energy exhibits itself as a temperature rise in the rotorcraft skin. Several parameters influence the magnitude of this temperature rise; the most important being skin thickness, material, surface condition, cooling air flow over skin surface etc; radiation of thermal energy to the atmosphere and conduction of nuclear energy to the atmosphere, and conduction of the incident energy to the inner layers of the skin and substructure.

3.3.2 Sure-safe conditions are based on an allowable temperature rise of the Rotorcraft skin. Melting of the Rotorcraft skin is a requirement for a sure-kill situation. In this case, the temperature rise of the skin is followed to the melt temperature, and further temperature and further heat input is necessary to produce melting. Typical calculations of the envelope that defines the sure-safe and sure-kill regions with respect to thermal radiation on Rotorcraft in-flight or parked are given in Reference 2 (Capabilities of Nuclear Weapons Part II Chapter 13 Page 60).

3.3.3 For sure-safe the criteria is assumed to be the skin panel temperature value which produces a 20% reduction in the Modulus of Elasticity when applied to the thinnest structural skin on the fuselage. For each burst orientation, skin panels located in the following regions should be considered:

(i) for a burst directly below the Rotorcraft, the lower surface of the fuselage within 45° of the normal to the bottom of the fuselage,

(ii) in a burst directly above the Rotorcraft, the upper surface of the fuselage within 45° of the normal to the top of the fuselage,

(iii) in a burst directly to the side of the Rotorcraft, the side surface of the fuselage not covered by (i) and (ii) above.

Rotorcraft improperly prepared may sustain serious damage at very low thermal levels as a result of ignition of items such as rubber and/or fabric seals, fixed landing gear tyres, cushions and headrest covers etc. Rotorcraft painted with dark paint are especially vulnerable to thermal radiation damage, because the dark painted surfaces absorb three to four times the thermal energy that is absorbed by polished aluminium surfaces protected with reflective paint.

3.3.4 Reference 1 - Appendix D Page D1.2-1 illustrates an alternative method of analysis for estimating thermal radiation effects on rotorcraft. For sure-safe, single skin panel, this effect is raising the temperature of the skin to a value which produces thermal strains in the skin, combined with any internal pressurisation, cause incipient yielding of the skin panel. This criterion is applied to the thinnest structural skin on the fuselage.

For sure-safe sandwich panels, the specified effect is deterioration or melting of the bond of the sandwich panel on the fuselage.

For sure-kill the specified effect is the melting of the thickest skin on the fuselage.

Because the thermal energy absorbed is a complex function of the range, the solution is an iterative process based upon varying the range until absorbed thermal energy becomes equal to the critical amount of heat required to produce the specified effect.

3.3.5 Reference 3 Section 7.1. (Thermal Data Book D/DP(N)21/5/17) describes and quantifies the thermal radiation effect for various classes of materials based on the results of both atmospheric weapon tests and simulation trials.

3.4 INITIAL RADIATION

3.4.1 Ref 1 (Handbook for Analysis of Nuclear Weapon Effects on Aircraft Vol 2), Appendix E, Paragraph E1.1 provides a method of analysis for the determination of sure-safe and/or sure-kill nuclear radiation envelopes based on specific gamma dose rate criteria (applicable to avionics in general).

3.4.2 Procedures to quantify the gamma and X-ray radiation effects on semi-conductors is contained in Section 3 of Reference 5 which covers both discrete devices and integrated circuits.

3.5 EMP

3.5.1 Degradation of system performance may occur as a result of functional damage or operational upset in which its performance is only impaired temporarily. Electronic components that are sensitive to functional damage or burnout are given in Ref 2, (Capabilities of Nuclear Weapons Part II), Part II, Chap 9, Para 58.

3.5.2 The nature of a circuit has a strong bearing on the transients that cause damage, however, in general, pulse lengths of microsecond and submicrosecond duration are required to cause problems. Table 9-27 of Ref 2 Part II Chapter 9 Para 59 shows a list typical of common active devices and the approximate energy required to cause functional damage. The minimum energy required to damage meters or ignite fuel vapours is about the same as that required to damage semiconductors. The energy level associated with operational upset is typically 10 to 100 times less than that which is required to damage sensitive semiconductor components.

3.5.3 A general approach to the analysis of a system with regard to its EMP vulnerability should include the following steps:

(i) identify susceptible subsystems and components,

(ii) determine the level of energy which will cause damage,

(iii) estimate exposure to coupled EMP energy,

(iv) estimate protection which may be provided by such devices as filters and clamps,

(v) estimate level of damage or upset and the effect on the ability of the system to carry out its function.

3.5.4 Information on the Coupling Mechanisms into Equipment, Electrically Short Cables in Incident EMP Fields, Coupling to Long Cables and Lines, and Aerials and Pseudo-aerials is given in Paras 5.2, 5.3, 5.4, and 5.5 of Ref 4).

3.5.5 The susceptibility of equipment to EMP energy is largely dependent on mechanical construction and the response to conducted and radiated electro-magnetic energy. Methods of overcoming some of the problems are dealt with in Sections 6 and 7 of Reference 5.

REFERENCES

1. Handbook for Analysis of Nuclear Weapons Effects on Aircraft, Vol I and II, DRIC Acc No P 243939 and 243940.

2. Capabilities of Nuclear Weapons (U) DNA EM-1(N) Part II, DRIC Acc No P205160.

3. Thermal Data Book D/DP(N)21/5/17.

4. Def Stan 08-4 (Part 4) Iss 2.

5. Nuclear Vulnerability Handbook - A Handbook and Guide for Designers of Electronic Equipment NH 9006 BAe Dynamics Group.

6. SIRE (Semiconductor Index to Radiation Effects) Data Book.

LEAFLET 717/3

PROTECTION FROM THE EFFECTS OF NUCLEAR EXPLOSIONS, LASER WEAPONS, CHEMICAL AND BIOLOGICAL WARFARE AGENTS

GENERAL RECOMMENDATIONS - CHEMICAL AND BIOLOGICAL WARFARE AGENTS

1 INTRODUCTION

1.1 This leaflet gives recommended or acceptable methods of meeting certain of the basic requirements stated in Chapter 717.

1.2 Chemical and biological weapons are primarily anti-personnel weapons. They are intended to kill or incapacitate or to enforce the adoption of protective measures which degrade military efficiency. Chemical hardening is the use of designs and materials which resist the damaging effects of chemical agents and their decontaminants and additionally, facilitate the reduction or elimination of the hazard to personnel from any residual chemical contamination which may be found on equipment so that protective measures may be relaxed.

1.3 The use of chemical and biological weapons against airfields poses a threat to rotorcraft, aircrew and maintenance crews during ground operations such as replenishment, start-up, taxi, in-ground effect hovering, take-off, landing and parking operations.

2 AIRFRAME AND ENGINE CONTAMINATION BY CHEMICAL AGENTS

2.1 THE CHEMICAL HAZARD

2.1.1 From the viewpoint of chemical hardening, only those agents which remain liquid on a surface long enough for solution in the substrate to take place or to allow spreading and penetration into capillaries of various kinds represent a hazard. Agents of particular concern are therefore the semi-persistent or persistent agents.

2.1.2 Agents may be encountered as pure or thickened liquids depending on the method of delivery. Pure agents are generally disseminated as a fine spray whilst thickened agents will be found as in the size range 1-5 mm dia.

2.1.3 Chemical agents may contact equipment and material directly if they are subject to on-target attack, or contaminated terrain e.g. on landing gear, by vegetation brushing the undersides of rotorcraft or by transfer from contaminated personnel or loads.

2.1.4 Contamination of the Rotorcraft and its equipment may be significantly increased if operations are carried out on wet runways, due to the combination of chemical agents and generated water spray.

2.2 AIRFRAME

2.2.1 Chemical agents when disseminated as liquids may adhere to the surface of equipment or spread over the surfaces and penetrate into capillary spaces such as screw threads, rivets, joints, flanges etc. The solvent powers of the liquids also enable them to penetrate into permeable and porous materials such as rubber, plastics, paints, wood, foams, concrete etc. Hazards arise from the inhalation of vapour evaporating from the free liquid from vapour desorbing from permeable materials; skin contact with free or absorbed liquids also presents a hazard. Since the avoidance of contamination reduces the level of the chemical hazard, the design of equipment should as far as possible permit its operation in such a way as to minimise the degree of contamination received; the interaction of design and the development of standard operating procedures (SOPs) should be recognised. Contamination may be dangerous both to the equipment user and maintenance personnel but the nature of the hazard is different to each. In particular the danger to the maintenance personnel of contact with, and transfer of, liquid chemical agent which may have been trapped under coverplates and in screwthreads etc and which could be exposed when stripping equipment down should not be overlooked.

2.2.2 In general operational and training rotorcraft are equipped with either sliding or hinged windows/doors which can be left partially or fully open during engine run-up and taxying in case emergency exit is required. It should be assumed that under a CW or BW threat the windows/doors would be closed during ground operations.

2.2.3 Rotorcraft equipment subject to direct attack would include the following:

(i) transparencies including canopy,

(ii) radomes,

(iii) aerial arrays,

(iv) external sensing equipment including counter-measure suites,

(v) all equipment in the undercarriage bays including the landing gear,

(vi) any other equipment exposed to atmosphere.

2.2.4 Fans on air cooled equipment or in equipment bays should be provided with particulate filters (see Chap 717 para 4.2.11) and installed such that the equipment interior is under positive pressure. Consideration should be given to the hermetic sealing of the equipment.

2.3 ENGINE

2.3.1 Use is generally made of engine compressor bleed air for air conditioning, and for other systems which use air. Pre-cooled bleed air, further conditioned in an open-loop air cycle environmental control system (ECS) may be delivered to the cockpit and other systems requiring conditioned air. In the event of an attack with Chemical warfare (CW) agents rotorcraft operating in this environment will be subjected to contamination. Air from the surroundings containing CW agent, either vapour or particulate, could be drawn into the engine inlet, flowing through the engine compressor to the compressor outlet port and into the cockpit via the ECS, and other using systems.

2.3.2 The quantity of chemical or biological agent that may be ingested by the engine air inlet is dependent on engine power, rotorcraft speed and attitude and the concentration of the agent in the atmosphere or on the ground.

2.3.3 During idle or taxying the air will be ingested from a large area forward of the engine inlet duct but with power increasing up to maximum some of the inlet air will be in contact with the ground before entering the engine inlet ducting. The change in aircraft attitude during take-off, however, will decrease the quantity of air in contact with the ground until at lift-off, the effective conditions for agent contamination are similar to that of taxying operations.

3 AIRFRAME AND ENGINE DECONTAMINATION

3.1 AIRFRAME DECONTAMINATION

3.1.1 There are six decontamination methods which may be considered with special reference to their compatibility with all relevant metallic or non-metallic materials and components used in the airframe likely to be subjected to CW or BW agents. The methods are as follows:

(i) the use of active decontaminants which rapidly destroys chemical agents,

(ii) washing with a detergent and water. The action of this method is to dilute the chemical agent and to redistribute the resulting solution to additional areas of the equipment,

(iii) dusting with a solid powdered decontaminant, which leads to the absorption and retention or destruction of the chemical agent,

(iv) washing with a non-aqueous solvent, with or without the addition of a solid powdered decontaminant,

(v) chemical agent removal using sacrificial coatings. The method is being developed in which agent removal takes place by preferential absorption into a readily removable polymer film. The coating can be readily removed using a water-based stripping solution,

(vi) consideration should be given to the use of hot-air to aid the decontamination of gearbox oil cooling systems by inducing evaporation of CW agents which may have penetrated the cooling system heat exchanger or other relevant components.

3.1.2 The general effects of some decontaminants on the airframe are given below but with the exception of methods (ii) and (v) above the decontaminants present risks to the airframe and/or personnel:

(i) Chemical Agent Decontaminant (CAD). This is an aqueous solution of hydroxide and sodium dichloroisocyanurate buffered at pH 10.5 with boric acid. The resulting alkaline chlorine solution rapidly destroys chemical agents on skin, materials and equipment without dissolving plastics or damaging paintwork. It is, however, harmful if it goes in the eyes and encourages the corrosion of metals, particularly light alloys,

(ii) STB or HTH (Super-Tropical Bleach or High-Test Hypochlorite). These chlorinated limes contain 30% or 37% of available free chlorine by weight. A variety of HTH containing about 70% chlorine is also available for water purification. The limes are made into a thin slurry with water and applied to surfaces with a brush. After about 30 mins the lime is hosed away with water. Whilst on the surfaces, any camouflage protection is lost due to the "whitewash". The mixture is corrosive to metals and skin. The dry powder can cause spontaneous ignition of organic matter,

(iii) Detergent and Water. A detergent/water wash is not strictly a decontamination procedure in that the chemical agents are not destroyed, only moved to another location. This treatment is recommended where the use of a more aggressive decontaminant is inadmissible or when the agent in its new location no longer presents an operationally significant hazard,

(iv) Fuller's Earth (FE). Fuller's Earth is mainly issued as a personal decontaminant but is also reasonably effective for small equipments, except in the case of thickened agents. Agent is physically absorbed and is effectively held by the powder provided that the chemical loading of the absorbent is not excessive. The dry material is abrasive to optical surfaces. The contaminated powder is brushed or mechanically removed from the airframe by personnel wearing NBC equipment,

(v) Sacrificial Coatings. A method has been developed to remove threat agents using a sacrificial polymer coating which is able to immobilise and selectively absorb the chemical agent out of paint. The polymer system, which can be applied as an ammonia-stabilised water suspension, crosslinks on drying to form a hard coating that is resistant to wind, rain and weathering. The coating is readily removed by a 2-step process. First, the stripping solution breaks the divalent metal ion cross-links making the residue water soluble. A small amount of wash water then removes the residue from the surface.

3.2 ENGINE DECONTAMINATION

3.2.1 As CW or BW agents could attack the metallic materials used in the construction of engines, it is recommended that subsequent to the ingress or possible ingress of these agents that a liquid compressor washing procedure be carried out as a precautionary measure.

3.2.2 Decontaminants which are corrosive to metals or could lead to the blockage of engine blade cooling systems should, wherever possible, be prevented from entering the engine or any other vulnerable equipments during the decontamination process.

4 DESIGN CONSIDERATIONS

4.1 To achieve chemical hardening the design should minimise the penetration of liquid chemical agents into capillary spaces and facilitate the decontamination process. Capillary spaces are very difficult to decontaminate completely. Experiments have shown that a drop of agent falling on a screw head will penetrate along the thread; after decontamination the residual vapour hazard is about 20 times greater than from a similar drop on a plain surface.

4.2 In general the external design is of greater importance in controlling the residual vapour hazard level than the choice of materials of construction. Features which permit agent penetration and militate against efficient decontamination should be avoided or sealed where possible and include the following:

(i) joints in casings and inspection plates,

(ii) screw threads, rivets, fasteners,

(iii) transparency seals,

(iv) switch, controls, meters,

(v) connectors,

(vi) rotary and sliding seals,

(vii) hinge joints,

(viii) metal braiding on cables,

(ix) bowden and spun wire cables,

(x) chains,

(xi) surface texture of materials.

4.3 Deep surface concavities, unsealed or partially sealed quick-release fastener pockets, trap liquid CW agents and prevent access and run-off during decontamination. The deeper and narrower the concavity, the greater is the possibility of CW agent trapping and the more likely it is that chemical decontamination will be totally ineffective. The surface design should be smooth with radiused edges and corners especially where internal corners are unavoidable. Crinkle or textured finishes should be avoided.

4.4 CDE Tech Note 79 illustrates some general design guidelines to minimise penetration of liquid agents into capillary spaces and to facilitate decontamination.

5 MATERIALS

5.1 DAMAGE TO MATERIALS BY CHEMICAL AGENTS

5.1.1 Chemical agents are powerful polar organic solvents (solubility parameters in the range 8-10 Hilderbrands) and will dissolve in many polymeric materials resulting in blisters and softened spots. If blistering or sticky spots are produced and are not removed on decontamination an unacceptably dangerous contact hazard to Service personnel will exist and must be avoided by choosing appropriately resistant materials. The maximum likely agent contamination density expected is the range 10-50 gm^{-2}. The effect of such agents on the mechanical properties of bulk structural plastics is likely to be insignificant except for a few polymers where stress crazing of components under load is possible. Of much greater importance is the degradation of transparencies due to surface pitting and crazing. Such components as acrylic or polycarbonate lenses, instrument covers, windscreens and cockpit canopies are particularly vulnerable unless protection is given. Some of the nerve agents hydrolyse on standing in the liquid form in air to give dilute hydrofluoric acid. This can lead to etching of glass and germanium surfaces. The choice of suitable blooming coatings will give adequate protection to vulnerable optical components.

5.2 DAMAGE TO MATERIALS BY DECONTAMINATING SOLUTIONS

5.2.1 Current UK decontaminants may be regarded as alkaline (pH 10.5) aqueous solutions of hypochlorite. The solution will give rise to mild corrosion of light alloys. This is not generally significant operationally except for such items as rotorcraft frames where equipments may operate at stresses close to the design limits and also for electrical contact resistance. Corrosion is however a long term effect and adequate servicing schedules should largely eliminate problems from this source. Special attention, however, should be paid to the training situation in which large amounts of decontaminants are likely to be used over long periods.

5.3 PERMEABILITY OF MATERIALS TO CHEMICAL AGENTS

5.3.1 Although present decontamination techniques will efficiently remove free liquid agents from impermeable external surfaces, tactical considerations may prevent immediate decontamination from being carried out. In the interval before decontamination, agents will be absorbed into the body of many plastics, rubbers and porous materials. Decontamination is not able to neutralize this absorbed agent and subsequent desorption constitutes a continuing residual vapour hazard to personnel in the vicinity when ventilation is restricted. It is important, therefore, when trying to impart a high level of chemical hardening to select materials for the exposed parts of equipments which do not absorb CW agents. The area and rate of desorption are the important parameters - more so than absorption which is often quoted. The absorption characteristics of some materials are outlined below; desorption characteristics are also significant but are not as well defined:

(i) BARE METAL, GLASS, GLAZED CERAMICS. These surfaces are impermeable and can be decontaminated readily to a level at which desorption is negligible,

(ii) FINISHES. Alkyd and acrylic paints absorb CW agents and subsequent vapour desorption can continue for up to 2 weeks. Catalytically hardened aliphatic (2 Part) polyurethane and epoxy paints are impermeable to CW agents and can confer resistance to inferior permeable and porous substances,

(iii) FABRICS. Materials such as canvas, cottonwool, paper, leather etc rapidly absorb large quantities of CW agents. In most applications, reinforced impermeable plastics may be substituted,

(iv) WOOD. This material is absorbent unless protected by a CW agent resistant finish,

(v) RUBBERS. Rubbers vary widely in their absorptive properties. Fluorinated rubbers (viton) and butyl rubber are the most agent resistant whilst silicone rubber is generally the most permeable. Some absorptive properties of rubbers are given in Reference 1,

(vi) PLASTICS. Plastics also vary widely in their absorption of CW agents and individual plastics vary in their properties from one manufacturer to another (due to variations in molecular weight, branching, plasticizer and crystallinity). The moulding and mechanising processes involved in the fabrication of components also has an important bearing on the surface properties of polymer. Data from tables should therefore be treated as qualitative only and confirmatory tests should be carried out on the particular candidate materials chosen. PTFE (Teflon, Fluon etc) is practically impermeable. Polyolefins (polypropylene and polyethylene) are quite agent resistant. Plasticizers tend to make materials more permeable so that plasticized PVC is one of the most absorbent of the common plastics. Fillers and other loadings have little effect on permeability. The properties of glass reinforced plastic (GRP) are determined by the nature of the plastic used. Polyester-based GRP is very agent permeable but catalytically hardened epoxy resins are agent resistant. Lists of the properties of various plastics are provided in Reference 2.

6 OPERATIONAL CONSIDERATIONS

6.1 Compliance with Chapter 717 para 2.2 requires that the aircraft and equipment be operated by personnel wearing NBC clothing with the minimum loss of efficiency. In particular the following should be given considerations:

(i) To prevent damage to NBC clothing, sharp edges and corners should be avoided,

(ii) Deployment activities carried out by personnel wearing NBC gloves such as the operation of controls, adjustments, maintenance functions, replenishment etc will necessitate special considerations being given to control spacings for accurate manipulation with a gloved hand in which there is a loss of touch sense and dexterity,

(iii) Use of Contamination Monitors.

DEF STAN 00-25 (Part 10)/1 gives minimum spacing for controls etc to enable free manipulation with a low risk of incorrect operation. The control spacings are more important than control size.

6.2 When consideration is being given to the tasks to be performed by personnel wearing the Service respirator, due allowance should be made for the operator's limited field of vision. For focusing optical systems on eye relief of at least 30 mm should be allowed.

REFERENCES

1. Annex C and D to CDE Guide to Chemical Hardening CDE Technical Memorandum No 79.

2. NATO Allied Engineering Publication AEP7 Chemical defence factors in the design of military equipment.

3. Def-Stan 00-25 (Part 10)/1. Controls-Optimum size, shape, and dynamics relevant to the control task.

CHAPTER 718

INSTALLATION OF EXPLOSIVE DEVICES

1 INTRODUCTION

1.1 The requirements of this chapter apply to all installations of explosive devices in rotorcraft.

2 REQUIREMENTS FOR DESIGN

2.1 Only explosive devices and installations approved by the MOD(PE) (D.A.Arm) shall be used.

2.2 Explosive devices shall be so installed as to be easily accessible and shall not require excessive handling during installation. Consideration shall be given to making the component in which the explosive is installed readily replaceable so that the actual handling of the explosive charge can be done under servicing bay conditions.

2.3 Particular attention shall be paid to protecting the insulation of circuits connected to explosives. Protective sheaths or grommets shall be used wherever vibration or handling can cause damage.

2.4 Electrical circuits concerned with the installation of explosive devices shall conform to the requirements of Chapter 706, para 5.2.2.

3 TESTS

3.1 The Contractor shall carry out such tests as the MOD Rotorcraft Project Director may require to demonstrate the efficiency and safety of the explosive devices and their detonating systems.

LEAFLET 718/1

INSTALLATION OF EXPLOSIVE DEVICES

GENERAL RECOMMENDATIONS

1 INTRODUCTION

1.1 Explosive devices are inevitably a potential hazard to both air and ground crews. They are a hindrance during servicing, and may also necessitate additional tradesmen and installations at Service units to deal with them. They should not therefore be used if the object can be achieved without them. However, the increasing difficulty in accomplishing many operations, particularly those of an emergency character (removal of canopies and the like) seems likely to lead to an increase in their use. Their potential danger and their use in emergency functions demands a very high standard of reliability. This Leaflet therefore reviews their characteristics and makes recommendations concerning their use.

2 CHARACTERISTICS OF EXPLOSIVES

2.1 DETERIORATION

2.1.1 All explosives suffer from chemical deterioration, which, in general, increases rapidly with rising temperatures. On a purely temperature/chemical change basis alone the Ordnance Board report that, for a typical explosive device, one month at 140°F corresponds to 1½ years at 90°F.

2.1.2 Temperature changes, jolts and vibrations may affect mechanical properties, cause alterations in dimensions, break seals or fragile parts and cause crumbling of explosives.

2.1.3 A combination of these physical and chemical effects is likely to lead to the entry of moisture as the store breathes, thus providing a further potential source of deterioration.

2.2 REDUCTION IN SAFETY

2.2.1 Explosives may deteriorate in such a way that they become less safe than when new. Materials for explosive devices for use in rotorcraft are selected as far as possible so that degradation does not make them unsafe, consistent with having a material of the high degree of reliability necessary if the device is to function correctly.

2.2.2 Explosives used in rotorcraft are frequently detonated electrically. One way in which deterioration, (particularly that caused by excessive vibration or rough handling) reduces safety margins, occurs because dust from the explosive can be ignited by much less energy than that normally required. For instance, a case is recorded in which ignition was deemed to have occurred through a static charge unknowingly acquired by a man engaged on servicing operations. Action has been taken with the Services to deal with the static charge risk. Other accidental firings have occurred through induced currents from nearby circuits or from radio frequency radiations, and from the application through defective insulation of a potential difference from the metallic structure of the rotorcraft which, except in the case of some explosive circuits, is used as a common negative lead.

2.3 LIFING

2.3.1 Explosives deteriorate with time, and to ensure that they remain serviceable while they are installed, a life must be allotted. In the case of new types, or of existing types used under substantially different conditions, this may involve carrying the devices as passengers in the actual rotorcraft environment in which they are to be used so that they may be examined and tested periodically until a realistic life has been determined. The Services naturally wish for the longest possible life consistent with fitting into the normal rotorcraft servicing pattern.

3 RECOMMENDATIONS

3.1 DUPLICATION

3.1.1 In view of the high degree of reliability required, consideration should be given to the possibility of duplication of the system or of individual explosive devices.

3.1.2 The decision on the extent of duplication to be provided will need to be considered for each particular application and will depend on such factors as:

(i) the reliability of the explosive devices,

(ii) the seriousness of a failure,

(iii) the complexity of the circuits,

(iv) the power demands,

(v) the feasibility of designing and accommodating duplicated installations, neither of which will interfere with the correct functioning of the other, and

(vi) the environment of the device, which may affect its reliability.

In considering the above factors, the advice of MOD(PE) (D.A.Arm) should be sought, particularly on items (i) and (vi).

3.2 ACCESSIBILITY

3.2.1 Chapter 718 calls for explosives to be easily accessible. It has been found, particularly with explosive bolts used to operate jettison devices, that the degree of handling necessary, which is largely determined by accessibility, is a major factor in the prevalence of damage and in the production of dust from the explosive charge. This, as noted above, materially increases its sensitivity. Cases have also been found of the insulation of detonator leads being damaged, and in some cases the leads themselves being broken during the fitting of difficult bolts. Wherever possible, leads should be further protected at vulnerable points by the use of insulating sleeves.

3.3 PROXIMITY OF RADIO FREQUENCY AND OTHER CIRCUITS

3.3.1 The precautions necessary in connection with the proximity of other circuits are detailed in Chapter 710.

CHAPTER 719

PRESSURISED GAS STORAGE VESSELS

1 INTRODUCTION

1.1 This chapter applies to all pressurised gas storage vessels, made from metallic materials, for installation in rotorcraft and in equipment installed in rotorcraft. The requirements shall be read in conjunction with the appropriate parts of Chapter 703 for pneumatic systems.

1.2 This chapter provides:

(i) the basic design requirements for pressurised gas storage vessels,

(ii) the approval tests to be applied to verify the design and to establish a production standard,

(iii) the mandatory tests to be applied to production vessels,

(iv) the information to be marked on the vessel and on the drawing.

1.3 This chapter states essential design and test requirements. Conditions of service use (particularly service life and permissible charging pressures) shall be determined by the designer and agreed with the Rotorcraft Project Director taking into account the design requirements and the results of all tests.

2 CLASSIFICATION

2.1 The requirements apply to five distinct types of vessel, depending on design and purpose as follows:

Type 1 Vessels charged by a ground rig before flight and not re-charged in flight and not having a pressure relief valve on the vessel or in the system. But see para 4.3.

Type 2 Vessels charged by a compressor on the rotorcraft, whether initially charged by a ground rig or not, in a system having pressure regulating and pressure relief devices in accordance with Chapter 703 paras 2 and 3.

Type 3 As Type 2, but having a solenoid or other component in the system which can cause transient pressures higher than the relief valve relief pressure.

Type 4 Vessels fitted uncharged, not having a relief valve, in a system charged by a cartridge or similar gas source. But see para 4.3.

Type 5 Vessels charged and fitted in equipment which may be stowed in the rotorcraft or carried by the crew or other occupants, fitted with a piercing disc which may also act as a burst disc where necessary, discharged only in emergency and during refurbishing.

2.2 Any particular vessel may be designed to meet the requirements of more than one type. Such vessels shall meet the requirements for both types.

3 DEFINITIONS

3.1 DESIGN PRESSURE (P_d). The maximum pressure expected to arise in service from all causes (All types) (See paras 4.6 and 4.7 below).

3.2 DESIGN CHARGING PRESSURE (P_c). The maximum permissible charging pressure at 20°C, for which the vessel is designed. (Types 1 and 5).

3.3 DESIGN PRESSURE RATIO (R). The ratio of the pressure at maximum design temperature to the pressure at 20°C, obtainable from standard tables for the gas concerned. The temperature used to determine this ratio shall include an allowance for any local rise in temperature during normal operations caused by the position of the vessel in the Rotorcraft. (Type 1).

3.4 WORKING PRESSURE (P_w). The maximum normal working pressure for the relevant part of the system as defined in Chapter 703 para 2.1 (Types 2, 3 and 4).

3.5 RELIEF PRESSURE (P_r). The maximum pressure at which the pressure limiting device, as defined in Chapter 703 para 3.1, operates. A value of 1.33 P_w is implied by the requirements of Chapter 703 and covers the effects of variability in maximum delivery pressure (nominally 10%), ingress of foreign matter, filter blockage (Leaflet 703/1), and temperature changes. In some rotorcraft a value greater than 1.33 P_w may be necessary to prevent excessive loss of air from the system following cold soak at altitude and rapid descent to ground level in a high temperature. (Types 2 and 3).

3.6 BURST DISC FAILURE PRESSURE (P_b). See para 4.3. The maximum pressure at which the burst disc is expected to operate including an allowance (normally 20% above nominal) for variability. A recommended value is 1.5 Pd. (Types 1, 4 and 5).

3.7 MOUNTING STRESS ALLOWANCE (MSA). The allowance required to take account of clamping and inertia forces and of their reactions. The value is to be agreed with the system designer for the Rotorcraft. For strapped vessels the allowance should be the pressure stress equivalent to the maximum local acceleration in the worst flight manoeuvre. For vessels mounted on bosses the local stress caused by this acceleration may be applied separately.

3.8 TRANSIENT PRESSURE ALLOWANCE (TPA). The pressure allowance above P_r for short duration transient increases in pressure arising from solenoid operation in some systems and from explosive forces in others. The allowance should be based on relevant experimental evidence, if available. It should be not less than 50% of P_r (Type 3) or 50% of P_w (Type 4) if no relevant evidence is available.

3.9 SAFE LIFE. The maximum number of inflations (See para 3.10) permitted during the Service Life of the vessel. The safe life will be stated in the appropriate specification and shall be not less than the equivalent of 10 years service use unless otherwise agreed with the Rotorcraft Project Director.

3.10 INFLATION. In service, an inflation will be deemed to have taken place when, on charging the vessel, the pressure passes through a level of 80% of P_c for Type 1 vessels, once per engine start or per flight for Types 2 and 3 vessels and once per usage for Type 4 and 5 vessels. In tests, a cycle is an inflation from zero to P_d to zero except in some Type 2 and 3 vessels (para 5.5.2).

3.11 LIMIT OF EXPANSION. The time at which, during a pressure test, the rate of change of volume drops to zero.

4 DESIGN REQUIREMENTS

4.1 The design of all Pressurised Gas Storage Vessels shall be such that the finished vessel, including its protective coating:

(i) passes the proof, fatigue, and ultimate strength tests of para 5 appropriate to the type,

(ii) passes the vulnerability tests appropriate to the design energy capacity. See Leaflet 719/1.

4.2 The design of all Pressurised Gas Storage Vessels should also, so far as it is possible to determine in advance:

(i) meet the strength and life requirements in the environmental conditions, such as temperature, humidity, vibration, shock and acceleration, to which it may be subjected in service,

(ii) not be affected detrimentally by any gases or liquids that may come into contact with it during normal use,

(iii) not introduce contaminants into the gas contained in it beyond the level permitted by the appropriate gas specifications.

4.3 A pressure relief device known as a burst disc shall be fitted to all Type 1, 4 and 5 vessels:

(i) having a high energy rating (See Leaflet 719/1),

(ii) having a medium energy rating located where they could cause injury to personnel in a crash.

4.4 The burst disc shall be designed to release the pressure in a controlled manner as far as is possible.

4.5 Manufacturing processes for vessels used for type approval tests shall be adequately defined and all subsequent production shall be to this defined standard. Proposed departures from this standard shall be referred to the Rotorcraft Project Director.

4.6 The design of vessels shall be based on:

(i) the Design Pressure (P_d),

(ii) the Safe Life required.

4.7 The design pressure (P_d) shall be not less than:

(i) Type 1 $(P_c \times R) + MSA$,

(ii) Type 2 $P_r + MSA$,

(iii) Type 3 P_r + TPA + MSA,

(iv) Type 4 P_w + MSA + TPA,

(v) Type 5 (P_c x R) + MSA.

4.8 All vessels shall be designed to withstand without leakage or distortion a proof pressure of:

(i) 1.5 P_d where P_d is derived from P_w or P_c

(ii) 1.125 P_d where P_d is derived from P_r

4.9 All vessels shall also be designed to withstand without fracture or bursting an ultimate pressure of:

(i) 2.0 P_d where P_d is derived from P_w or P_c

(ii) 1.5 P_d where P_d is derived from P_r

4.10 Where it is proposed to fill the vessel with a liquefiable gas, the filling ratio shall be such that, in all conditions within the environmental envelope, the vessel cannot be full of liquid.

4.11 For high and medium energy vessels (See Leaflet 719/1) irrespective of the results of the vulnerability test (See para 5.8) and the allowance made for MSA (See para 3.7) consideration shall be given to the case where a bullet or fragment makes a hole in the vessel and releases the energy. Possible effects on the vulnerability of structure and systems shall be considered and included in the Vulnerability Analysis of Chapter 112.

5 TYPE APPROVAL TESTS

5.1 SAFETY

5.1.1 These tests are hazardous. Attention is drawn to the need for precautions in accordance with the Health and Safety at Work Acts. Tests will normally be done in the local ambient temperature but see para 5.8.2.

5.2 NUMBER OF VESSELS

5.2.1 All vessels used for type approval tests shall be individually numbered and each initially subjected to the proof test of para 5.4.

5.2.2 Eighteen vessels shall be provisioned for the tests and allocated as follows:

(i) Nos 1 to 6 for the tests of para 5.5,

(ii) Nos 7 to 12 for the tests of paras 5.6 and 5.7,

(iii) Nos 13 to 18 for the tests of para 5.8.

5.2.3 For any one vessel of a group of six that fails one of the following tests, a further two shall be successfully tested, unless the failure indicates the need for redesign, when the procedure must be started again

5.3 PHYSICAL CONDITION

5.3.1 All vessels used for type approval tests shall be fully representative of subsequent production (See also para 7).

5.3.2 All vessels shall be complete with all permanent end fittings and representative labels. They shall be marked with all incised or embossed production marks of DEF STAN 81-24. Hardness marks shall be to the same standard as on production vessels. No pressure greater than P_d shall have been previously applied to them.

5.3.3 In order to check subsequent performance, the mass, dimensions, and internal volume of each vessel shall be recorded before the commencement of tests.

5.3.4 Where a burst disc is provided it shall be blanked-off in a manner which will not invalidate the test results.

5.4 PROOF TEST

5.4.1 Each of the vessels used in a type approval test shall first be subjected hydraulically to the proof pressure (para 4.8) for long enough to verify by the volume measuring apparatus that the limit of expansion has been reached or for 30 seconds, whichever is the greater.

5.4.2 The method of test and the results shall be in accordance with BS 5045 Part 1.

5.4.3 Where the vessel is subsequently to be filled with oxygen the hydraulic fluid used for this test must be non-toxic and sufficiently volatile to be easily removed.

5.5 FATIGUE TEST

5.5.1 The vessels numbered 1 to 6 shall be subjected to pressure cycles (see para 3.10) applied hydraulically from atmospheric pressure to Pd. Failure shall be deemed to have taken place as soon as cylinder leakage is detectable. The test pressure and number of cycles shall be recorded. The results shall be interpreted according to the requirement of para 4.11, above.

5.5.2 If variations of test pressure are considered desirable either:

(i) to represent variations in working or charging pressure, or

(ii) because P_r is greater than 1.33 P_w and is not reached every flight,

then a spectrum more representative of service conditions may be used subject to the agreement of the Rotorcraft designer.

5.5.3 The geometric mean life of the fatigue tests shall be F times the required life where F is given by the following table and v is the sample coefficient of variation of the logarithm of the life of each specimen. Intermediate results may be interpolated. See Leaflet 719/2. Note that F=2.8 for all values of v up to 0.02.

v	0.02	0.025	0.03	0.035	0.04	0.045	0.05
F	2.8	3.8	5.4	7.5	10.5	14.3	20.0

5.6 VIBRATION, SHOCK, ACCELERATION, CLIMATIC AND AGEING TESTS

5.6.1 If agreed by the Rotorcraft Project Director, some of these tests may be combined with the fatigue tests of para 5.5 to produce a more realistic representation of service conditions.

5.6.2 After the proof test of para 5.4 the vessels numbered 7 to 12 shall first be subjected to one safe life of the fatigue test of para 5.5 without leakage or failure. The vessels shall then be subjected, whilst pressurised pneumatically to P_C (Type 1) P_W (Types 2 and 3), or 10% of P_W (Type 4), as appropriate, and mounted in a representative fashion, to tests to a schedule (based on BS 2G100, BS 3G100, DEF STAN 07-55 and Chapter 101) agreed with the Rotorcraft Project Director. They shall withstand these tests without failure or leakage.

5.7 ULTIMATE AND BURST TESTS

5.7.1 The vessels numbered 7 to 12 shall, following the tests at para 5.6 above, be submitted hydraulically to the ultimate pressure for a period of two minutes without fracture or bursting. The pressure shall not vary during this time.

5.7.2 The pressure shall then be increased steadily in increments and the value at which fracture or bursting occurs shall be recorded.

5.7.3 The arithmetic mean and standard deviation, and hence the co-efficient of variation (v), of the burst test results shall be calculated. The ratio of the mean to the design ultimate pressure shall not be less than F as shown in the table below. Intermediate results may be interpolated. See Leaflet 731/2 para 3.

v	0.02	0.03	0.04	0.05	0.06	0.07	0.08
F	1.05	1.06	1.09	1.12	1.16	1.20	1.25

5.7.4 If a co-efficient of variation greater than 0.08 is obtained this indicates a need for redesign or improved manufacturing practices.

5.8 VULNERABILITY TEST

5.8.1 The preferred vulnerability test is the 'Cuboid Test' described below and in Leaflet 719/1. This test shall be done on vessels Nos 13 to 18, by an MOD Approved authority, after they have passed the proof test of para 5.4. Exceptionally, the Rotorcraft Project Director may authorize the 'Gunfire Test' described in Leaflet 719/3 as an alternative test for gas vessels and that described in Leaflet 719/4 for gas/oil vessels, provided that in the case of cylinders less than 64mm dia., the Vulnerability Analysis of Chapter 112, para 4 shows this does not increase the overall vulnerability unacceptably.

5.8.2 Each vessel shall be charged with the appropriate gas to a pressure not less than P_d. For liquefiable gases this shall be done by charging the vessel with the appropriate mass and raising the temperature. For permanent gases this may be done at ambient temperature by pressurising the vessel.

5.8.3 Each vessel shall then be attacked, at a striking velocity of 1830 m/s ± 3%, with a 7g cuboid silver steel fragment (BS 1407) hardened and tempered to between 300 and 350 Vickers Hardness Number.

5.8.4 All strikes shall be at approximately the centre of the vessel and normal to the surface. There shall be no fragmentation of the vessels and they shall not sustain combustion.

5.9 REDUCED TYPE APPROVAL TESTING

5.9.1 A reduced level of testing for type approval may be agreed with the Rotorcraft Project Director, provided that there is sufficient relevant test evidence available for similar types of vessels and the vulnerability requirements of Leaflet No.719/1 are not reduced.

5.10 RECORD OF TYPE APPROVAL, TESTS

5.10.1 The results of all tests shall be submitted to the Rotorcraft Project Director.

5.10.2 A record shall be made, in accordance with DEF STAN 05-123, of the satisfactory completion of the type approval test and the modification standard at which the tests were made. This should include relevant details of P_d, P_c, P_r and P_w, all test failures including fatigue, inflation limitations, and the safe life of the vessel.

5.11 DISPOSAL

5.11.1 When all tests and investigations are complete and the results have been accepted by the Rotorcraft Project Director, the vessels used in Type Approval Tests shall be destroyed in accordance with BS 5430.

6 PRODUCTION TESTS

6.1 PRODUCTION PROVING TEST

6.1.1 Each production vessel shall be subject to a proof test as described in para 5.4.

6.1.2 Provided that there are no differences between the type test vessel and the production vessels, in design and manufacturing processes, this is the only test applicable to the first 100 production vessels.

6.2 PRODUCTION FATIGUE TESTS

6.2.1 Six of the second hundred vessels, selected at random, having already passed the proving test of para 5.4, shall be subject to the fatigue test of para 5.5. The results shall be interpreted according to the requirement of para 4.10, and the batch or production run shall be accepted or rejected accordingly.

6.2.2 From each subsequent batch or production run of 100, 6 shall be similarly tested and the batch or production run similarly accepted or rejected.

6.2.3 If a batch of less than a hundred is made, six shall nevertheless be tested. These will clear further production up to a hundred provided there are no changes.

6.2.4 Where more than a hundred are made in one batch without varying the manufacturing conditions in any way the sample size (n) may be determined by the formula:

$$n = N/100 + \sqrt{N/2}$$

Where N is the batch size. (Round up or down to nearest integer).

6.2.5 Where the fatigue life is estimated by the results of the type approval tests, extended if necessary for the purpose, is assessed as being infinite or very large compared to the service requirement, as in the case of vessels discharged only in emergency and during refurbishing, the production fatigue test requirements may be waived if agreed by the Rotorcraft Project Director.

6.3 MARKING

6.3.1 All vessels shall be clearly marked in accordance with DEF STAN 81-24. The date of manufacture shall be taken as the date of the proof test.

6.4 ACCURACY

6.4.1 Particular care shall be taken to ensure the accuracy of all measuring equipment and methods to avoid overstressing the vessels. After production testing, the vessels shall be adequately dried and cleaned before further use.

6.5 DISPOSAL

6.5.1 When all tests and investigations are complete and the results have been accepted by the Rotorcraft Project Director, the vessels used for the fatigue tests shall be destroyed in accordance with BS 5430.

7 CHANGES

7.1 MODIFICATION

7.1.1 If any change is made to the design or the method of manufacture which could affect any of the properties tested or any of the requirements of Para 4, the relevant type approval tests shall be repeated.

7.2 CHANGE OF MANUFACTURER

7.2.1 If it is proposed to change the manufacturer but not the design or methods of manufacture, it may be necessary to repeat some of the type approval tests to validate the new source of supply. Any such proposal shall be discussed with the Rotorcraft Project Director and tests to an agreed programme shall be done.

7.3 DERATING

7.3.1 Where a vessel is supplied for use in a derated condition (that is when the original design pressure (P_d) is larger than would be necessary for the purpose for which the vessel is now to be used) all production tests shall be based on the original P_d whether or not the vessel is fitted with a new pressure relief valve or burst disc having a lower rating.

LEAFLET 719/1

PRESSURISED GAS STORAGE VESSELS

DEFINITION OF ENERGY LEVELS FOR VULNERABILITY TEST REQUIREMENTS

1 INTRODUCTION

1.1 This leaflet defines energy levels for pressurised gas storage vessel vulnerability test requirements.

2 HIGH ENERGY LEVELS

2.1 Defined as energy levels greater than 44.75kJ (equivalent to 20 mm explosive shell). Tests to check liability of vessels to shatter explosively are mandatory.

3 MEDIUM ENERGY LEVELS

3.1 Defined as energy levels less than 44.75kJ but greater than 13.6kJ. Consultation with the Rotorcraft Project Director is required to decide whether vulnerability tests are necessary. Where the vessels are situated within containers or equipment able to absorb the released energy and particles, the risk of damage to adjacent equipment, structure, or personnel, may be low and no special tests may be required. In addition the vessels may have been constructed in such a manner as to remove the danger of fragmentation.

4 LOW ENERGY LEVELS

4.1 For vessels pressurised to low energy levels defined as less than 13.6kJ it can be assumed that there will be no special vulnerability hazards and vulnerability tests are not required.

5 STORED ENERGY

5.1 The following formulae may be used for the calculation of energy level(W):

5.1.1 <u>For diatomic gases (e.g., Air, Oxygen, Nitrogen)</u>

W = 2pv where W is in Kilojoules (kJ), p is Megapascals (MPa) and v in litres,

5.1.2 <u>For monatomic gases (e.g., Helium)</u>

W = 1.5pv where W is in Kilojoules (kJ), p is in Megapascals (MPa), and v in litres,

5.1.3 <u>For all liquefiable gases (e.g. Carbon Dioxide or Freon)</u>. Simple formulae are not accurate. The stored energy must be calculated as the change of internal energy when the gas expands reversibly from Pd to atmospheric pressure.

5.2 In the formulae of para 5.1: p shall be P_d and v the design volume of the vessel at 20°C.

5.3 If p is in bars instead of MPa divide W by 10 (1 Bar = 0.1 MPa).

LEAFLET 719/2

PRESSURISED GAS STORAGE VESSELS

FATIGUE AND STATIC TEST EXAMPLES

1 INTRODUCTION

1.1 This leaflet contains fatigue and static test examples for pressurised gas storage vessels.

2 FATIGUE TEST EXAMPLE (see Chapter 719 Para 4.10)

2.1 Suppose a particular vessel is required to have a Service Life of 10,000 cycles and that the following results are obtained in the fatigue type test of Chapter 719 para 5.5:

Cycles	**$\log_{10}$**
25020	4.39829
31570	4.49927
39980	4.60184
40100	4.60314
48770	4.68815
50190	4.70062

2.2 The mean ($\bar{x}$) of the logs is 4.58188 and the standard deviation is given by:

$$S = \left(\sum_{i=1}^{n} (x_i - \bar{x})^2 / (n-1) \right)^{1/2}$$

Where x_i are the statistics (the logs) and n the number of statistics.

2.3 In this case the standard deviation is 0.11556 and this gives a coefficient of variation v = S/ x = 0.02522. From the table in Chapter 719 para 5.5.3 the required factor F = 3.8 and hence the required geometric mean life is 38,000 cycles.

2.4 The achieved geometric mean life is given by:

$$\text{Antilog} \left(\sum_{i=1}^{n} \log x_i \right) / n$$

= Antilog 4.58188

= 38,184 cycles

Which is just satisfactory.

3 STATIC TEST EXAMPLE (see Chapter 719 Para 5.7)

3.1 Suppose the following results are obtained in the burst tests (all expressed as percentages of the design ultimate pressure):

105, 107, 110, 112, 113, 115

3.2 The mean of these six statistics is 110.333% DUP. Using the above formula the standard deviation is 3.77712% DUP; and the coefficient of variation is 0.03423. The factor F in the table of Chapter 719 para 5.7.3 is the ratio of the minimum acceptable mean value of the results to the design ultimate. From the table by interpolation F is 1.07 and the minimum acceptable value of the mean is therefore 107. As 110.333 is greater than 107 the results are considered satisfactory.

LEAFLET 719/3

PRESSURISED GAS STORAGE VESSELS

FRAGMENTATION TEST REQUIREMENTS

1 INTRODUCTION

1.1 This leaflet provides details of the Gunfire Test, (see Chapter 719 para 5.8) to check the liability of vessels to shatter explosively. The test is based on Military Specification MIL-C-7905E 'Cylinders, Compressed Gas, Non Shatterable'.

2 FRAGMENTATION RESISTANCE

2.1 The cylinder, when tested as specified in para 3 below shall remain in one piece, except cylinders that are 64 mm or less in diameter, will be permitted to separate into 2 pieces, and exhibit no evidence of shattering. If wire or fibre wrapping is used, the wrapping may come loose from the cylinder.

3 GUNFIRE TEST

3.1 Cylinders greater than 64 mm in diameter shall be subjected to gunfire under the following conditions:

(i) Each vessel shall be charged with the appropriate gas to the design pressure P_d at gun range ambient air temperature,

(ii) The cylinder may be supported but not constrained,

(iii) The ammunition shall be 0.5 calibre M.2, armour-piercing,

(iv) The range shall be 46 metres maximum,

(v) The various cylinders taken for test shall be tested, each progressively, in a different position, as follows:

(a) With the longitudinal axis of the cylinder normal to the line of fire,

(b) With the longitudinal axis of the cylinder 45 degrees (.785 rad) from normal toward the gun position,

(c) With the longitudinal axis of the cylinder parallel to the line of fire with inlet port face away from the gun position,

(vi) All shots shall be tumbled,

(vii) The tumbled projectile shall have a minimum velocity of 792 m/s at the point of impact with the cylinder,

(viii) The minimum size entry hole made by the tumbled projectile shall be 13 mm by 39 mm,

(ix) Verify that the projectile trajectory and tumble are satisfactory. This may be determined by the location and visual appearance of a hole made by the passage of the projectile through vertically suspended sheets of paper at the target area.

The cylinder shall pass the requirements of para 2.1

3.2 Cylinders 64 mm or less in diameter shall be subjected to gunfire under the following conditions:

(i) The cylinder shall be charged to its design pressure at gun range ambient air temperature,

(ii) The cylinder may be supported but not constrained,

(iii) The ammunition shall be .30 calibre armour piercing, with a muzzle velocity of 853 ± 30 m/s,

(iv) The range shall be approximately 18 m,

(v) Shots shall not be tumbled,

(vi) The various units taken for tests shall be positioned as in para 3.1 (v).

The cylinder shall pass the requirements of para 2.1.

Further details of the test and the number of vessels to be tested can be obtained from MIL-C-7905E.

LEAFLET 719/4

PRESSURISED GAS STORAGE VESSELS

FRAGMENTATION TEST REQUIREMENTS FOR GAS/OIL HYDRAULIC ACCUMULATORS

1 INTRODUCTION

1.1 This leaflet provides details of the Gunfire Test, (see Chapter 719 para 5.8) to check the liability of vessels to shatter explosively. The test is based on MIL-C-7905E - 'Cylinders, Compressed Gas, Non Shatterable' and also encompasses the requirements of para 4.7.10 of MIL-A-8897A - 'Accumulators, Hydraulic, Cylindrical, 3000 PSI Aircraft, Type II Systems'.

2 FRAGMENTATION RESISTANCE

2.1 The cylinder, when tested as specified in para 3 below shall remain in one piece and exhibit no evidence of shattering. The material of the cylinder shall not tear excessively in any one direction. If wire or fibre wrapping is used, the wrapping may come loose from the cylinder.

3 GUNFIRE TEST

3.1 Cylinders shall be subjected to gunfire under the following conditions:

(i) Each vessel shall be charged with the appropriate gas and fluid to the design pressures at gun range ambient air temperature,

(ii) The cylinder shall be supported in a manner similar to a typical rotorcraft mounting,

(iii) Attached to the oil port shall be a length of tubing with a shut-off valve located 900mm from the port,

(iv) The ammunition shall be 0.5 calibre incendiary projectile,

(v) The range shall be 23 metres maximum,

(vi) The projectile shall be so directed that it will hit the fluid side of the cylinder approximately mid-way between the piston and the mounting strap,

(vii) All shots shall be tumbled by the use of a non-metallic tumbling board placed close enough to the target to provide adequate tumbling,

(viii) The tumbled projectile shall have a minimum velocity of 792 m/s at the point of impact with the cylinder,

(ix) The minimum size entry hole made by the tumbled projectile shall be 13mm x 39 mm.

(x) Verify that the projectile trajectory and tumble are satisfactory. This may be determined by the location and visual appearance of a hole made by the passage of the projectile through vertically suspended sheets of paper at the target area.

CHAPTER 720

AIR LAUNCHED WEAPON INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this Chapter, unless specifically excepted in the Rotorcraft Specification, apply in all cases where air launched weapons are carried on a Rotorcraft. This chapter is complementary to Chapter 710 Armament Installations. The weapon installation shall comply with Def Stan 08-5, Design Requirements for Weapon Systems, (Guided weapons, torpedoes and airborne armament stores). However it should be noted that where weapons are procured under foreign purchase or collaborative arrangements the weapons may have been designed to other design requirements.

1.2 The weapons to be fitted to the Rotorcraft will be stated in the Rotorcraft Specification. Paragraph 2 of this chapter refers to other information which will be found in the Specification. This includes the weapon launch pattern, the Operational Flight Envelopes for the Rotorcraft, when carrying, launching and jettisoning a weapon.

1.3 For each weapon installation an interface specification will be prepared by the Rotorcraft Design Authority, agreed through the MOD Rotorcraft/Weapon Project Director with the Weapon System Design Authority and referenced in the Rotorcraft Specification. Def Stan 05-123 Chapter 203 refers.

2 OPERATIONAL REQUIREMENTS

2.1 The firing of the weapons shall not affect the operational function or safety of the Rotorcraft. The effects of discarding weapon containers, frangible covers and protective devices at launch, including the random impact of discarded pieces on the Rotorcraft structure, shall also be considered.

2.2 It shall be possible to fire or launch the weapon singly or in multiple launches throughout the Rotorcraft Operational Flight Envelope for weapon carriage and launch stated in the Rotorcraft Specification. See also Leaflet 600/3.

2.3 Provision shall be made for landing and, when required by the Rotorcraft Specification, arresting the Rotorcraft with the weapon in position.

2.4 The Rotorcraft Specification will state whether a jettison system is required and, in the event of a crash landing or ditching with the jettison system failed, whether on initial impact:

(i) the weapon is required to remain attached to the Rotorcraft.

(ii) the attachments are required to be frangible so that the weapon parts from the Rotorcraft. See also para 4.2.

2.5 It shall be possible to carry and jettison the weapon safely throughout the Rotorcraft Operational Flight Envelope for which the Rotorcraft Specification requires a weapon jettison capability. This may require bomb doors to be opened.

2.6 When the Rotorcraft is on the ground it shall be possible to release any weapon or its jettisonable carrier without entering the cockpit.

3 INSTALLATION REQUIREMENTS

3.1 The installation shall satisfy, where applicable, the requirements of Chapter 710.

3.2 Any special requirements for clearance between the weapon system and the Rotorcraft structure shall be included in the weapon installation interface specification together with permissible tolerances for mounting the weapon relative to the Rotorcraft axes.

3.3 The installation shall be such that its efficiency and safety are not impaired by the operation of other weapons or systems.

3.4 Provision shall be made so that the weapon cannot be launched while any part of the Rotorcraft is in such a position as to obstruct its line of travel. Additionally, any such provision shall not affect the ability to jettison the weapon safely, when jettison is required.

3.5 The installation shall be such that the presence of the weapons does not interfere with the aircrew's means of escape from the Rotorcraft in an emergency.

3.6 Protection of the aircraft against boost rocket blast shall be provided if necessary.

3.7 Consideration shall be given in the design to minimise the drag from the installed weapons, and to avoid static and oscillatory air loads which could damage the weapons.

3.8 The location of the installation on the Rotorcraft structure shall allow sufficient clearance for the Rotorcraft, with the weapon installed, to traverse those obstacles commonly found on airfields (e.g. arresting gear hook cables) and, where relevant, on the decks of aircraft carriers.

3.9 The installation shall be such that it permits the most simple and efficient method of handling, fitting and removing the weapon. A clear access shall be provided to the installation.

4 STRENGTH REQUIREMENTS

4.1 STRENGTH FOR NORMAL OPERATION

4.1.1 The strength of the installation carrying the air launched weapon and of the weapon shall be such as to withstand all conditions of take off, flight, weapon launch, landing and, when applicable, arresting of which the Rotorcraft is capable when carrying all or some of the weapons. The structural design requirements for the weapon system are given in DEF STAN 08-5 Part 5. Requirements specific to the carried weapon are contained in Chapter 5-201 ‘Structural Requirements for

Weapons Prior to and During Launch, Release or Ejection' and those for the interface system in Chapter 5-402 ' Structural Requirements for Aircraft/Weapon Interface Equipment.'.

4.2 STRENGTH FOR CRASH LANDING AND DITCHING

4.2.1 Where it is required that the weapon shall remain in position on the Rotorcraft throughout a crash landing or ditching, the weapon, its attachments, and the components of the interface systems, shall meet the static and dynamic strength requirements of Chapter 307.

4.2.2 Alternatively, where frangible attachments are provided, the requirements of para 4.1 will apply. However, frangible attachment may have significant implications for safety in the normal use of the weapon system and therefore shall be provided only where called for in the Rotorcraft Specification. Additional tests to demonstrate that the design meets the carriage strength especially where cyclic loading is involved, may be required.

5 TESTS

5.1 The Rotorcraft Design Authority shall demonstrate on a working rig of the installation, by wind tunnel tests and by ground and air launching trials on a Rotorcraft, that the installation functions satisfactorily and meets the requirements of the Rotorcraft Specification.

5.2 The wind tunnel tests, ground rig tests and ground tests on the Rotorcraft shall demonstrate that:

(i) the weapon can be loaded easily and released when the Rotorcraft is at rest on the ground.

(ii) as far as is practicable, the weapon can be launched in flight under the conditions specified.

(iii) as far as is practicable, there is no unacceptable interference with the flying characteristics of the Rotorcraft.

(iv) the jettisoning arrangements will function satisfactorily.

5.3 The flight tests shall demonstrate that:

(i) the weapons can be launched under the conditions specified in the Rotorcraft Specification.

(ii) there is no unacceptable engine malfunction.

(iii) there is no unacceptable interference with the flying characteristics of the Rotorcraft and,

(iv) the jettisoning arrangements will function satisfactorily

5.4 The requirements for flight tests of air launched weapon system installations are given in Chapter 1014.

CHAPTER 721

EMERGENCY LIFERAFT INSTALLATIONS

1 INTRODUCTION

1.1 The requirements of this chapter apply to the installation of multi-seat emergency liferafts.

2 BASIC REQUIREMENTS

2.1 Materials, components, standard parts, processes, corrosion protection and design features, shall comply with the provisions of Chapter 801. Unless otherwise specified, the materials and components used in the construction of the release systems for the ejection of the liferafts from the rotorcraft shall be selected such that the effect of elapsed time from manufacture either in equipment being stored, on the shelf, or on the rotorcraft, will have no detrimental effects upon successful operation of the system.

2.2 The location of liferafts and their methods of release shall be suitably chosen in relation to the ditching and flotation characteristics of the rotorcraft, the escape facilities of the rotorcraft, and the disposition of the occupants. Under no circumstances shall the compartment opening be below the estimated water line of the ditched rotorcraft. The structural strength of the area shall be capable of withstanding an emergency ditching without collapsing or damaging the compartment. Furthermore, the compartment shall not be located in an engine nacelle, nor close to hot areas such as jet pipes or exhaust pipes. Heat is very detrimental to rubber and can be dangerous to a charged gas cylinder.

2.3 Every effort shall be made to ensure that the location of the compartment in relation to the slip stream and downdraft of the rotorcraft and the location of the rotors, aerials and stabilizers shall be such that, not withstanding para 2.8, no part of the rotorcraft critical to continued safe flight shall be damaged or fouled by the compartment cover, liferaft, or by any component of the liferaft in the event of a release system malfunction causing an inadvertent operation of the release system during flight.

2.4 Liferaft installations shall be designed so as to ensure that the liferaft, together with any equipment, provisions, etc., which are required to be carried, will be serviceable when needed and that, so far as is practical, the installations will not be damaged by any defect likely to necessitate an Emergency Alighting on water (eg engine fire or fuel leakage) or by any damage liable to occur during such an alighting.

2.5 Following the operation of the releases provided, the liferaft and all equipment stowed with it shall be ejected from its compartment in such a way that the liferaft is launched the right way up (unless it is of the reversible type) with its equipment within it secured in a position which will enable the crew to board it with the least possible risk of getting wet. The installations shall be so arranged that the necessary action for launching any liferaft is not likely to be either an undue hindrance to or unduly hindered by, the occupants vacating the rotorcraft. The liferaft shall be attached to the rotorcraft by a line. This attachment, however, shall be incapable of submerging or capsizing the loaded liferaft when the rotorcraft sinks.

2.6 The correct operation of the liferaft installation shall not be prevented by the formation of ice.

2.7 It shall be impossible to complete the installation unless all components have been correctly and securely fitted, and all safety devices used for transit purposes, etc., removed.

2.8 The stowage of all liferafts shall be such that the probability of unintentional inflation or release is not greater than Extremely Remote, unless it can be shown that such inflation or release will not hazard the rotorcraft.

3 LIFERAFT COMPARTMENT

3.1 Each liferaft and its associated inflation and survival equipment shall be installed in a pre-packed pan. The pan shall provide positive location for inflation cylinders and any other articles which might otherwise shift and damage or disarrange the liferaft or the release of operating mechanism. Similarly the operating mechanism for the release of the liferaft from the rotorcraft shall be installed as a self-contained unit.

3.2 The liferaft pre-packed pan and the release operating mechanism unit shall be capable of being removed from or fitted to the rotorcraft quickly and easily and without the use of special tools. It shall be possible to remove either item independently and without disturbing the other.

3.3 The liferaft stowage compartment and its pre-packed pan, its operating mechanism and its cover, shall be free from all sharp corners, edges or projections, which might trap or damage the liferaft either while stowed or during inflation.

3.4 A cover shall be provided so arranged that under all possible flight conditions it will be held firmly shut by the cover locks with no possibility of even partial opening of its edges. The security of the locks shall be readily ascertainable by visual inspection when the rotorcraft is on the ground.

3.5 The stowage compartment shall be substantially weathertight and yet designed to drain completely in both ground and flight attitudes.

3.6 Provision shall be made to prevent any excessive build up of pressure within the liferaft stowage due to air or due to high temperature discharge or seepage of the gas cylinder.

3.7 The stowage of all liferafts (including valise-types) shall be such that the liferafts will not be disturbed except for inspection purposes and, under normal conditions, will not be liable to damage by passengers or by the loading or unloading of freight.

3.8 Stowages shall be such as to prevent damage to or deterioration of the liferaft package due to spillage of or contact with any likely contaminants, including deleterious quantities of water.

3.9 The total volume of the compartment shall be 10 percent greater than the full volume of the equipment which it is to contain. With the full equipment installed only light hand pressure shall be required to close and lock the lid correctly.

4 RELEASE

4.1 Two independent liferaft release systems shall be provided, one operating automatically when the rotorcraft has ditched and the other one manually. Such manual release shall function even if the automatic control fails to function. The release systems shall be capable of operating satisfactorily in all climates, when subject to any ambient atmospheric temperature between 70 degrees Centigrade (158 degrees Fahrenheit) and minus 54 degrees Centigrade (minus 65 degrees Fahrenheit).

4.2 The automatic release system shall operate so that there is no danger of the liferaft inflating prematurely and being lost. The release system shall not be activated by electromagnetic radiation, oil leakage, washing down, condensation, flight and gunfire vibration, take-off, landing, change in altitude nor by the retardation effects of ditching or crashlanding. In addition the automatic release system shall have its own source of power so that it will function in the event of failure of the primary rotorcraft electrical system.

4.3 The entire mechanism of the manual and automatic systems, when operated by any method, shall not foul upon operation and shall automatically and completely disconnect the release connections, after the compartment cover has released and the gas filled cylinder on the stowed liferaft has been actuated. The compartment cover or any component of the manual or automatic release systems shall not interfere with the ejection of the liferaft or with the liferaft once it has been ejected.

4.4 The system shall be installed in accordance with the provisions of Chapter 100, para 9.

4.5 The manual release system shall consist of local and remote release handles which shall be connected, by suitable means to the compartment cover locking device and to the liferaft pull cable. A pull on any release handle shall simultaneously release the compartment cover, actuate the gas filled cylinder on the stowed liferaft, thereby inflating the raft, and eject the liferaft. The local and remote manual release handles may be connected in series and shall be identified and protected so that they cannot be confused with the other controls or operated inadvertently. Jamming of the remote control shall not prevent functioning of the manual release. Release handles shall be provided as follows:-

(i) Adjacent to the ditching exit and easily accessible to a person, both immediately before and after passing through the exit.

(ii) Adjacent to the liferaft stowage and capable of being operated by a person standings on the ditched rotorcraft.

(iii) Capable of being operated by a person in the sea in the vicinity of the escape hatch unless (i) or (ii) meets this requirement.

(iv) There shall be at least one convenient release means inside the rotorcraft by which all remotely controlled liferafts can be released. In addition, release means on the outside of the rotorcraft and accessible to survivors in the water shall be provided.

(v) Release systems shall be safeguarded against spontaneous or inadvertent operation in any condition of flight, and shall be so designed that it is impossible to operate the release partially and return the control to its normal position, without it being obvious that the release has been partially operated.

4.6 It shall be possible to easily remove the liferaft from its stowage manually without the use of tools and then to inflate the liferaft by means of the bellows which are provided.

4.7 The position of the release handles shall be marked in accordance with the appropriate identification requirements (see Chapter 103 para 4) and easy identification in total darkness is essential. It shall also be borne in mind that the Rotorcraft may even be inverted after ditching, and the occupants disorientated. The pull required to operate any of the liferaft releases shall not exceed 135 N under the most adverse conditions.

4.8 A liferaft released by remote control shall be attached to the rotorcraft by its line. This line, however, shall be incapable of submerging or capsizing the liferaft if the rotorcraft sinks with the line still attached. The effects of wind on the raft with any inflatable canopies erected should be taken into account when deciding the breaking strength and length of the line. Consideration shall also be given to the load involved if it is intended to tow the liferaft.

4.9 After launching it shall be possible to hold liferafts in such a position that the likelihood of the occupants of the rotorcraft being immersed while boarding them is reduced to a minimum.

5 STRENGTH

5.1 The liferaft installation shall comply with the strength requirements for the rotorcraft as a whole, including those for, crashlanding and in particular, the loads resulting from inertia, preloading, internal and external aerodynamic forces and pressure. Altitude loading due to residual air in the liferaft shall be considered.

6 INFLATION CYLINDER HIGH TEMPERATURE DISCHARGE INDICATOR

6.1 An indicator shall show if the liferaft inflation cylinder has discharged itself to atmosphere. This unit shall normally be situated in the cover of the liferaft stowage but if the indicator would not be readily seen in that position it can be fitted in some position slightly remote from the liferaft stowage. Consideration shall be given to displaying a warning indicator in the flight deck console.

7 LIFERAFTS STOWED IN VALISES

7.1 In particular circumstances on transport rotorcraft, liferafts stowed in valises may be accepted in addition to or in lieu of the pre-packed stowage type of installation. In such cases the liferaft valise shall be stowed in a designated compartment so as to be readily accessible after ditching and a suitably placed anchorage shall be provided for the static line. This line shall be incapable of submerging or capsizing the loaded liferaft when the rotorcraft sinks.

7.2 Internal liferaft stowages shall be so arranged that adequate space is available for access to and easy manipulation of the liferaft package prior to launching.

7.3 When launching is by hand, the necessary actions shall be within the capacity of an untrained person of average strength, and shall not require exceptional agility or skill. This is to cover the use of rotorcraft in the transport role.

7.4 The overall dimensions of the liferaft package shall be such that it can be easily passed through any emergency exit likely to be used for launching.

7.5 A suitable line which can be attached to the rotorcraft prior to launching is assumed to be provided as part of the liferaft.

8 ENGINEERING DATA

8.1 Engineering drawings and other data required by the detail specification for a particular rotorcraft shall include the following:-

(i) Location of all the liferaft compartments in the rotorcraft.

(ii) Design of the cover and locking device.

(iii) Provisions for the removal of the compartment cover for inspection of the liferaft and equipment without operating the rotorcraft release systems and the stowed liferaft's inflation system.

(iv) Design of the local and remote manual release systems and the force and amount of travel required to operate each manual release.

(v) Design of the basic components of the automatic release system.

(vi) Description of any electrical installation.

(vii) Signs or labels detailing operating and servicing instructions and warnings.

(viii) Internal volume of each liferaft compartment.

9 GROUND TESTS

9.1 The Contractor shall demonstrate by ground tests that:-

(i) The pre-packed container and release mechanism unit can be fitted in accordance with para 3.2.

(ii) The stowage cover can be locked safely in position using only light hand pressure and,

(iii) The liferaft inflates correctly and unless it is of the reversible type launches the right way up on operation of any of the automatic or manual releases.

LEAFLET 721/0

EMERGENCY LIFERAFT INSTALLATIONS

REFERENCE PAGE

<u>RAE Technical Notes</u>

SME 192 .. .. .. Determination of loads arising on liferaft stowage retaining devices in flight.

LEAFLET 721/1

EMERGENCY LIFERAFT INSTALLATIONS

GENERAL RECOMMENDATIONS

1 INTRODUCTION

1.1 This leaflet gives information on the multi-seat liferafts in prepacked stowages developed by MOD(PE) and makes recommendations for their installations in rotorcraft.

1.2 Designers are free to design their own liferaft stowages if they wish, but to meet the requirements of Chapter 721 they must be of the pre-packed type.

2 GENERAL

2.1 Emergency liferafts are fragile pieces of equipment and are damaged extremely easily, thus, a pre-packed liferaft container or pan has accordingly been developed and is available. It consists of a removable box containing the liferaft and its equipment complete, with the operating mechanism separately mounted. There is provision for the stowage cover to be fitted without the pre-packed stowage so that the rotorcraft can be flown with the liferaft removed. These items can all be removed from the rotorcraft in a matter of minutes without the use of tools. Servicing of the equipment and unpacking and packing of the liferaft can then be carried out in ideal conditions after removal to the safety equipment section, and a serviceable replacement set can be fitted to the rotorcraft with comparative ease.

3 LOCATION OF LIFERAFTS

3.1 The overriding consideration in the choice of location of the liferaft stowage is to ensure safe ejection and inflation of the liferaft and ease of boarding for the crew. To this end the following points are relevant:

(i) The position should be such that the risk of damage due to an internal surge of water during, ditching is minimised.

(ii) The liferaft should be ejected onto the water as near as possible to an emergency exit which is likely to be well clear of the waterline after ditching.

(iii) There should be no risk of the liferaft being damaged or trapped by aerials, hot jet pipes, or other projections during ejection.

3.2 The possibility of leakage of fuel or oil into the stowage should not be overlooked as both have a rapid and highly detrimental effect on liferaft fabrics. A position should therefore be chosen which minimises this risk and when possible, advantage should be taken of the protection afforded by existing airframe structure to meet the possible case of the entry of fuel or oil due to enemy action.

3.3 The whole liferaft installation should be readily accessible for examination and servicing. The need for easy access to carry out a routine inspection is particularly important.

4 LIFERAFT RELEASE

4.1 The automatic release should normally be by means of a submersion actuator, which when immersed in water operates on the electrolytic principle. The actuator, upon immersion, should close a circuit, permitting electricity to flow and actuate the release system. The system designer should be aware that the voltage output from an electrolytic cell varies in proportion with the salinity of the water in which it is immersed, thus when selecting the minimum voltage necessary to initiate system operation due cognisance should be taken of the voltage output from the cell when immersed in fresh water. The submersion actuator should be attached to an "always live" bus of the primary rotorcraft electrical system so that the actuator is always ready to complete the circuit, by permitting electricity to flow and actuate the release system at a time of emergency. The time taken to reach sufficient voltage can be varied by design and should be arranged so that release does not occur prematurely but nevertheless takes place as rapidly as possible after the rotorcraft has stopped following ditching.

4.2 In addition the submersion actuator should have its own source of power so that in the event of failure of the primary electrical system the actuator will function. The submersion actuator should be located inboard, at the lowest possible point below the flotation waterline of the rotorcraft. The actuator should be identified and protected against inadvertent operation. The manner of the protection should not interfere with the operation of the actuator nor prevent its inspection.

4.3 The release handles should be easily identifiable and this can be assisted by locating them near some prominent structural feature of the rotorcraft. Care should be taken to ensure that unintentional operation of the liferaft release control is avoided and that no dangers of inadvertent release can result from expansion effects in the control circuits due to temperature changes or flexing of the airframe structure.

5 GAS CYLINDER DISCHARGE INDICATOR UNIT

5.1 This system normally incorporates a diaphragm which ruptures when the gas bottle discharges to leave a visual indication of the bottle discharge. The units should normally be mounted in the cover of the liferaft stowage, but if the indicators would not be readily seen in that position they can be fitted in some position slightly remote from the liferaft stowage. They should not be located where engine exhaust gases can impinge upon them or where they can be obscured by oil or other contaminants.

6 GAS CYLINDER SEEPAGE DIVERSION VALVE

6.1 For protection against the possibility of a slow leak from the gas cylinder valve prematurely inflating the raft, consideration should be given to a suitable additional valve which, at low pressure, would divert the gas to atmosphere.

7 LIFERAFTS STOWED IN VALISES

7.1 When liferafts stowed in valises are permitted by the Service staff care should be taken to choose a location that will permit manhandling through a suitable exit. When a number of valises are carried they should be distributed between various ditching exits.

7.2 The liferaft emergency pack should be stowed within the appropriate valise and as this has to be manhandled within the rotorcraft in an emergency, it should be noted that the combined weight of the two may be as much as 55 kg.

7.3 The valises should preferably be secured within a compartment of appropriate size, without the need for special securing straps but, if the latter are used an appropriate type of quick release should be provided which can be readily undone when wet.

CHAPTER 722

FOLDING COMPONENTS

1 INTRODUCTION

1.1 The requirements of this chapter apply, unless otherwise specified, to all folding components, e.g. Rotor blades, tail unit, to meet their specified limiting rotorcraft dimensions.

2 WIND SPEED CONDITIONS

2.1 It shall be possible to fold or spread any component in wind speeds up to 45 knots from any quarter. For Naval rotorcraft see the requirement of Chapter 100 para 16.

3 METHODS OF OPERATION

3.1 Folding and spreading may be by hand or by power operation, as required by the rotorcraft specification. If the latter is specified, then

(i) At the specified normal load, either operation shall be capable of completion within the specified time, either with the engine running at the minimum governed condition for accessory drive, or with an APU, if fitted, and

(ii) Folding and spreading shall be possible by an alternative means when the normal source of power from engine driven units and power accumulators are not available. Alternative means of folding or spreading should make provision for independent operation of blades and tail unit to facilitate servicing.

4 OPERATING CONDITIONS

4.1 It shall be possible to fold and spread the main rotor and tail unit with any normal armament stores in place and with any quantity of fuel in the tanks, including auxiliary and drop tanks when fitted, without loss of fuel.

4.2 The design shall be such that no fouling of moving parts (e.g. locks) can occur during folding or spreading.

4.3 The design shall be such that damage and wear occurring from repeated folding and unfolding cycles must be minimal or, if possible, avoided altogether.

5 LOCKS

5.1 The design shall provide for the securing of the component in the spread position so that it satisfies the strength requirements specified for the rotorcraft as a whole. (see also Chapter 309 para 2.4)

5.2 The device for locking the component in the spread position shall be such that it cannot unlock as a result of any technical fault or defect, vibration, or rotorcraft acceleration. The design shall ensure that accidental reversal of forces or a tendency to creep towards the unlocked position cannot occur.

5.3 Means shall be provided for holding the component securely (e.g. by a jury strut or built-in lock) when transporting the rotorcraft or when power is not available).

5.4 The provision of jury struts/blades support gear shall not require the use of platforms or ladders to fit, and the need for personnel to climb along rotorcraft structure shall be avoided.

6 CONTROL AND INDICATORS

6.1 The controls and indicators shall be in accordance with Chapter 107, Table 4, Item 2.

7 STRENGTH

7.1 An ultimate factor not less than 1.5 shall be achieved on the loads which arise from folding and spreading while the rotorcraft is standing on the deck or ground under the wind speeds specified in para 2.1.

CHAPTER 723

RESCUE HOIST AND SONAR HOIST INSTALLATIONS

1 INTRODUCTION

1.1 This Chapter states the design requirements for the installation of Sonar and Rescue hoist equipment in all rotorcraft. Sonar and Rescue Hoist equipment for the purpose of these requirements, comprise the Sonar and Rescue hoist installations associated hydraulic and electrical circuits and components concerned with powering, controlling, and monitoring the equipment to ensure safe rotorcraft operation.

1.2 If an auxiliary structure is provided and this is detachable, the requirements apply to the auxiliary structure and its attachments to the rotorcraft.

1.3 Existing, in-service equipment shall be utilised wherever possible provided the system performance is not prejudiced.

2 STRENGTH

2.1 The rotorcraft as a whole and that part of the load suspension system associated with the rotorcraft shall have proof and ultimate factors of not less than 1.5 and 2.0 respectively on the heaviest load to be carried.

2.2 The factored forces of para 2.1 shall be assumed to be applied to the rotorcraft in any direction within a cone of semi-angle 30° having its apex at the suspension point on the rotorcraft and its axis normal to the rotorcraft horizontal axis.

2.3 The effects of dynamic, aerodynamic and turbulence excitations on both the load and the rotorcraft shall be considered with respect to strength, stability, controllability and handling of the rotorcraft.

3 FATIGUE

3.1 The fatigue life of those parts of the rotorcraft structure subjected only to the forces arising from the carriage of rescue and/or sonar hoist loads and of the removable auxiliary structure, if provided, shall be established in accordance with the requirements of Chapter 201, to a load spectrum to be agreed with the Rotorcraft Project Director (in conjunction with the Airworthiness Division, R.A.E.) The fatigue life of the fixed parts shall be at least equal to the specified life of other parts of the rotorcraft structure. The fatigue life of the removable auxiliary structure, if provided, shall be agreed with the Rotorcraft Project Director (in conjunction with Airworthiness Division, R.A.E.).

3.2 When formulating fatigue spectra for other parts of the rotorcraft, the effects of carriage of rescue and/or sonar hoist loads shall be taken into account to the extent agreed with the Rotorcraft Project Director (in conjunction with Airworthiness Division, R.A.E.).

4 ENVIRONMENTAL

4.1 The rescue and/or sonar hoist installations shall satisfy the environmental requirements for the rotorcraft stated in the Rotorcraft Specification or agreed with the Rotorcraft Project Director.

4.2 Provision shall be made to minimise the ingress of water, particularly from sonar hoists. Any provision to contain ingressed water shall be provided with adequate drainage. (see Leaflet 711/3).

5 INSTALLATION

5.1 The layout of the completed installation shall be such that no part of the installation, loading apparatus or equipment will foul any other part of the installation, loading apparatus, equipment or the rotorcraft structure under the normal conditions of loading, operation or unloading. In the case of ship-borne rotorcraft these requirements shall be met under the conditions for securing of naval rotorcraft given in Chapter 309, with the rotorcraft heading in any direction.

5.2 The position of the rescue and/or sonar hoist installation for any of the specified combinations shall be chosen to minimise undesirable rotorcraft trim changes during operation and loading.

5.3 When the rotorcraft has folding components (e.g., rotor, fuselage, tail) it shall be possible to fold and spread these components with rescue and/or sonar hoist(s) fitted.

5.4 It shall be possible to load and install the rescue and/or sonar hoist(s) with the folding components folded.

5.5 Adequate clearance shall be provided for the loading and unloading of equipment with the rotorcraft at maximum all up weight and its alighting gear compressed and tyres deflated in the most adverse manner. Allowance for uneven surface conditions shall be made.

5.6 Adequate ground clearance between rescue and sonar hoist installations to cater for rotorcraft landings in the most adverse conditions shall be provided.

5.7 Adequate escape routes for occupants shall be provided with sonar and/or rescue hoists installed.

6 ACCESSIBILITY

6.1 The layout of the complete installation shall be such that it is possible to carry out easily and quickly all necessary equipment fits.

6.2 All parts of the installation requiring inspection and servicing shall be readily accessible.

6.3 It shall be possible to carry out all servicing and inspection wearing the appropriate protective clothing.

7 EMERGENCY OPERATION

7.1 Means shall be provided for emergency jettison of rescue and sonar hoist cables.

7.2 Control of emergency jettison shall be available to both pilot and equipment operator.

7.3 All switches for control of emergency jettison shall be guarded. (see also Chapter 105, para 11.5).

7.4 It shall be possible to operate the jettison circuits correctly despite a complete failure of the normal generating system.

7.5 Jettison systems shall be so engineered that no single functional failure can cause inadvertent cable jettison or prevent cable jettison when required.

8 CONTROL HANDLE AND CONTROL COLUMN ELECTRICAL CIRCUITS

8.1 Rescue and sonar hoist cable jettison electrical circuits in control handles and control columns shall be designed and manufactured to prevent faults leading to inadvertent jettison. The close proximity of standing and switched voltages for rescue and/or sonar hoist and non-hoist services in these locations, makes it necessary to ensure that the hoist circuits are properly segregated from all other services.

9 COMMUNICATIONS

9.1 Intercommunication shall be provided between the crew members and pilot(s).

10 COMPLIANCE WITH OTHER DEF STAN 00-970 VOLUME 2 CHAPTERS

10.1 Rotorcraft rescue and/or sonar hoist(s) installations shall comply with the requirements of the following chapters as appropriate.

10.1.1 Chapter 704 - Hydraulically powered hoists and hoist systems.

10.1.2 Chapter 706 - Electrically powered hoists and hoist systems.

10.1.3 Chapter 703 - Pneumatically powered hoists and hoist systems.

10.1.4 Chapter 718 - Explosive devices.

10.1.5 Chapter 708 - Bonding and screening.

10.1.6 Chapter 711 - Ice protection, in particular para 2.3.2, 2.6.1 and Leaflet 711/3.

10.1.7 Chapter 714 - Role equipment installation, particularly paras 2.1.3, 2.4, 2.8, 2.9, 3.6 and 3.8.

10.1.8 Chapter 722 - Folding hoist installations.

10.1.9 Chapter 105 - Layout of crew stations.

11 TESTING

11.1 Consideration shall be given to the need for model tests, and proposals shall be agreed with the Rotorcraft Project Director.

11.2 Flight testing shall be in accordance with Chapter 1017.

CHAPTER 724

INSTRUMENT/DISPLAY INSTALLATIONS

1 INTRODUCTION

1.1 This Chapter states the requirements for the installation of rotorcraft instruments and displays. Note the term instruments or displays used throughout this document shall include associated remote sensors, ancillary equipment and fittings.

2 MECHANICAL REQUIREMENTS

2.1 The case type for a particular instrument/display shall, where practicable, be selected from the standard range specified in DEF STAN 66-26 (Part 1).

2.2 Flanged cases are normally mounted from the front of the instrument panel. Mounting shall be by means of screws through the corner ears of the flange into fixed locknuts at the back of the panel. For flangeless cases suitable clamps in accordance with DEF STAN 53-96 (in preparation from STANAG 3492) shall be used around the periphery of the case. Quick release fasteners shall be considered for face-up horizontal, or nearly horizontal, panels. Long or heavy cases may require additional support at the rear of the panel.

2.3 Certain cases may be mounted from the rear of the panel. In these instances the fixing holes in the corner ears of the case shall be replaced by integral locking nuts, which shall have unified or metric threads. A warning notice as defined in DEF STAN 66-26 (Part 1) para 4c shall be displayed to identify unified threads. A notice of similar dimensions as depicted in DEF STAN 05-13 Annex D Type 1 shall be displayed to identify metric threads.

2.4 Positioning of instruments/displays in the rotorcraft shall be in accordance with the requirements of Chapters 105, 106 and 107.

2.5 CONNECTIONS

2.5.1 Electrical

(i) Connectors shall be selected as practicable from suitable types listed in DEF STAN 59-35 (Part 0) Certain circular connectors therein are approved specifically for airframe fit and avionics applications. The type or types selected shall be agreed with the Rotorcraft Project Director.

(ii) The rotorcraft manufacturer shall ensure that adjacent connectors cannot be incorrectly mated on any one instrument or on adjacent instruments.

(iii) Where free connectors are used there shall be a minimum 0.75 in (20mm) radial clearance between any two adjacent connectors.

(iv) Where a flying lead with connector is used it shall be of sufficient length for the instrument to be withdrawn to clear the panel. The lead shall be securely stowed when the instrument is fitted.

(v) Bonding shall meet the requirements of Chapter 708.

2.5.2 Air. Pitot and static connections shall be made in accordance with the requirements of Chapter 716 and DEF STAN 66-27

2.5.3 Hydraulic. Connections to hydraulic instruments and gauges shall be in accordance with Chapter 704 and DEF STAN 66-27.

2.5.4 Pneumatic. Connections to pneumatic gauges and instruments shall be in accordance with Chapter 703 and DEF STAN 66-27.

2.6 LIGHTING

2.6.1 Instrument and panel lighting shall meet the requirements of Chapter 105, para 15.

2.6.2 The Rotorcraft Project Director shall define the requirements for Emissive Displays monochrome or multicolour.

2.6.3 Where Night Vision Goggles are to be used cockpit lighting shall comply with the compatibility requirements defined by the Rotorcraft Project Director.

2.7 INFORMATION PRESENTATION

2.7.1 The Presentation of Instrument Information shall meet the requirements of DEF STAN 66-26 (Part 5 and 6). Pointer quantity information shall increase with clockwise movement of the pointer, from left to right on horizontal scales and from bottom to top on vertical scales. An exception can be made where round scales are mirrored (e.g., brake indicator); the left scale may have counter clockwise movement with increase in pointer quantity information.

2.7.2 The above principles of presentation shall apply to Electronically and/or Optically generated displays. The requirements for Electrically and/or Optically generated displays will be stated in DEF STAN 66-28 (in preparation), but in the meantime will be defined by the Rotorcraft Project Director. They will include:

(i) The location of display information.

(ii) General symbology for display flight and combat information.

2.7.3 Units of displayed information shall be the same as the units of measurement applying in other aircrew instruments in the same rotorcraft. Rotorcraft requirements for such units of measurement shall be stated by the Rotorcraft Project Director.

2.7.4 The display shall be clearly readable in all lighting conditions. Guidance in the testing of displays for this requirement is provided in Leaflet 724/1. Alternative fonts can be used[1].

3 POWER SUPPLIES

3.1 Primary power supplies are listed in Chapter 706, Table 1.

3.2 The characteristics of electrical power supplied to the terminals of airborne equipment are specified in BS 3G100, Part 3.

3.3 Where a particular instrument/display adversely affects the characteristics of the power supplies to other equipment, the effects shall not exceed the Limits of BS 3G100, Part 3.

3.4 Instrument/display/systems shall be capable of withstanding without damage the interruption and transients specified in BS 3G100, Part 3.

3.5 IDENTIFICATION OF CONNECTING WIRES

3.5.1 Circuits and cable runs shall be identified in accordance with Chapter 806, para 7.1.

3.5.2 Instrument terminals shall be identified with letters A, B, C to define the phases of the ac supply.

4 ENVIRONMENTAL CHARACTERISTICS

4.1 VIBRATION

4.1.1 Instruments/displays shall be designed to operate under the vibration conditions which prevail in their particular allocated position in the rotorcraft.

4.1.2 Where the levels are appropriate compliance with para 4.1.1 shall be established by testing to BS 3G100, Part 2, Section 3, Subsection 3.1.

4.1.3 Where the levels prescribed in para 4.1.2 are inappropriate then specific test levels shall be defined by the rotorcraft manufacturer.

4.2 ACCELERATION

4.2.1 Normal Conditions. The instruments/displays shall continue to function during, and after the appropriate normal flight conditions specified by BS 3G100, Part 2, Section 3, Subsection 3.6 or the maximum conditions envisaged for the rotorcraft in flight manoeuvres. This shall be demonstrated by appropriate tests with the instruments/displays mounted in a representative condition.

4.2.2 Angular Acceleration. This requirement is concerned with the Accident Data Recorder (ADR) and any instrument or piece of equipment which uses angular motion in any of its working parts and which supplies information to the pilot, or the ADR, during the period leading up to an accident or incident. Designers of such instruments shall determine from the rotorcraft manufacturer the likely severity of such angular acceleration and establish by calculation and by testing that the instrument/displays will continue to function during or after this acceleration, as appropriate.

4.3 TEMPERATURE AND PRESSURE

4.3.1 Instruments/displays shall function satisfactorily in the temperature and pressure conditions that prevail in their local environment throughout the operational range of the rotorcraft. The instrument/display designer shall undertake the thermal analysis of the instruments/displays and inform the rotorcraft manufacturer of the heating/cooling requirements.

4.3.2 The rotorcraft manufacturer shall provide heating/cooling systems, as appropriate, in accordance with Chapter 101, para 5 and shall be responsible for the overall thermal management of the local environments.

4.3.3 The thermal management of heating or cooling shall ensure that acceptable temperatures are maintained for correct functioning in all conditions of operation.

4.3.4 Essential instruments/displays as defined by the Rotorcraft Project Director shall continue to function satisfactorily for an agreed period after a failure of any air conditioning system. Other instruments/displays shall fail "safe" in the event of a failure of the air conditioning systems.

4.3.5 Instruments/displays whose operation is a function of dynamic pressure shall have been calibrated in accordance with BS G199[2].

4.4 HUMIDITY

4.4.1 Instruments/displays shall be designed to function satisfactorily within the humidity limits of Chapter 101, para 2. Compliance shall be established by testing to the requirements of BS 3G100 Part 2, Section 3, Sub-section 3.7. Tropical exposure tests.

4.5 DIRECT EXPOSURE TO SUNLIGHT

4.5.1 Instruments/displays likely to be exposed to direct sunlight shall not be degraded by such exposure. In particular materials and panel information markings shall not deteriorate or fade. Compliance shall be established by testing to DEF STAN 07-55 (Part 2), Section 2, Test B3. The rotorcraft manufacturer shall state the severity and duration of such tests.

4.5.2 The intensity of illumination of all instruments/displays shall be controlled in accordance with the requirements of Chapter 105, paras 15.8 and 15.9.

4.5.3 Electronically generated displays including Head Up (HUD's) and Multi-Function (MFD's) shall be clearly defined when superimposed against a background luminance to be defined by the Rotorcraft manufacturer/Project Director. See Leaflet 724/1.

4.6 DUST AND SAND

4.6.1 Chapter 101, para 7 states the dust and sand proofing requirements for the rotorcraft. Whilst dust and sand proofing of the rotorcraft fuselage may be a desirable aim there may be occasions when the proofing is insufficient or not possible.

4.6.2 DEF STAN 07-55 (Part 2) Section 4 specifies test methods for the ingress of dust and sand.

4.6.3 The Rotorcraft Project Director shall state the tests required and the severity of the tests. The instruments/displays shall function satisfactorily during and after these tests.

4.7 FUNGAL CONTAMINATION

4.7.1 DEF STAN 00-29 provides particulars of fungal germination and growth on material. Methods of preventing or limiting fungal growth on material are also given.

4.7.2 Use of materials that may support fungal growth shall be avoided. Instruments/displays shall not be damaged or deranged and shall continue to function in the presence of fungal growth.

4.7.3 Tests. DEF STAN 07-55 (Part 2) Section 3 Test C1 specifies tests to determine the extent of fungal growth under short term exposure and assess the effect of a fungus growth on the functioning of the Instruments/Displays by longer exposure.

4.7.4 The Rotorcraft Project Director shall state the tests required and the severity of the tests.

4.8 CONTAMINATION FROM FLUIDS

4.8.1 Instruments/displays and installations in certain parts of the rotorcraft may be subjected to contamination from fluids used in their particular location in the rotorcraft.

4.8.2 BS 3G100, Part 2, Section 3, Sub-section 3.12 lists groups of fluids likely to be encountered in rotorcraft and details tests to be carried out on equipment. Only where items are likely to be contaminated by the fluid should contamination tests be carried out.

4.8.3 The Rotorcraft Project Director shall state the tests required and the severity of the tests.

4.9 CHEMICAL ATTACK

4.9.1 Information concerning chemical attack is provided by DEF STAN 00-50.

4.9.2 Should there be any likelihood of chemical attack within the rotorcraft then tests will be made. Procedures for tests are specified in DEF STAN 07-55 (Part 2), Section 3.

4.9.3 The Rotorcraft Project Director shall state the tests required and the severity of the tests.

4.10 HANDLING TESTS

4.10.1 Instruments/displays shall be sufficiently robust to withstand handling and transit shocks.

4.10.2 Test procedures for Bump (rough ride in the back of a four wheeled vehicle without packaging), bench handling and topple tests are stated in DEF STAN 07-55 (Part 2) Section 1, Test A5 Bump Test A4 Drop and Topple Test A3 Shock.

4.10.3 The Rotorcraft Project Director shall state the tests required and the severity of the tests.

4.11 ELECTROMAGNETIC COMPATIBILITY

4.11.1 Requirements applicable to the limitation of propagated electromagnetic energy whether radiated or conducted and to the limitation of susceptibility of equipments, installations and systems to such energy are stated in DEF STAN 59-41. Further information and requirements particular to rotorcraft are stated in AvP 118.

4.11.2 Methods and conditions for testing are stated in DEF STAN 59-41 (Part 3) (and part 4 when published) and in BS 3G100 Part 4, Section 2.

4.11.3 The Rotorcraft Project Director shall state the tests required and the severity of the tests. These will include testing of equipments, subsystems and systems of the rotorcraft.

4.12 EXPLOSIVE ATMOSPHERES

4.12.1 Instruments/displays which may come into contact with flammable vapours shall meet the requirements of Chapter 712, para 6.

4.13 NUCLEAR HARDENING

4.13.1 Consideration shall be given to nuclear hardening (Chapter 717 is being revised).

REFERENCES

Reference	ASCC Air Std	STANAG	BS
1	-	3329	-
2	10/46	-	G199

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INSTRUMENT/DISPLAY INSTALLATIONS

REFERENCE PAGE

British Standards

N 100	General design requirements for aircraft oxygen systems and equipment
3G 100	Specification for general requirements for equipment for use in aircraft
G199	Schedule for tables relating to altitudes, airspeed and Mach numbers for use in aeronautical instrument design and calibration
2011	Basic environmental testing procedures

Defence Standards

00-10	General design and manufacturing requirements for service electronic equipment
07-55	Environmental testing of service material
53-96	Clamps, mounting (imperial and metric) for aircraft instruments
59-35 (Part 0)	Guide to connectors, electrical, and their application Part 0: Connectors, electrical for dc and low frequency application
59-41	Electromagnetic compatibility
61-7	Identification of electrical and electronic systems wiring and components
66-26	General requirements for aircraft instruments and display
66-27	Pressure connections on aircraft instruments and associated equipment
66-28	Electronically and/or optically generated aircraft displays for fixed wing aircraft (when available)

US Mil Specifications

MIL-D-81641(AS) ..	Display, head-up, general specification
MIL-D-87213(USAF)..	Displays, airborne, electronically optically generated

MOD(PE) Publications

AvP 118	Guide to electromagnetic compatibility in aircraft systems

LEAFLET 724/1

INSTRUMENT/DISPLAY INSTALLATIONS

EXPOSURE TO SUNLIGHT

1 SPECIFICATION AND TESTING REQUIREMENTS

1.1 Under certain conditions direct or reflected sunlight may provide a background luminance that submerges the information symbology on electronic display screens.

1.2 Guidance in the specification and testing requirements of such displays will be found in the following U.S. Military Specifications.

MIL-D-81641 (AS) - DISPLAY, HEAD-UP - General Specification

MIL-D-87213 (USAF) DISPLAYS, AIRBORNE,
ELECTRONICALLY/OPTICALLY GENERATED.

CHAPTER 725

AVIONIC EQUIPMENT INSTALLATIONS

1 INTRODUCTION

1.1 This Chapter states the requirements for the installation of the rotorcraft avionic equipment. Whenever possible, standard items shall be used throughout the installation in accordance with the requirements of Chapter 706. In addition to the requirements of Chapter 706, reference shall be made to the design requirements of DEF STAN 00-10.

NOTE: The term equipment used throughout this document shall include remote sensors, ancillary equipment and fittings.

1.2 Leaflet 725/2 gives guidance and advice on the transmission of data and inter-aircraft compatibility of data transfer.

2 PHYSICAL CHARACTERISTICS

2.1 SIZE

2.1.1 Where possible standard items shall be used provided their use is consistent with efforts to minimise weight. Where non standard items are used they shall be designed to a minimum size and mass consistent with the specified performance, mechanical strength, reliability, maintainability and project economic philosophy. Electronic Line Replaceable Items (LIRs) shall be hand manageable by one man. (See Chapter 804, para 6.3).

2.2 SHAPE

2.2.1 The Rotorcraft Designer/Project Director shall state the preferred equipment case design specification. Normally, cases will be rectangular hexahedral in shape and the content will be of modular construction. (Leaflet 725/3).

2.2.2 Where the shape of the equipment cannot be configured to a standard specification, due to lack of space, the particular envelope shape of the rotorcraft or ergonomic considerations, the limiting requirements shall be defined by the Rotorcraft Designer/Project Director.

2.3 PHYSICAL/MECHANICAL INTERFACE

2.3.1 Equipment shall be mounted to meet the crashlanding requirements of Chapter 307, para 1.1 and the vibration requirements of para 6.1.1.

2.3.2 Equipment shall be secured in place by means of hold-downs or quick release fasteners subject to meeting the requirements of para 2.3.1.

2.3.3 Pitot and Static Connections shall be made in accordance with the requirements of Chapter 716, and DEF STAN 66-27.

2.3.4 Equipment shall be designed to meet the requirements of Chapter 100, para 7. (Incorrect Assembly).

2.3.5 Moisture traps shall be avoided where practicable in accordance with DEF STAN 00-10. Provision shall be made for drainage of moisture traps where these are unavoidable.

2.3.6 Consideration shall be given to the capability to withstand secondary damage i.e., in the event of any form of failure of an equipment, interfacing equipments shall not incur damage.

2.3.7 Interchangeability requirements shall be in accordance with DEF STAN 00-10.

2.3.8 Elapsed Time Indicators (ETI) shall be fitted, when applicable, in all principal LRIs so that indications can be easily seen when equipment is fitted with covers installed.

2.4 ELECTRICAL INTERFACE

2.4.1 Requirements for the electrical installations in rotorcraft are stated in Chapter 706. The types of connector to be used shall be specified by the Rotorcraft Project Director to meet the functional and environmental requirements of the rotorcraft. For guidance see Leaflet 725/3.

2.4.2 Where connectors are used on free cables from the rotorcraft or the equipment, means shall be provided to prevent mismating with similar adjacent connectors.

2.4.3 Electrical Bonding shall meet the requirements of Chapter 708.

2.4.4 Where free connectors are used there shall be a minimum 20mm radial clearance between any two connectors when connected.

2.4.5 Conformal coatings shall be avoided where possible but, if they are used they shall be non toxic and easy to remove and replace.

2.4.6 Volatile memories, if an essential part of the design, shall be provided with a means of data retention during power interruptions (see Chapter 706, para 2.7.3 and Leaflet 725/3).

3 TESTABILITY AND BUILT IN TEST (BIT)

3.1 TESTABILITY

3.1.1 Testability is an essential element of equipment design. See DEF STAN 00-13.

3.2 BIT

3.2.1 The requirement for BIT will be stated in the Rotorcraft Specification or by the Rotorcraft Project Director as defined in Chapter 800, para 9. Among the items to be considered are:

(i) Fault detection and location down to LRI or module level including inter LRI module connectors.

(ii) Indication of fault status during system operation, in the air and on the ground.

(iii) Recording of fault data during operation, in the air and on the ground, for maintenance information.

(iv) Provision for testing shall be so designed that any failure of the BIT shall not degrade the equipment operation or cause the equipment to shut down. (See DEF STAN 00-13).

(v) The maximum reduction as a percentage of the Mean Time Between Failure (MTBF) due to the inclusion of BIT (see Leaflet 725/3).

(vi) Minimising of the false alarm rate (see Leaflet 725/3).

(vii) The probability rate of BIT detecting faults.

3.3 AUTOMATIC TEST EQUIPMENT (ATE)

3.3.1 The installation shall facilitate the use of ATE where required by the Rotorcraft Specification or the Rotorcraft Project Director. The use of ATE will depend on the maintenance philosophy adopted and the levels of testing envisaged.

4 A FAILURE MODE EFFECTS AND CRITICALITY ANALYSIS (FMECA)

4.1 When required by the Rotorcraft Project Director an FMECA shall be conducted to establish the effects of failure within the systems. Where practical those items liable to failure, or critical in the function of the systems, shall be readily accessible and replaceable.

5 POWER SUPPLIES

5.1 Primary power supplies are listed in Chapter 706, Table 1.

5.2 The characteristics of electrical power supplied to the terminals of airborne equipment are specified in BS 3G100, Part 3.

5.3 Where a particular equipment adversely affects the characteristics of the power supplies to other equipment, the effects shall not exceed the limits of BS 3G100, Part 3.

5.4 Utilisation equipment/systems shall be capable of withstanding without damage the interruption and transients specified in BS 3G100, Part 3.

5.5 IDENTIFICATION OF CONNECTING WIRES

5.5.1 Circuits and cable runs shall be identified in accordance with Chapter 806, para 7.1.

5.5.2 Equipment terminals shall be identified with letters A, B, C for the phasing of the ac supply.

6 ENVIRONMENTAL CHARACTERISTICS

6.1 VIBRATION

6.1.1 Equipments shall be designed to function under the vibration conditions which prevail in their particular position in the rotorcraft.

6.1.2 Where the levels are appropriate compliance with para 6.1.1 shall be established by testing to Chapter 501, para 4.

6.1.3 Where the levels prescribed in para 6.1.2 are inappropriate, then specific levels shall be defined by the rotorcraft manufacturer.

6.2 ACCELERATION

6.2.1 Normal Conditions. The equipment shall continue to function during and after the extreme conditions experienced by the rotorcraft in flight manoeuvres or in the absence of suitable information the appropriate normal flight conditions specified by BS 3G100, Part 2, Section 3, Subsection 3.6, Table 1. This shall be demonstrated by appropriate tests with the equipment mounted in a representative condition.

6.2.2 Angular Acceleration. This requirement is concerned with any piece of equipment sensitive to angular acceleration. Designers of such equipment shall determine from the rotorcraft designer the likely severity of such angular acceleration and establish by calculation and by testing that the equipment will continue to function during and after this acceleration, as appropriate.

6.3 TEMPERATURE AND PRESSURE

6.3.1 Equipment shall function satisfactorily in the temperature and pressure conditions which prevail in its local environment throughout the operational range of the rotorcraft. The equipment designer shall undertake the thermal analysis of the equipment and inform the rotorcraft designer of the heating/cooling requirements. (See Leaflet 725/1).

6.3.2 The rotorcraft manufacturer shall provide heating/cooling systems, as appropriate, in accordance with Chapter 101, para 5 and shall be responsible for the overall thermal management of the local environments.

6.3.3 The thermal management of heating or cooling shall ensure that acceptable temperatures are maintained for correct functioning in all conditions of operation.

6.3.4 Essential equipment as defined by the Rotorcraft Project Director shall continue to function satisfactorily for an agreed period after a failure of any air conditioning systems. Other equipment shall fail "safe" in the event of a failure of the air conditioning systems.

6.4 HUMIDITY

6.4.1 Equipment shall be designed to function satisfactorily within the humidity limits of Chapter 101, para 2. Compliance shall be established by testing to the requirements of BS 3G100, Part 2, Section 3, Sub section 3.7, Tropical Exposure tests.

6.5 DIRECT EXPOSURE TO SUNLIGHT

6.5.1 Equipment likely to be exposed to direct sunlight shall not be degraded by such exposure. In particular, materials and panel information markings shall not deteriorate or fade. Compliance shall be established by testing to DEF STAN 07-55, Part 2 Section 2 Test B3. The rotorcraft manufacturer shall state the severity and duration of such tests.

6.6 EXPOSURE TO RAIN OR SALT SPRAY

6.6.1 Equipment mounted to the outside of the rotorcraft or in exposed parts such as engines or undercarriage shall function satisfactorily when subjected to rain or salt spray as appropriate. Satisfactory functioning, shall be established by testing to BS 3G100, Part 2, Section 3, Sub section 3.11, Waterproofness Tests and Sub section 3.9, Salt Mist Test Severity 1.

6.7 DUST AND SAND

6.7.1 Chapter 101, para 7 states the dust and sand proofing requirements for the rotorcraft. Whilst dust and sand proofing of the rotorcraft fuselage is a desirable aim there may be occasions when the proofing is insufficient or not possible.

6.7.2 DEF STAN 07-55 (Part 2) Section 4 specifies test methods for the ingress of dust and sand. The Rotorcraft Project Director shall state the requirement for dust and sand tests and the severity. The equipment shall function satisfactorily during and after these tests.

6.8 FUNGAL CONTAMINATION

6.8.1 DEF STAN 00-29 provides particulars of fungal germination and growth on material. Methods of preventing or limiting fungal growth on materials is also given.

6.8.2 Use of materials that may support fungal growth shall be avoided. Equipment shall not be damaged or deranged by and shall continue to function in the presence of fungal growth.

6.8.3 DEF STAN 07-55, Part 2, Section 3, Test C1 specifies tests to determine the extent of fungal growth under short term exposure and assess the effect of fungal growth on the functioning of the equipment by longer exposure.

6.8.4 The Rotorcraft Project Director shall state the tests required and severity of the tests.

6.9 CONTAMINATION FROM FLUIDS

6.9.1 Equipment and installations shall continue to function when subjected to contamination from fluids used in their particular location in the rotorcraft.

6.9.2 BS 3G100, Part 2, Section 2, Sub section 3.12 lists groups of fluids and details tests to be carried out on equipment. Contamination tests should only be carried out on items which are likely to be contaminated by fluid.

6.9.3 Requirement for tests, their severity and the appropriate fluid shall be stated by the Rotorcraft Project Director.

6.10 CHEMICAL ATTACK

6.10.1 Information concerning chemical attack is provided by DEF STAN 00-50. (See also new Chapter 717 - JAC Paper No. 1178 in preparation).

6.10.2 Should there be any likelihood of chemical attack within the rotorcraft then the Rotorcraft Project Director shall state the chemicals and the severity of tests to be carried out.

6.10.3 Procedures for tests are specified in DEF STAN 07-55 (Part 2) Section 3.

6.10.4 Contaminated equipment shall continue to function satisfactorily during the tests of para 6.10.3.

6.11 HANDLING TESTS

6.11.1 Equipment shall be sufficiently robust to withstand handling, and transit shocks.

6.11.2 Test procedures for bump (rough ride in the back of a four wheeled vehicle without packaging) bench handling and topple tests are stated in DEF STAN 07-55, Part 2, Section 1, Test A5 Bump, Test A4 Drop and Topple, Test A3 Shock.

6.11.3 Tests to be performed and the severity of the tests shall be specified by The Rotorcraft Project Director.

6.12 ELECTROMAGNETIC COMPATIBILITY

6.12.1 Requirements applicable to the limitation of propagated electromagnetic energy whether radiated or conducted and to the limitation of susceptibility of equipments, installations and systems to such energy are stated in DEF STAN 59-41. Further information and requirements particular to rotorcraft are stated in AvP 118.

6.12.2 Methods and conditions for testing are stated in DEF STAN 59-41 Part 3 (and Part 4 when published) and in BS 3G100 Part 4 Section 2. (See also Chapter 1011).

6.12.3 Specific requirements and severity of tests shall be stated by the Rotorcraft Project Director. These will include testing of equipments, subsystems and systems of the rotorcraft.

6.13 EXPLOSIVE ATMOSPHERE

6.13.1 Equipment which may come into contact with flammable vapours shall be designed to meet the requirements of Chapter 712, para 6.

6.14 NUCLEAR HARDENING

6.14.1 Consideration shall be given to requirements for nuclear hardening (see new Chapter 717 - JAC Paper 1178 in preparation). Specifically dose rates and EMP fields shall be included as choice of microprocessor is affected.

6.15 TEMPEST

6.15.1 Requirements for Tempest clearance shall be considered in conjunction with the Rotorcraft Project Director.

6.16 REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

6.16.1 See Chapter 112.

6.17 SAFETY CRITICAL SOFTWARE

6.17.1 See Interim DEF STAN 00-31 for requirements, design, verification and validation of safety critical software. (See also Interim DEF STAN 00-55 when available).

LEAFLET 725/0

AVIONIC EQUIPMENT INSTALLATIONS

REFERENCE PAGE

Defence Standards

DEF STAN 00-10	General design and manufacturing Requirements for Service Electronic Equipment
DEF STAN 00-13	Guide to the achievement of Testability in Electronic and Allied Equipment
DEF STAN 00-14	Guide for the Defence Industry in the use of ATLAS
DEF STAN 00-18	Avionic Data Transmission Interface Systems
DEF STAN 00-29	Fungal Contamination Effecting the Design of Military Materiel
DEF STAN 00-31	(Interim) The Development of Safety Critical Software for Airborne Systems
DEF STAN 00-41 (PART 1)	MOD Practices and Procedures for Reliability and Maintainability
DEF STAN 00-50	Guide to Chemical Environmental Contaminants and Corrosion Effecting the Design of Military Materiel
DEF STAN 00-52	General requirements for Test Specifications and Test Schedules
DEF STAN 07-55	Environmental Testing of Service Material (Progressively superseded by DEF STAN 00-35)
DEF STAN 59-35	Connectors Electrical and Connectors Fibre Optic
DEF STAN 59-36	Electronic Components for Defence Purposes
DEF STAN 59-41	Electro Magnetic Compatibility
DEF STAN 66-27	Pressure Connections on Aircraft Instruments and Associated Equipment

ARINC Documents

SPEC IEEE/ARINC	Abbreviated Test Language for all systems (ATLAS)
ARINC 404A	Air Transport Equipment Cases and Racking
ARINC 600 ..	Avionic equipment interfaces
DRAFT ARINC 604	Guidance for Design and use of Build in Test Equipment (BITE)
DRAFT ARINC 607	Design Guidance for Avionic Equipment

US Mil Specs/Stds

MIL-T-23103	Thermal Performance Evaluation, Airborne Electronic Equipment and Systems, General Requirements for
MIL-STD-1760	Aircraft/Stores Electrical Interconnection System
DOD-STD-1788	Avionics Interface Design Standard
MIL-STD-2165	Testability Program for Electronic Systems and Equipments

British Standards

3G100	General Requirements for Electrical Equipment and indicating Instruments for Aircraft

LEAFLET 725/1

AVIONIC EQUIPMENT INSTALLATIONS

THERMAL MANAGEMENT

1 INTRODUCTION

1.1 This Leaflet amplifies the requirements of Chapter 725, para 6.3.1.

2 EVALUATION

2.1 The Equipment Manufacturer should evaluate the Equipment Cooling requirements to ensure that the equipment will operate within the requirements defined by the rotorcraft manufacturer.

2.2 Initial requirements should be obtained by analysis and calculation. This calculation should be confirmed by evaluation tests on the prototype or early production equipment.

2.3 The cooling requirements should be determined when the Equipment is operating in the mode which provides the maximum heat output.

2.4 The requirements to maintain the Equipment at or below the maximum permissible operating temperature should be confirmed for all variations of ambient temperature and pressure which are applicable.

3 REFERENCES

3.1 Procedures for acquiring thermal performance data, and evaluating thermal performance of Airborne Electronic Equipment and Systems, are detailed in MIL-T-23103.

LEAFLET 725/2

AVIONIC EQUIPMENT INSTALLATIONS

DATA HIGHWAYS AND AIRCRAFT TO AIRCRAFT COMPATIBILITY

1 INTRODUCTION

1.1 Guidance and advice on the transmission of data and inter aircraft compatibility of data transfer is provided in DEF STAN 00-18. Guide to Avionic Data Transmission Interface Systems.

(i) Serial, Time Division, Command/Response Multiplex Data Bus.

(ii) Single Source, Single/Multiplex Link, Serial Digital Transmission Interface System.

(iii) Discrete Signal Interface.

(iv) Fibre Optic, Single Source, Digital Data Transmission.

2 STANDARDISATION

2.1 Standardisation has been attained by the adoption of US Specification MIL-STD-1553. This standard is embodied in DEF STAN 00-18 (Part 2) Serial, Time Division, Command/Response Multiplex Data Bus. It defines the requirements for digital, command/response time division multiplexing techniques for a 1MHz serial data bus and specifies the data bus and its interface electronics.

2.2 All aircraft/aircraft store interfaces which require electrical and information transfer will be designated in accordance with MIL-STD-1760.

LEAFLET 725/3

AVIONIC EQUIPMENT INSTALLATIONS

GENERAL AND INTERFACE REQUIREMENTS

1 CASE DESIGN

1.1 Typical equipment case design specifications are ARINC 404A and ARINC 600. These specifications define, standard modular case designs, standard racks or mounting trays, hold downs, temperature control systems for the equipment, and standard electrical connectors for connection of the equipment to the rotorcraft wiring.

1.2 ARINC 404A is primarily the standard for Civil Rotorcraft equipment, it is however suitable for many military applications, and is referenced, and its requirements quoted in US specification DOD-STD-1788, "AVIONICS INTERFACE DESIGN STANDARD".

2 CONNECTORS

2.1 Guidance in the selection and use of connectors is stated in:

DEF STAN 00-10, Part 5
DEF STAN 59-35 (Part 0)
DEF STAN 59-36 (Part 2)
CECC MUAHAG Preferred Products List Vol 3 (Connectors)

2.2 Rectangular connectors of the direct rack and panel type should be used on rack or tray-mounted equipment units. Engagement of mating connectors is by movement of the unit relative to the rack or tray. The connectors should incorporate an alignment device, such as guide pins, to ensure correct mating. Equipments using this form of interconnection require means of locking or clamping, and may, also require means of separation/removal.

3 GENERAL REQUIREMENTS

3.1 Units should comprise discrete physical modules, readily changeable at second line and capable of being tested to an approved specification.

3.2 Line Replacement Item (LRI) pressurisation should be avoided where possible.

3.3 Memory circuits should be non-volatile and programmable in service.

3.4 The maximum acceptable reduction as a percentage of the Mean Time Between Failure (MTBF) due to the inclusion of BIT should be no more than 5 per cent.

3.5 Minimising The False Alarm Rate. The false alarm rate due to the inclusion of BIT should be typically no greater than one per cent of system failure rate.

CHAPTER 727

HEALTH AND USAGE MONITORING SYSTEMS

1 INTRODUCTION

1.1 The purpose of health and usage monitoring (HUM) is to improve flight safety, rotorcraft availability, maintainability, the ability to complete a mission, and to reduce life cycle costs. System functions include the diagnosis of development faults, measurement of damaging loads, and performance monitoring. The system provides in-flight indications restricted to events likely to result in mission failure. After-flight indications are provided of the rotorcraft's fitness to fly and of servicing actions necessitated by component degradation. Information of use in maintenance planning is output via a data transfer device for analysis in a ground station. The HUM system provides additional support to mission effectiveness during wartime operations by reduced reliance upon subjective in-flight assessment of battle damage, and by improved diagnosis of damage repair requirements.

1.2 In order to fully utilise the benefits of Health and Usage Monitoring the outputs from the system should be fully integrated with the maintenance philosophy of the rotorcraft.

1.3 Functionally, in this chapter HUM is considered separately from other rotorcraft systems. However, many of the parameters which are required for such a system are necessarily provided for other functions, and on-aircraft HUM system indications may make use of existing aircrew display facilities. As a result of this interaction, the HUM system will generally constitute an integrated component of the total rotorcraft avionics system, although stand-alone HUM systems (or sub-systems) are not necessarily excluded.

1.4 This chapter sets out the requirements for the functional design and testing of the system. Equipments provisioned shall also comply with the requirements of the Basic Design, Manufacture, and Testing of Avionics system (DEF STAN 00-10), and with Avionic Equipment Installations (Chapter 725). Software shall comply with the requirements for 'essential' software in interim DEF STAN 00-31 (The Development of Safety Critical Software for Airborne Systems).

1.5 The HUM system encompasses health monitoring, usage monitoring, status monitoring and other functions.

1.6 The definition of terms used is given in Leaflet 727/1 together with a functional block diagram showing interrelationships.

1.7 The status of BCAR requirements for health and usage monitoring is indicated in Leaflet 727/3.

2 SCOPE

2.1 The scope of HUM system provisions shall be agreed with the Rotorcraft Project Director. This will include such issues as the extent of integration of the HUM system with other rotorcraft systems and the interfaces with these systems (e.g., avionics equipment and engines).

2.2 Whilst HUM relating to installation aspects of engines and APUs is included within the scope of this chapter, HUM relating to engine components falls within the scope of the DEF STAN 00-971 (General Specification for Aircraft Gas Turbine Engines). It is essential that integration of the engine monitoring functions is considered during the design and development of the rotorcraft HUM system, either in terms of providing an interface with HUM facilities furnished with the engine, or by incorporation of the engine monitoring functions within the rotorcraft HUM facilities.

3 DESIGN AIMS

3.1 The design aims of a Health and Usage Monitoring (HUM) System are to:

(i) Minimise the possibility of catastrophic accidents though:

(a) improved in-flight indications of serious degradation or failure or wear of flight-critical systems

(b) improved post-flight indications of trends of serious degradation of flight-critical systems.

(c) replacement of assumptions concerning fatigue usage with actual load measurements in service.

(ii) Assist the completion of the mission in the event of degradation of flight-critical systems, in peacetime and in wartime, through improved diagnostic information to the pilot.

(iii) Minimise front line maintenance time through accurate identification of faulty components, and speed battle damage repair.

(iv) Enhance rotorcraft availability through improved warning of incipient failures and pre-scheduling of component retirements on the basis of actual usage.

(v) Reduce life cycle costs through reduced accidents, and minimised scheduled maintenance and supporting procedures (e.g., TBO extension), and potentially by retiring components on the basis of actual usage and condition.

(vi) Minimise exposure to damaging flight condition by providing appropriate cockpit displays.

4 IMPLEMENTATION

4.1 Health and usage monitoring requirements shall be considered at all stages in the design and development of the rotorcraft and its installations, including the feasibility studies, during which a weight and installation allowance shall be made for the system.

4.2 A development plan shall be defined which permits caution and rejection criteria to be established.

4.3 Installations for flight safety is dependent upon health or usage monitoring shall be designed to ensure the effectiveness and practicality of monitoring, particularly in respect of sensor installations. The integrity of the monitored installation shall not be compromised by the monitoring provisions.

4.1 Consideration shall be given to the need for duplications of health and usage monitoring equipments and data transmission links to ensure a high level of system integrity, in safety critical areas.

4.5 The system shall function throughout the period between rotorcraft electrical power on to power off including for those periods of maintenance which influence monitored component lives. The system need not function during power on for other maintenance actions. The system shall be treated as an essential system, and sufficient non-volatile memory to satisfy all data recording requirements shall be provided and be compatible with the pattern of operation of the rotorcraft.

4.6 Should data acquisition or processing be interrupted or aborted at any time the incidence and duration shall be flagged in post-flight data displays so that usage data can be modified by contingency allowances.

4.7 Monitoring provisions, both hardware and software, required for wartime operations shall be installed and validated concurrently with peacetime provisions and shall be capable of rapid implementation.

4.8 In specifying requirements for sensors and associated signal conditioning attention shall be paid both to the functional requirements (e.g., frequency range required for gearbox accelerometers) and the environmental conditions, using measured data from previous rotorcraft components where appropriate. Particular attention shall be paid to the vibration, EMC, temperature, and the maintenance environment.

4.9 Wherever practicable sensors provided for other systems such as engine control systems and accident data recorders shall be used, but in a non-intrusive manner by the Health and Usage Monitoring system.

4.10 Wherever possible monitoring systems provided with government furnished equipment (GFE) such as engines and avionics equipment shall be integrated with the Health and Usage Monitoring system in accordance with para 2.

4.11 Specifications for sub-contracted systems shall include HUM requirements in accordance with this Chapter.

4.12 Where Accident Data Recorders are to be installed in the rotorcraft, consideration shall be given to the most effective means of combining functions that can have commonality with the Health and Usage Monitoring System.

4.13 Data to be transferred between the rotorcraft and the ground station shall be specified at an early stage by the Design Authority and agreed by the Rotorcraft Project Director so that the Services may determine maintenance policies and ground analysis requirements.

4.13.1 Data transfer times shall be less than normal refuelling times, and shall not interfere with the operational requirement of the rotorcraft.

4.13.2 Data ports shall comply with the requirements of DEF STAN 00-13 (Avionic Data Transmission Interface Systems).

4.13.3 Data transferred is to be fully annotated with time of day, date, and full identification of the parameter being monitored.

4.14 DATA SAMPLING AND PROCESSING

4.14.1 Sampling and processing intervals for health monitoring data shall be compatible with predicted or demonstrated damage propagation rates.

4.14.2 Sampling and processing intervals for usage monitoring data shall be at a rate sufficient to permit the detection of all peak values of monitored parameters.

4.14.3 Data sampling requirements for all monitoring functions shall not require special flights or damaging flight conditions.

4.15 SYSTEM RELIABILITY

4.15.1 The Design Authority shall indicate those areas in which the HUM system is contributing to the safety of the rotorcraft, and shall give an assessment of the likelihood of the mature HUM system detecting failure and giving spurious indications in those areas.

5 BASIC SYSTEMS ELEMENTS

5.1 DATA DISPLAY

5.1.1 The system shall provide to the pilot warnings and cautions, and advisories, relating to the functional status of flight critical systems (see Chapter 105). Advisories shall be displayed during ground operations but may be suppressed for flight where appropriate. The extent of coverage of flight critical systems shall be agreed with the Rotorcraft Project Director.

5.2 DATA PROCESSING AND STORAGE

5.2.1 The system shall provide, at the completion of each flight, structured indications of rotorcraft serviceability and system unserviceabilities as appropriate. The system shall detail adjustments, replenishments and installation replacements that are essential before the next flight. The system shall also detail maintenance actions which are essential before a period of time or number of operations to be agreed with the Rotorcraft Project Director.

5.2.2 The HUM system will need both working and archive storage. Where the system is integrated with other avionics systems parts of this storage may be shared. The following aspects shall be considered:

(i) The size of the working memory and the extent of provision of spare capacity for subsequent enhancements.

(ii) The size of the memory for archive storage.

(iii) The length of time for which the archive storage will retain valid data in the absence of rotorcraft power.

(iv) A strategy to be adopted in the even that the archive store becomes full, to ensure retention of the most important information. (This could occur if maintenance information could not be downloaded within the time scale assumed at the stage).

5.3 DATA TRANSFER

5.3.1 The system shall have provisions for transferring data between the rotorcraft and the operator's ground station with minimum risk of data loss or corruption. Telemetering of data is not to be considered unless required by Rotorcraft Project Director.

5.3.2 Data transfer may be accompanied by transfer of a physical medium or through a communication channel. For the former, robustness and reliability shall be considered. For the latter, data transfer times shall be optimized whilst retaining data integrity. In both cases, a procedure for checking the validity of the transferred data and a strategy to be adopted in the event of data corruption shall be included.

5.3.3 Where practicable, provision shall be made to download maintenance data independently from any security classified operational data.

5.4 Background information relating to system elements is given in Leaflet 727/2.

6 MONITORING FUNCTIONS

6.1 DEFINITION

6.1.1 The definitions of monitoring, and the functions to be considered are given in Leaflet 727/1.

6.2 EXTENT OF COVERAGE

6.2.1 The systems to be monitored, and the functions to be provided shall be agreed with the Rotorcraft Project Director.

6.2.2 Status Monitoring. Systems and installations for which indications of failure status and identification shall be considered shall include fuel systems, propulsion systems, transmission systems, main and auxiliary rotor systems, flight control systems, missions systems, and critical parts of the structure such as installation attachments.

6.2.3 Health Monitoring. In selecting health monitoring functions, consideration shall be given to:

(i) Deterioration of Grade A parts, however their service lives are substantiated.

(ii) All components involved in the phase separation of rotors where lack of such separation would result in catastrophic failure.

6.2.4 Usage Monitoring. In selecting usage monitoring functions, consideration shall be given to:

(i) The incidence and effect of operational hazards, abnormal loads, manoeuvres, or duty cycles.

(ii) Retirement of components on the basis of measured load history or other parameters determining cumulative damage.

(iii) Components potentially life-limited through the application of load scatter factors in design or substantiation, which have not had such factors applied.

6.3 TECHNIQUES - GENERAL REQUIREMENTS

6.3.1 The Design Authority shall inform the Rotorcraft Project Director of the details of the monitoring techniques and algorithms to be used.

6.3.2 Techniques selected shall be practicable and effective, and substantiated by relevant experience, sound design and test philosophies, and engineering judgement. Background information on preferred techniques is given in Leaflet 727/3.

6.3.3 Techniques selected shall be amendable to the application of simple caution and rejection criteria. The implementation of algorithms shall be such that adjustment of caution and rejection criteria can be made in service, by simple menu and keyboard inputs. Adjustments requiring new software, firmware or hardware changes are to be avoided. Means shall be provided by the Design Authority to ensure that unauthorised modification of these criteria does not occur.

7 TESTING

7.1 In all rig and flight tests of components for which monitoring functions are specified, action shall be taken to employ the definitive monitoring techniques intended for the rotorcraft HUM system, where practicable, and to use the results to further the development of the HUM system. Background information on the testing of HUM algorithms is given in Leaflet 727/4.

7.2 Test requirements covering the components of the Health and Usage Monitoring system are detailed in Leaflet 725/4 (Avionics Systems).

8 COMPLIANCE

8.1 It shall be demonstrated to the satisfaction of the Rotorcraft Project Director that the requirements of this chapter have been met.

8.2 Information on demonstration of compliance is provided in Leaflet 727/5.

LEAFLET 727/1

HEALTH AND USAGE MONITORING SYSTEMS

DEFINITION OF TERMS

1 INTRODUCTION

1.1 A number of monitoring functions are needed to provide an overall monitoring system capability as specified in Chapter 727. These functions are identified and defined in the following paras. It is considered impracticable to include all monitoring functions in the title of the system - a Health and Usage Monitoring System therefore may contain functions additional to health and usage.

2 DEFINITION OF TERMS

2.1 HEALTH MONITORING

2.1.1 Health Monitoring is a process which provides a means of determining the continued serviceability of components, systems, or structures, without the need for component removal for inspection. The process involves repetitive tests or inspections, and when analysis is performed by an on-board system the time delay between data sampling and output of results an be minimised.

2.1.2 Examples of health monitoring outputs include:

(i) Systems serviceable.

(ii) General alert - impending unserviceability.

(iii) Diagnosis of specific component, mode of degradation, and severity of degradation.

(iv) Prognosis of continued serviceability.

(v) Confirmation of degradation and unserviceability.

2.1.3 The term 'health monitoring' is used in preference to 'condition monitoring' which in civil aircraft usage has a quite different meaning unrelated to preventive maintenance (see Ref 1).

2.2 USAGE MONITORING

2.2.1 Usage Monitoring is a process which assesses the life consumption of life-limited components, systems, and structures by monitoring actual damage exposure (e.g., due to combinations of loads, speeds, temperatures, etc.). Allied to this is the recording of information relating to the exceedances of placarded limits, whether in emergency conditions or normal operations. The activity relates primarily to components substantiated on a 'safe-life' basis (see Chapter 200).

2.3 STATUS MONITORING

2.3.1 Status Monitoring covers:-

(i) Functional Status - cockpit indication of 'red warning' or 'amber cautions' (see Chapter 105) initiated by limit exceedances relating to health, usage or other status monitoring outputs.

(ii) Redundancy Status - monitoring the degree of redundancy remaining in systems containing additional components to provide continued serviceability following the failure of any of those components.

(iii) Switch/Valve Position Status - monitoring the position (selected and actual) of any multiple-way switch or valve which affects the operation of the rotorcraft (e.g., fuel valves, anti-icing system switches).

2.4 PERFORMANCE MONITORING

2.4.1 Performance monitoring is the primary method of relating system deterioration to consequence, and is primarily applied to engines but can be applied to all fluid dynamic systems, gearbox lubrication systems being one example.

2.5 VIBRATION CONTROL MONITORING

2.5.1 Vibration control monitoring is the means of signifying unacceptable vibration levels in relation to:-

(i) pilot comfort and performance.

(ii) damage to the structure, components, and equipment.

(iii) requirements in the Specification for the Rotorcraft.

2.6 ON-BOARD MONITORING.

2.6.1 On-Board Monitoring signifies any monitoring function for which data sampling and analysis is performed by a system on the rotorcraft, whether the system be installed or portable (i.e., no reliance upon ground equipment). A functional block diagram is shown in Fig 1.

2.7 GROUND BASED MONITORING

2.7.1 Ground-Based Monitoring signifies any monitoring function for which data sampling is obtained from the operational or, stationary rotorcraft, but analysis is performed by ground-based equipment at the rotorcraft or in a laboratory.

2.7.2 It is recognised that some functions may be analysed on board, but be subject to further analysis off the rotorcraft.

2.8 ON-LINE DATA PROCESSING

2.8.1 On-line Data Processing refers to the processing of data in 'real time', i.e., no significant delay between data equipment and processing.

2.9 BUILT-IN-TEST (BIT)

2.9.1 BIT is a diagnostic facility incorporated in self-contained systems such as engines or avionics equipment that is available for interfacing to the HUM system or for ground inspection.

2.10 ON-CONDITION-MAINTENANCE

2.10.1 On-Condition Maintenance is a primary maintenance process having repetitive inspections or tests to determine the condition of components, systems, or portions of structure with regard to continued serviceability. Corrective action is taken when required by item condition, (see Ref 1).

REFERENCES

No.		Title, etc.
1	CAA	Condition Monitored Maintenance an Explanatory Handbook CAP 418. CAA London July 1978.

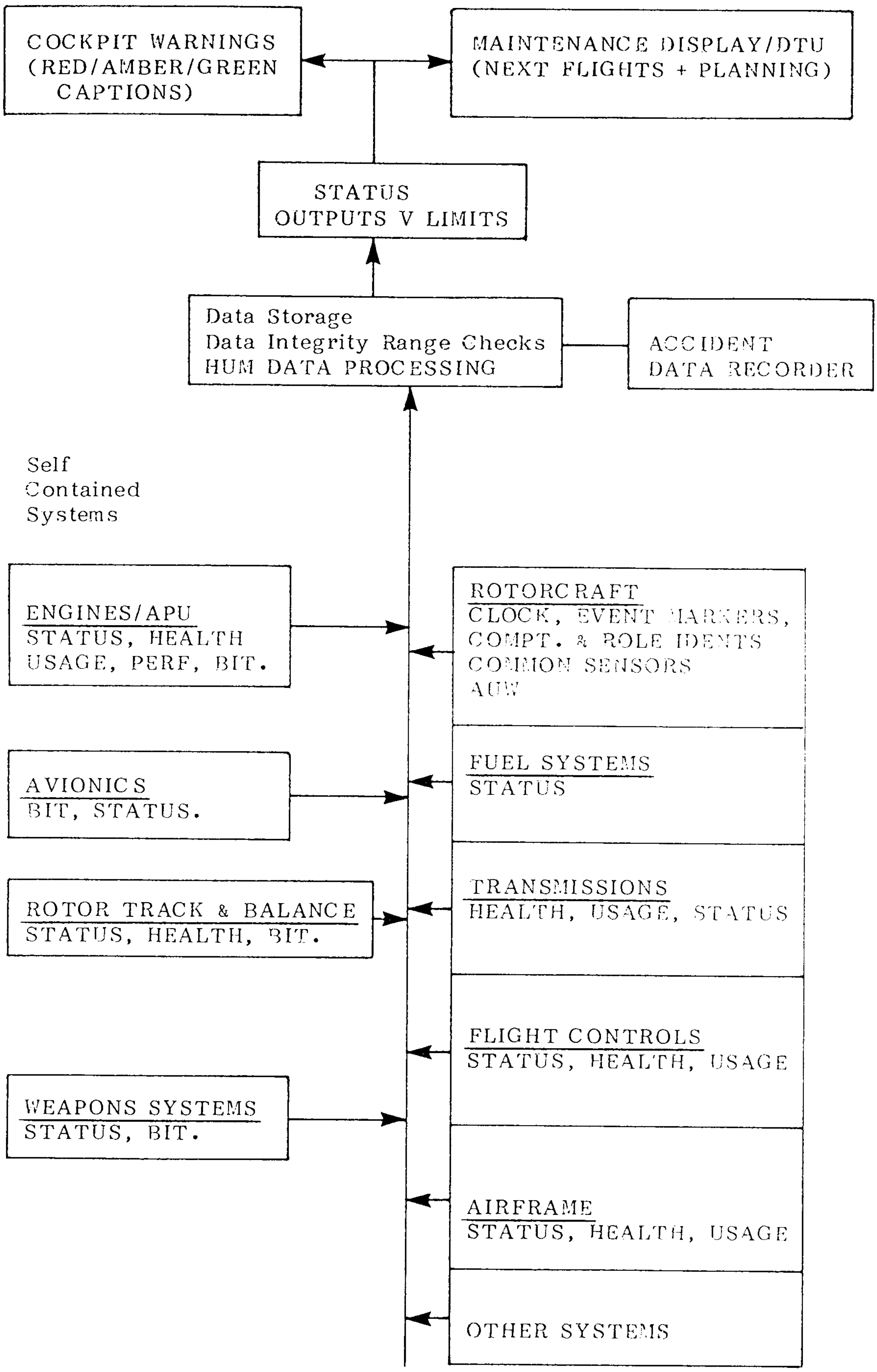

FIG 1 FUNCTIONAL BLOCK DIAGRAM

LEAFLET 727/2

HEALTH AND USAGE MONITORING SYSTEMS

SYSTEM COMPONENTS

1 INTRODUCTION

1.1 This leaflet describes equipments that may satisfy the system requirements detailed in Chapter 727, paras 4 and 5. An example is the system architecture proposed for the HUM system on the EH101 Helicopter as discussed in Ref 1.

2 SENSORS

2.1 In addition to those sensors fitted to the rotorcraft for engine and flight control purposes there are a wide range of flight-worthy sensors available that are appropriate to the monitoring functions required. Care should be exercised in selecting sensor types to ensure compatibility with the operational and maintenance environment to which they are to be subjected.

2.2 Where possible redundancy should be provided, either within the sensor selected, or through back-up sensors that can satisfy this need, if necessary with reduced sensitivity.

2.3 Where appropriate, sensors should have mounting provisions which facilitate removal for inspection or replacement without jeopardising the integrity of the monitored component, or requiring time-consuming maintenance action. An example is the need for self-sealing housings for gearbox wear sensors.

2.4 In selecting sensor types calibration burdens should be minimised.

3 SIGNAL INTERFACING

3.1 Data from particular sensors may be required for other systems in addition to the HUM system (e.g., electronic flight instruments and engine control systems). Sensor interface units separate from data processing units could therefore be beneficial in avoiding duplication of analog to digital conversion etc (see Ref 1).

4 DATA PROCESSING AND STORAGE

4.1 Dedicated processing of health and usage monitoring data may have significant benefits in avoiding derogation of the monitoring function, and in minimising re-certification effort relating to software changes (see interim DEF STAN 00-31 - The Development of Safety Critical Software for Airborne Systems).

4.2 Some health monitoring inputs such as vibration require analogue to digital conversion of high frequency signals and result in high rate digital data. Such data rates may necessitate dedicated data links between signal interface units and signal processing.

4.3 The acquisition of vibration data at high sample rates may create a large volume of data. Selection of the processing hardware should pay due attention to the required accuracy (method of representation of values in integer or real form) and to the speed of processing. Use of multiple processor schemes or schemes employing dedicated signal processing hardware should be considered.

4.4 A crash-protected recorder is not required for HUM purposes because these relate to assisting continued airworthiness of the rotorcraft. However HUM output data could provide useful information to accident investigators, and consideration should therefore be given to data transfer to the Accident Data Recorder or Cockpit Voice Recorder where fitted.

4.5 An example of how HUM systems can be integrated with Flight Data Recording systems is given in Ref 2.

4.6 Inclusion of a universal time of day and date clock within the HUM system is necessary for off-rotorcraft correlation of independently recorded data.

4.7 Flexibility in memory usage is useful. For example the memory may be segmented to store recent data which is being continually updated, and historical data which is updated less frequently.

4.8 In determining the data storage capacity and partitioning required consideration should be given to the operational and training roles of the particular rotorcraft. Down-loading of HUM data from rotorcraft employed on combat roles or deployed on small ships may not be possible for periods measured in weeks.

4.9 Health monitoring of the HUM system is addressed in Leaflet 727/3, para 10. It is anticipated that the HUM system will be adequately furnished with Built-in Test indications suitable for status monitoring. Considerations should be given to precisely what loss of data and what loss of functionality should constitute HUM system Caution and Warning Status conditions provided by the HUM system.

5 DATA DISPLAY

5.1 A display with alphanumeric keys and function keys to control data acquisition and data display should be placed within easy reach of the pilot (e.g., Refs 1, 3 and 4). If a multi-function display is used, consideration should be given to the hierarchy of display priorities.

5.2 The cockpit display needs to be large enough to accommodate status flags for all cautions and warnings that might arise at any time, plus keyword and legend information to identify the subject system and its condition, not necessarily simultaneously displayed.

REFERENCES

No.		Title, etc.
1	Roe J D	The EH101 Health and Usage Monitoring System from Conception to Implementation. I.Mech.E Conference 'Aerotech 87', Session S/606, Birmingham, October 1987.
2	CAA	Proposal to Amend Schedule 5 of the Air Navigation Order 1985 - Flight Recorders for Helicopters, D37/80/1 8 April 1988.
3	Astridge D G and Roe J D	The Health and Usage Monitoring System of the Westland 30 Series 300 Helicopter. Paper 81, Tenth European Rotorcraft Forum, The Hague, The Netherlands, August 1984.
4	Pardi R	Compact Display Alarm System. Paper 45. Ninth European Rotor Forum, Stresa, Italy, September 1983.

LEAFLET 727/3

HEALTH AND USAGE MONITORING SYSTEMS

RECOMMENDED TECHNIQUES

1 INTRODUCTION

1.1 The purpose of the health and usage monitoring (HUM) system is to improve the safety, availability, and the life cycle costs of the rotorcraft. To achieve this purpose it is essential that the health Monitoring techniques selected are effective and practical. This requires high reliability and accessibility of sensors and associated cabling, high reliability of the algorithms used in terms of accurate numerical assessment of conditions or damage accumulated, rapid and efficient processing of algorithms and acceptable cost and weight. This leaflet provides guidance on the selection and implementation of monitoring functions for major components, and makes recommendations concerning appropriate techniques. Engines and APU's are excluded from consideration in this leaflet, but need to be included in the design of the HUM system. (See DEF STAN 00-971 - General Specification for Aircraft Gas Turbine Engines).

1.2 Whilst the Civil Aviation Authority recognises the importance of health and usage monitoring to their requirements for higher levels of safety, particularly in transmissions and rotor systems (see Ref 1), the approach adopted is not considered acceptable by the Ministry of Defence for the design of new military rotorcraft. The approach of the Civil Aviation Authority is to allow airworthiness credit, where appropriate, against the more stringent safety requirements which have been introduced. The Ministry of Defence, however, considers it necessary to introduce specific requirements in DEF STAN 00-970 to ensure that the benefits of advanced health and usage in monitoring technology are realised in military rotorcraft.

1.3 The relative contribution of components and systems is to airworthiness failures reported in Military rotorcraft is given in Ref 2 (Fig A) and Ref 3 (para 1.7), and for civil rotorcraft in Ref 4 (para 3) and Ref 5 (p 61-6).

2 GENERAL

2.1 The primary cause of vibration in rotorcraft, and of the problems that result, is the unsteady nature of aerodynamic loading of the rotor(s). In addition to causing damage to the airframe, rotor-induced vibration causes much of the unreliability experienced in installed equipment, stores, and problems for the aircrew which can affect their ability to operate the rotorcraft.

2.2 Where a safety analysis (e.g., failure modes, effects, and criticality analysis) is carried out in the assessment of a component design, the results should be studied to assess the health monitoring requirements. Allowance should be made for the damage tolerance properties of certain components and systems as this will affect this assessment, both in terms of monitoring technique and rejection criteria.

2.3 It is recognised that the nature of substantiating health monitoring algorithms and intervention criteria requires extension of the activity beyond the in-service date. It may not always be practicable to make modifications to hardware after the in-service date.

2.4 Procedures need to be established for feed-back of health and usage monitoring data from operators to the Design Authority:

(i) to establish that the rotorcraft is being used within limits assumed in design and that the substantiation of components remains valid.

(ii) to confirm the anticipated correlation between health monitoring indications and component condition. This will require data relating to healthy, degrading, and degraded state of components.

(iii) to permit controlled changes to algorithms or intervention criteria to be made to all rotorcraft of the type in service.

2.5 A facility should be provided to implement, with the agreement of the Rotorcraft Project Director, any software modifications found necessary with increasing operational experience of the rotorcraft.

2.6 Evidence should be provided of the effectiveness of the proposed techniques when applied to the proposed rotorcraft components, whether or not the techniques correspond closely to those described in this leaflet.

2.7 Review of health and usage monitoring developments in rotorcraft are given in Refs 6, 7 and 8.

3 COMPONENTS AND APPROPRIATE TECHNIQUES

3.1 The full definition of a major component or system given in the appropriate installation chapter should be noted in selecting monitoring provisions.

3.2 Components are reviewed under the following para headings:-

Rotor systems	-	para	4
Flight control systems	-	para	5
Transmission systems	-	para	6
Airframe and Undercarriage	-	para	7
Fuel systems	-	para	8
Electrical systems	-	para	9
Avionics systems	-	para	10
Anti-Icing system	-	para	11
Central warning system	-	para	12
Event recording	-	para	13

3.3 Where appropriate an indication will be given where ground-based health monitoring techniques and laboratory facilities are advocated, either to supplement on-board provisions or as the only effective technique for that potential failure mode. Organisational requirements, and some of the potential problems associated with off-rotorcraft health monitoring facilities are discussed in Ref 9.

4 ROTOR SYSTEMS

4.1 SCOPE

4.1.1 This para includes rotor blades, hubs, dampers, anti-vibration devices, control and all other components that form part of the rotating assemblies, but excludes the transmission and rotor drive shafts.

4.2 BACKGROUND

4.2.1 A review of health monitoring methods and the nature of rotor loading applied to rotor systems is given in Ref 7 P4-16, and 10.

4.3 ROTOR ASSEMBLIES - HEALTH AND VIBRATION CONTROL MONITORING

4.3.1 Rotor Track and Balance Monitoring (RTBM)

(i) Advanced methods of RTBM are capable of diagnosing faults within rotor blades, lag-plane dampers, and rotor assemblies. Some of the faults may be maladjustments or other conditions for which corrective adjustments are permitted. Other faults may require component replacement or battle damage repair. It is important to correct adjustment errors to minimise the effects of rotor vibration on aircrew, the rotorcraft, installed equipment, and stores. An RTBM system is therefore required to identify conditions in which vibration limits are exceeded, to diagnose the cause, and to advise appropriate remedial action and degree of urgency:-

(a) adjustments within permitted limits.
(b) component replacement.
(c) battle damage repair.

(ii) Permanently mounted sensors should be provided for monitoring the track and lag of the rotor blades, and triaxial vibration adjacent to the main and auxiliary rotors. Flight trials may indicate other (or additional) vibration sensor locations. The positioning of the sensors should be optimised and defined in the maintenance manual. Where blade tracking requires attachments to the blades, on a permanent or temporary basis, they should not affect adversely the aerodynamic performance (particularly in autorotation). Neither should the radar, infra-red or acoustic signature be affected, nor most importantly the visibility of the rotorcraft by day or by night.

(iii) The system should be capable of day and night, all weather operation, and capable of operation by a single member of the aircrew.

(iv) Vibration limits at the defined sensor positions should be included in the display in addition to the actual values.

(v) The installed sensor measurement accuracies should be better than:-

(a) Blade Track: ± 3 mm in flap, ± 3 mm in lag at 10 m radius. This accuracy is necessary to establish trends with forward speed, ground, and hover, and for diagnosis of faults.

(b) Balance: Imbalance amplitude 0.05 ips, phase angle ±0.5°. All vibration data should be digitised at a sampling rate equal to 512 multiples of rotor speed.

(vi) Equipment associated with RTBM has been known to cause EMC problems. Therefore RTBM equipment should conform to the EMC specification for the rotorcraft, both in respect of radiated and conducted energy.

(vii) Hard copy output recommending rotor adjustments should be considered for use by maintenance personnel.

(viii) Rotor adjustments should be limited to track rods, specified tabs, chordwise and spanwise balance weights, where permitted. All permissible adjustments, limits, procedures, and tooling required should be fully defined in the rotorcraft maintenance manual.

(ix) Analysis software should control the acquisition and checking of track and balance data with display menu prompts. From time-averaging data, trends in the behaviour of each blade and the rotor assembly should be automatically computed for the ground and flight conditions specified, and stored for down-loading via the data transfer device. All adjustments made should be input to the on-board system, stored for down-loading, and the information used for improving the algorithms.

(x) Rotor track and balance measurements should be capable of completion within one flight, although it is considered that the 'best' balance may require additional flights. This is particularly true on twin-rotor rotorcraft.

(xi) Faults developing in the rotor system or pitch control system can manifest themselves through changes in blade track or rotor balance, and would be expected to continue producing changes with fault progression. Where practicable therefore the frequency of adjustments associated with a normal, healthy rotor should be established, and this frequency monitored to determine whether fault check list procedures should be invoked.

(xii) Sufficient recording allowance should be made for track and balance data including time/date and description of subjective assessment of vibration and changes made, for several test flights appropriate to the rotor design and rotorcraft operations.

(xiii) On rotorcraft fitted with an external Maintenance Data Panel (MDP) it should be possible to operate the RTBM facilities from the cockpit and from the MDP.

(xiv) The integrity status of track sensor, accelerometers, and rotor azimuth reference signals should be displayed together with rotor rpm and indicated airspeed.

(xv) Particular consideration should be given to relaxation of limits for wartime operation, to extending permissible adjustments, and to jury-rigged corrections to compensate for battle damage.

(xvi) Rotor track and balance checking procedures should have an on-request capability.

(xvii) Further information on rotor faults and the capabilities of RTBM is given in Refs 11-15.

4.3.2 Rotor Related Vibration

(i) Rotor induced vibration can result in potentially serious faults developing in the rotor and rotating control systems. Some faults cause changes in amplitude of rotor order frequencies and are traditionally traced through manual check-list procedures relating to known component characteristics. Other faults produce signature changes related to the frequency characteristics of that component which differ from rotor orders but occur in a similar frequency range.

(ii) Therefore facilities should be provided to:

(a) indicate rotor order vibration vectors or trends which are out-of-limits.

(b) indicate other specific characteristic frequencies which are out-of-limits on level or trend.

(c) detect the growth in amplitude and number of non predetermined frequencies in this range.

Exceedance (a) would precipitate track and balance checks followed by check-list procedures prompted on the display, or output of other diagnostic information included in the software.

Exceedance (b) would immediately invoke component identification(s), any built-in diagnostic functions, and check-list procedures prompted by the display.

Exceedance (c) could invoke amber caution indications with cause unknown, and maintenance indications requiring a manual/visual check of the complete rotorcraft.

Gross exceedances in any category should invoke red caption warnings together with component identification, where practicable, in accordance with Chapter 105.

(iii) Vibration 'snap-shots' should be recorded for down-loading to ground-based facilities, but processed on-board to enable status to be assessed relative to the three categories described above, and cautions or warnings displayed if the appropriate limits are exceeded. Provisions should be made to store a number of processed or raw vibration snap-shots signatures, together with rotor reference signals, as agreed with the Rotorcraft Project Director.

(iv) Minimal data processing requirements recommended are a 1024 point Fast Fourier Transform, frequency ranges 0-20R and 0-10T for conventional rotor blade complements, and a resolution of 0.02 inches per second.

(v) Further information on the application of these techniques is given in Ref 15.

4.4 ROTOR ASSEMBLIES - USAGE MONITORING

4.4.1 Parameters to be monitored may vary between different rotor designs especially in respect of the number of 'finite-life' components substantiated on a safe-life basis. Examples given below are based on a rotor usage monitoring system designed for a particular rotorcraft (Ref 17).

(i) Rotor Hours: e.g., time for NR ≥80%. This will eliminate errors experienced with manual recording.

(ii) Rotor Starts: e.g., when NR >80% rising.

(iii) Torque (main and tail rotors): maximum values and the number and magnitude of torque reversals at rotor speeds above 80%. Torque resolution should be better than ± 2%. Phase displacement and FM Telemetry strain-gauge torquemeters have proved satisfactory in rotorcraft applications.

(iv) Strain. Strain gauges can be bonded onto life-limited rotating components using the same technology as strain gauge type torquemeters. Particular care is required in the selection of the gauge and the bonding process and the provision of adequate protection against climatic and maintenance environments. It may be necessary to compromise the ideal gauge location (from a stress-measurement viewpoint) to achieve satisfactory reliability and life. The provision of additional gauges at the time of manufacture may reduce the effects of fatigue life scatter, although all gauges (in use or otherwise) start to consume fatigue life immediately upon installation. Signals can be transferred from rotating components to stationary structure by means of frequency modulated capacitive/inductive systems with high integrity.

In addition to cumulative damage and exceedance monitoring, strain gauge data from the rotor head can be used to compute and display ROTOR HEAD MOMENT where this is required to be controlled as in the example of Ref 17 during ground taxi manoeuvres. A potential problem with the strain gauging of large components is the calibration necessary for the instrumented component.

4.5 ROTOR BLADES

4.5.1 Health Monitoring

(i) Metal blades tend to be life-limited by potential fatigue fracture of the spars - these should be protected by pressure-leak systems, or crack detection wires or fibres, interfacing the health and usage monitoring system. Pressure-leak systems are well established and have proved effective (e.g., 'BIM', 'IBIM', & 'ISIS'). Crack detection wires or fibres require more development work and in-service experience. Conducting paint has been used for detection of surface cracks. Most experience to date is with ground-based NDT techniques.

(ii) Composite blades are less likely to fracture, but are likely to take up water at different rates, especially when new or when operated in very high humidity. This usually manifests itself through rotor imbalance (covered in para 4.4). Delamination of fibres near the surface in outboard regions would be expected to change the aerodynamic characteristics of the blade and show up in track monitoring and in vibration (1R vertical) monitoring. Embedded optical fibres for monitoring strain or fracture could be effective throughout the blade, but much development work still needs to be done. It is essential that incorporation of such fibres does not reduce the strength of the blade.

4.5.2 Usage Monitoring: The considerations of para 4.4 apply to rotor blades.

4.5.3 Blade Icing: Ice accretion on rotor blades may be detected by a change in vibration amplitudes at blade passing frequency.

4.6 ROTOR HUBS AND BLADE ATTACHMENTS

4.6.1 Health Monitoring:- Whilst on-line monitoring of metallic components with strain or fracture fibres may be considered feasible, experience to date is limited to ground-based inspections (visual and NDT) which still need to the carried out on a scheduled basis and in the course of check-list inspections following RTBM or Vibration (1R and NR) limit exceedances (see para 4.3.1. (xi)). For fibre-composite hub structures with redundant load paths, the inclusion of optical fibres for strain or fracture detection could be considered.

4.6.2 Usage monitoring:- The considerations of para 4.4 apply to rotor blades and blade attachments.

4.7 ROTOR HEAD BEARINGS - BLADE RETENTION AND ARTICULATION

4.7.1 Health Monitoring:

(i) Rolling element bearings are unlikely to feature in new rotor system designs because of adverse weight, life and reliability characteristics, but if featured could be monitored as per drive shaft bearings (see para 6.7) together with high integrity signal transmission as per para 4.4.1 (iv).

(ii) Elastomeric bearings may be amenable to thermal monitoring or stiffness checks, but current experience is limited to visual examination for delamination (cracks and debris) - see Ref 18, P32-19, and Ref 19.

(iii) Bearingless arrangements could be treated as per para 4.6.1.

4.7.2 Usage Monitoring: The considerations of para 4.4 apply to rotor head bearings.

4.8 LAG DAMPERS

4.8.1 Health Monitoring: Damper faults can manifest themselves through lead-lag track errors and at a later stage in airframe vibration whilst on the ground with rotors turning, especially during run-up and run-down. The provisions of para 4.3.2 are therefore appropriate.

4.8.2 Usage Monitoring: The considerations of para 4.4 apply to the structural components of lag plane dampers.

5 FLIGHT CONTROL SYSTEMS

5.1 SCOPE

5.1.1 This para includes pilot's controls, non-rotating pitch control systems, electrical and hydraulic power supplies and servo-control systems, mechanical linkages, stability augmentation systems, active control systems.

5.2 BACKGROUND

5.2.1 A review of health monitoring methods appropriate to flight control systems is given in Ref 7 p A4-18.

5.3 CONTROL LINKAGES AND BEARINGS

5.3.1 Health Monitoring: Control linkages have failed due to unforeseen vibration conditions arising from changes in stiffness with time. Vibration monitoring at strategic points could therefore be considered, monitoring excessive changes of amplitude at characteristic frequencies. Hollow linkages can be monitored for fracture by the use of pressure-leak systems. Whilst this has been implemented on a visual-inspection basis, the time to failure from crack detection would justify on-line provisions. Wear in rod-end bearings may manifest itself by a change in system vibration characteristics but otherwise will require ground-based inspections. Rolling element thrust bearings have been successfully covered with temperature monitoring and vibration.

5.3.2 Usage Monitoring: The functions listed in para 4.5 could be appropriate to rod-end bearings provided that sufficient consistency in manufacture can be maintained.

5.4 HYDRAULIC POWER SUPPLIES AND CONTROL SYSTEMS

5.4.1 Status Monitoring: The pressure in multiple circuits should be monitored to identify any loss of redundancy. Multiple servo-valves could likewise be monitored for jamming (e.g., the system described in Ref 17).

5.4.2 Health Monitoring: Wear in pumps and valves may be detectable by monitoring pressure and temperature performance characteristics relative to base-line. Ground-based analysis of fluid samples for contamination is well established practice, one effective method being described in Ref 20.

5.5 ELECTRICAL AND ELECTRO-OPTICAL CONTROL SYSTEMS

5.5.1 Status Monitoring: Components and circuits would be expected to be furnished with Built-in-Test facilities, which should be interfaced by the HUM system to continuously assess the level of redundancy existing.

6 TRANSMISSION SYSTEMS

6.1 SCOPE

6.1.1 This para includes gearbox internal components, lubrication systems, drive shafts and support bearings, couplings, freewheels, and rotor brakes; but excludes casings, load struts, and mounting features, which are more conveniently covered in para 7.

6.2 BACKGROUND

6.2.1 A reviews of health and usage monitoring methods applicable to rotorcraft transmissions are given in Refs 2, 6 and 7 (A4-8 to A4-14).

6.3 INTERNAL COMPONENTS - HEALTH MONITORING

6.3.1 Complementary Methods

(i) To provide adequate monitoring of gearbox internals, a suite of facilities is required to cover the range of possible failure modes, which include fracture of gears, hubs, splines and shafts which may not release much debris; surface fatigue of gear teeth and bearings which tend to release steel particles greater than 1 milligram before becoming a risk to airworthiness; and fatigue scuffing, micropitting, fretting, corrosion, etc., releasing very small particles (individually sub-visible) that may be ferrous or non-ferrous. The facilities should therefore comprise effective: -

(a) vibration monitoring (primarily for fracture modes).

(b) ferrous wear debris monitoring (primarily for surface fatigue modes).

(c) non-ferrous wear debris monitoring (primarily for retention failures and non-ferrous bearing cages).

(d) oil analysis (primarily for corrosion and small scale wear processes).

(ii) On-line systems exist with successful service experience for (a) and (b) but not for (c) and (d). Monitor filters of the coarse strainer type have successful application experience whilst much finer versions of order 100 microns (see Ref 6) are lacking in field experience - both are normally configured for ground inspections, although indicating screens of the coarser variety are available. A range of effective laboratory analysis techniques exist for (d) (see Refs 9 and 21).

Application experience with a suite of complementary techniques is given in Refs 6, 22, 23.

(iii) When implementing complementary monitoring methods, care is required to ensure that:-

(a) rejection status warnings are not generated by early indications from a system which responds to slowly developing wear modes (e.g., Ref 23).

(b) indications of damage initiation and progression are not derogated by 'satisfactory' indications from laboratory methods appropriate to slowly propagating modes (e.g., Ref 31, para 2.3).

6.3.2 Vibration Analysis

(i) A vibration analysis technique with application experience to service rotorcraft gearboxes of both fixed shaft axis and epicyclic configurations, and with demonstrated success in detecting localised gear tooth bonding fatigue failure, employs Signal Averaging. Important factors in defining the signal average computation include frequency range and resolution and averaging period. The choice of appropriate algorithms to post-process the signal average, prior to the application of numerical assessment parameters, will critically affect the ability to detect incipient failures at the earliest stage. Successes with this technique relate primarily to gear tooth cracks and fractures, and warnings several flights/ hours in advance of fractures have been demonstrated. The techniques have also successfully detected gear web and shaft fractures and gear tooth pitting (see Ref 24).

(ii) The technique is capable of pin-pointing precisely the shaft on which the affected gear is mounted and hence could be used to indicate appropriate flight control and maintenance actions.

(iii) It is not possible to guarantee that this technique will detect all possible fracture modes within every type of gearbox with sufficient warning to complete a mission, but it is capable of significantly enhancing the airworthiness of all gearboxes, and should therefore be included as a vital element of the health and usage monitoring systems.

(iv) Fractures of quill shafts and cracks in gearbox casings (bearing housings) have also been detected by vibration analysis, by examining the variation with time of amplitudes of gear meshing harmonics and shaft order 'sidebands'.

(v) Experience with detection of localised and distributed tooth surface damage and bearing spalling by vibration analysis has been moderately successful. Depending on gearbox complexity and gear tooth geometry, vibration analysis may usefully complement wear debris analysis for such failure modes.

(vi) Particular care is required in the application of vibration analysis to:-

(a) sample data at constant speed and in a prescribed torque window.

(b) obtain datum samples of data representing the new or overhauled condition of the gearbox. (Manufacturing and assembly tolerances within design limits can cause significant variations in processed vibration data as can natural frequency response changes). Some vibration monitoring parameters need to be assessed with respect to datum conditions.

(c) react to decreasing trends in vibration analysis parameters. In very many parameters both normalised and absolute, the extension of surface damage and of cracks result in decreasing trends of the descriptor following a rise from initial values relating to the undamaged condition.

(vii) Where vibration analysis packages are encapsulated in self-contained hardware, care must be taken to ensure that the requirements of Chapter 727, para 6.3.3 are satisfied.

6.3.3 Ferrous Wear Debris Monitoring

(i) Service arisings on transmission systems tend to be dominated by wear of steel components, which is manifested by ferrous debris. The size, morphology, and generation rate characteristics tend to vary with type of component and material, wear mode, and damage progression. It is important to both cost of ownership and monitoring system credibility issues to be able to distinguish between manufacture/assembly debris and acceptable or benign wear modes on the one hand, and potentially serious arisings on the other. Even potentially serious conditions can take several hundreds of hours to develop to the stage where they become a safety hazard (see Ref 38). Seldom can wear modes be arrested by service action other than component replacement. Monitoring provisions should therefore be capable of indicating the point at which component replacement is necessary, with sufficient safety margins included.

(ii) On-line ferrous wear debris monitoring systems exist with service experience that are capable of monitoring particle size and generation rate, but not particle morphology. It is necessary therefore to be able to extract sample wear debris for visual examination at the rotorcraft, and further analysis with laboratory equipment in order to identify the nature of the process, and the component releasing the debris.

(iii) Where the gearbox construction permits, it may not be necessary to fit on-line sensors to all sumps or scavenge lines of a gearbox. Thus it may be possible in complex gearboxes to provide one on-line sensor for status monitoring purposes and a number of magnetic plugs to facilitate fault isolation and diagnosis by inspection.

(iv) In the design of the gearbox and lubrication system consideration should be given to maximising debris transport efficiency to the sensors. Wherever possible probe type sensors should be located in areas of full flow such as sump outlets, or in main scavenge lines with 'dog-leg' arrangements or vortex separators which direct the debris towards the sensor. Sensors which encompass the oil flow should be placed as close as possible to the scavenge pipe entry. For sump-mounted probes consideration should be given to incorporation of lightweight debris catchment trays having smooth surfaces, shaped to facilitate oil washing of debris onto the probes.

(v) Wear debris monitoring systems capable of full-flow monitoring are preferred.

(vi) Preferably the sensor should have debris retention capability in order to facilitate post-flight inspection of the debris produced in flight, when outside acceptable limits.

(vii) Sensors having debris retention capability should be immune to 'chip dance', loss of retention, and should be fitted in self-sealing housings to facilitate inspection.

(viii) Wear sensors should have continuous operation characteristics to ensure debris is not missed.

(ix) Wear sensors should be capable of producing an output proportional to the mass of the debris as it is entrained or passes through the sensor, and thence be capable of providing trend and total accumulation information through on-line analysis.

(x) The wear sensor system should be capable of discriminating between 'large' and 'small' particles. It is particularly important to be able to assess the mass trends and accumulation of particles of length greater than 300 microns or mass greater than 0.1 mgms. It should not be possible for a slow accumulation of small debris to be interpreted as a number of large particles.

(xi) Care should be taken in the interpretation of wear trend data. Whilst a significant or even total reduction in wear rate following a significant period of increasing wear may signify a reduction in the wear process, the component could be in a hazardous condition and therefore be prone to rapid failure.

(xii) Wear debris monitoring outputs should not be affected by aeration of the oil nor by temperature, vibration or EMC environment of the sensors.

(xiii) The wear debris monitoring system should have full built-in-test capability including the facility to simulate the presence of a significant wear particle at the sensors.

(xiv) Where debris data analysis is encapsulated in dedicated hardware, care must be taken to ensure that the requirements of Chapter 727 para 6.3.3 are satisfied.

(xv) Further considerations relative to on-line wear debris monitoring may be found in Ref 25.

(xvi) Two on-line wear debris monitoring systems that exhibit many of the above characteristics, and have mature experience in rotorcraft transmission applications are described in Refs 26-30.

6.3.4 Non-Ferrous Wear Debris Monitoring.

(i) The release of non-ferrous wear debris tends to be common occurrence in service, also the movement of non-ferrous, often non-metallic compounds, associated with gearbox assembly. Whilst non-ferrous materials are not usually involved in highly stressed rolling or sliding contacts other than as a surface coating they do feature in support structures and in rotating assemblies. Wear modes may take longer to develop to a serious condition than in steel components such as gears and bearings, but they nevertheless can affect the integrity of the transmission and need to be effectively monitored.

Many of the considerations discussed in relation to ferrous wear debris monitoring also apply to non-ferrous debris. On-line monitoring systems are less well developed for non-ferrous debris, however, and greater reliance upon laboratory based methods is necessary (see Ref 9).

(ii) Whilst experience has been gained with on-line analysis of non-ferrous wear debris in space applications, and the technology developed for rotorcraft applications (Ref 32), there is little or no rotorcraft experience with the technique.

(iii) Rotorcraft experience exists (on the CH 47D transmission) in the form of electrical indicating screens. These would be appropriate for any conducting particles.

(iv) Industrial experience exists with filter blockage measurement configured to produce trend information, but there is no experience with rotorcraft applications. There is rotorcraft experience with remote indication of main oil filter-blockage, which could therefore be considered for wear rate determination until devices with greater precision become available.

6.3.5 Oil Parameters:

(i) Trend analysis of the parameters identified in para 6.5.1 can be performed to reduce the possibility of degradation in the gearbox reaching caution or warning status.

(ii) Transducers appropriate to the above are discussed in para 6.5.2 (ii).

6.4 INTERNAL COMPONENTS - USAGE MONITORING

6.4.1 The processing of torque data is sufficient to satisfy the usage monitoring requirements of rotorcraft gearbox internal components to determine:

(i) cumulative damage sustained by life-limited components.

(ii) limit exceedance data.

6.4.2 An example of a mandatory requirement for continuous assessment of cumulative damage is given in Ref 17 (MGB tail take-off gears with no design allowance for load scatter). The means for satisfying that requirement is also described.

6.4.3 The resolution accuracy required in torque-measurement is ± 1% which is achievable in the torque monitoring systems described in Refs 17, 33 and 34.

6.4.4 Mean torque signals should be sampled at a frequency of 10 Hz or greater.

6.4.5 The torque range corresponding to the lowest gear endurance limit to the maximum conceivable conditions should be divided into a number of bands as determined from stress-life considerations. In addition to storing the total time accumulated within each band, damage counts should be computed for each life-limited gear in the transmission by applying Miners Law and the fatigue damage characteristics of the material (S-N curve) to the appropriate torque time data. Damage counts are normally expressed as a fraction of the predicted life available such that unit value signifies that predicted life has been consumed.

6.4.6 When determining which gears and shafts should be included in the analysis by virtue of damage threshold exceedance, the most adverse of emergency conditions should be applied.

6.4.7 Each incidence of torque 'red-line' exceedance should be recorded together with maximum amplitude, elapsed time, time of occurrence, and gearbox serial number. The addition of keyed in sortie codes could assist with improving the design knowledge relating to operational load spectra.

6.5 LUBRICATION AND COOLING SYSTEMS

6.5.1 Status Monitoring. System warning and caution indications should be triggered by abnormal oil pressure and temperature, oil filter blockage, or low oil level.

6.5.2 Health Monitoring

(i) Trend analysis of the above parameters can be performed to reduce the possibility of degradation in the oil system leading to caution or warning alerts.

(ii) Solid state transducers for measuring oil temperature and pressure offer the resolution accuracy required. OAT and torque data may need to be included in the analysis to account for effects of ambient temperature and load. Electro-optical sensors with no moving parts are in use for measuring static oil and fuel levels in rotorcraft with multi-sensor configurations available to improve resolution accuracy.

6.6 DRIVE SHAFTS AND COUPLINGS - HEALTH MONITORING

6.6.1 Vibration monitoring is advocated for detecting damage in drive shafts and couplings (see Ref 4 Table 6). This may be accomplished by monitoring spectrum changes at shaft speed and sub-harmonics with signals from accelerometers mounted on gearbox casings or airframe structure, or possibly from vibratory torque.

6.7 DRIVE SHAFT SUPPORT BEARINGS - HEALTH MONITORING

6.7.1 Temperature measurement by use of thermocouples affixed to the bearing housing is advocated, based on successful tests. There is also test evidence to support the use of the 'Shock Pulse' method employing tuned accelerometers, with carrier frequencies of the order of 30 kHz (see Ref 35).

6.8 FREEWHEEL UNITS

6.8.1 Health Monitoring

(i) Both cam-and-roller freewheels and sprag type clutches are prone to wear with consequent risk of sudden loss of drive.

(ii) The nature of freewheel operation makes on-line monitoring of wear very difficult. Freewheels should therefore continue to be made accessible for post-flight inspection.

6.8.2 Usage Monitoring

(i) Freewheel actuation could be monitored via the electrical actuation signals and occurrences of slam engagement recorded and flagged up for post-flight inspection or replacement.

(ii) Rotor starts and freewheel actuations could be recorded and correlated with inspection findings.

6.9 ROTOR BRAKES

6.9.1 Health Monitoring

(i) Consideration should be given to monitoring Rotor Brakes for wear with special attention to possible fire hazards with hydraulically operated brakes. Correlation of temperature and rotor brake actuation would contribute to reducing the fire hazard.

(ii) Pad wear could be monitored by means of buried electrical contacts similar to automobile practice.

(iii) Actuation mechanisms could be monitored for intermittent electrical faults comparing input and response (rotor deceleration).

7 AIRFRAME AND UNDERCARRIAGE

7.1 SCOPE

7.1.1 This para includes the whole of the rotorcraft structure and attachments including gearbox casings, mountings, engine supports, active airframe vibration control systems, the undercarriage, and stores supports.

7.2 BACKGROUND

7.2.1 A review of health monitoring of structures is given in Ref 7, p A4-20/21 and Ref 8, p10.

7.3 STRUCTURE - HEALTH MONITORING

7.3.1 Some of the difficulties in using strain gauges and the precautions to be adopted are discussed in para 4.4.1 (iv). Calibration problems are likely to be more severe on large components such as airframes. Automatic no load offset checks may improve long term accuracy of mean load measurements. Where the strain data required can be obtained from detachable parts (such as gearbox mounting struts or tail-fold lugs), the calibration problems are greatly reduced and this strategy should be employed wherever practicable.

7.3.2 There remains the possibility of bonded wires or optical fibres to aid visual inspections and NTD methods, but experience in rotorcraft applications is lacking. Ample provision should therefore be made in the design of structures to permit visual inspection.

7.3.3 The problems of corrosion is expected to persist whilst metallic structures are used for gearbox casings, support decking, linkages, etc. Visual inspection is expected to remain the primary method of corrosion control.

7.4 STRUCTURE - USAGE MONITORING

7.4.1 Direct load measurement is desirable because of its ability to cater automatically for varying weight, centre of gravity position and configuration (e.g., under-slung load carriage), as well as abnormal conditions such as icing. The implementation problems which may be encountered are described in para 4.4.1 (iv) and 7.3.1.

7.4.2 In view of the practical instrumentation problems in adopting a direct load measurement technique, a parametric approach may therefore be appropriate in conjunction with a dynamic model of the structure (e.g., Ref 8). The dynamic stress at any point in the structure will depend upon the flight conditions (e.g., rotor torques, airspeed, flight control positions etc). The relationship between these will need to be established from strain-gauged development rotorcraft with the object of minimising the number of variables and structural points to be included in the algorithm. It should be recognized, however, that such algorithms are likely to be very complicated if they are to cover all flight regimes, and successful implementation of this technique has yet to be achieved.

7.5 ACTIVE CONTROL OF STRUCTURAL DYNAMIC RESPONSE (ACSR) - HEALTH MONITORING

7.5.1 The relatively large number of accelerometers employed in a system such as that described in Ref 36 provides a good opportunity for monitoring the structural integrity and usage of the airframe and major components. In addition the components of the ACSR system may require health and status monitoring. The structural health monitoring could be performed in the controller which is the heart of the ACSR system, or alternatively within the HUM system. Likewise the health monitoring of the hydraulic or electric actuation system could be performed in the ACSR or in the HUM system, employing the considerations of para 5.4 and 5.5. The controller would be expected to be furnished with full Built-in Test facilities which should be interfaced by the HUM System.

7.6 UNDERCARRIAGES

7.6.1 Health Monitoring

(i) Control of corrosion is expected to remain a ground inspection activity as indicated in para 7.3.3.

(ii) Fractures/damage developing in either skid type or wheeled undercarriages may be detectable before failure by vibration monitoring on the ground (see para 4.8.1) supported by check-list inspections.

7.6.2 Usage Monitoring. The number of take-off landing cycles may provide a first order control of undercarriage lifing. To improve on this would require the measurement or computation of gross weight, sink rate, and for wheeled undercarriages taxi parameters.

8 FUEL SYSTEMS

8.1 SCOPE

8.1.1 This para includes the fuel tanks, fuel lines, filters, pumps, valves, contents gauges, switches, and fuel gauging computers.

8.2 BACKGROUND

8.2.1 A review of the health monitoring of fuel systems is given in Ref 7 A4-21, and Ref 8, p12.

8.3 STATUS MONITORING

8.3.1 Conventional fuel contents monitoring could be supplemented by monitoring as a function of time, engine condition, and transfer algorithms to check for:-

(i) low fuel
(ii) leaks
(iii) failure of automatic fuel transfer.

8.3.2 Fuel pressure downstream of the booster pump could be monitored to detect:-

(i) booster pump failure
(ii) leaks
(iii) fuel filter blockage.

8.3.3 Built-in Test indications on the Fuel Gauging Computer (FGC) should be interfaced by the HUM system.

8.3.4 Fuel filter blockage could be monitored by remote blockage indicator.

8.4 HEALTH AND USAGE MONITORING

8.4.1 Health and usage monitoring of pumps, valves and transfer valve actuation systems could be considered in relation to the safety analysis, redundancy provisions, maintenance policy, and FGC monitoring provisions.

(i) Health monitoring techniques described in para 5.4.2 are appropriate to fuel system components.

(ii) Usage monitoring could be related to total engine operating hours.

9 ELECTRICAL SYSTEMS

9.1 SCOPE

9.1.1 The systems include the batteries, AC generators, Transformer Rectifier Unit (TRU), DC and AC Bus systems, Electrical Anti-icing system.

9.2 HEALTH MONITORING

9.2.1 Temperature monitoring of batteries, generators, and TRU's could provide warnings in advance of overheat STATUS indications.

10 AVIONICS SYSTEMS

10.1 SCOPE

10.1.1 This para includes all avionics systems and data highways installed for the operation and maintenance of the rotorcraft, including the HUM system; internal and external communications systems; and all mission related equipment.

10.2 STATUS MONITORING

10.2.1 It is anticipated that all avionics systems will be furnished with all monitoring functions necessary for establishing operational status and diagnosis of faults.

11 ANTI-ICING SYSTEM (Air)

11.1 SCOPE

11.1.1 Included are all components in heated air flow systems (e.g., for engine intake anti-icing) - principally heaters and valves.

11.2 HEALTH MONITORING

11.2.1 Temperatures could usefully be monitored to provide warnings in advance of failure status indications.

12 CENTRAL WARNING SYSTEM (CWS)

12.1 SCOPE

12.1.1 The system includes the Central Warning Panel, Master caution and warning lights, audio warnings, and any other displays (e.g., Power Systems Display Unit) other than dedicated HUM system displays which are used for cautionary or warning messages. The CW lights and Power Systems data may be included in the same display unit (e.g., Ref 37).

12.2 In addition to monitoring the status of all Central Warning Systems HUM system displays, master warning light and audio warnings could be used to provide back-up CWS functions.

13 EVENT RECORDING

13.1 The HUM system can also be used to supplement subjective assessments be aircrew of abnormalities occurring in the rotorcraft, and external occurrences, by making provision for aircrew initiated event recordings.

13.2 In determining Non-Volatile Memory requirements in the HUM system consideration should be given to the recording of 'snap-shots' of all sensor inputs and processed data, and the number of such events between data down-loads. Such 'snap-shots' should be annotated with time and date information.

13.3 Memory buffers could be continuously up-dated with the above data to provide snap-shot recordings before and after incidents identified by aircrew.

13.4 In certain circumstances, 'snap-shot' recordings could be initiated by the HUM system.

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LEAFLET 727/4

HEALTH AND USAGE MONITORING SYSTEMS

DEVELOPMENT AND VALIDATION TESTING

1 MONITORING FUNCTIONS

1.1 Each monitoring function will be designed to provide varying levels of data and information depending on the nature and criticality of the system being monitored. Methods of certification and validation therefore need to be chosen accordingly.

2 ALGORITHMS AND SENSORS

2.1 The best vehicle for developing and validating algorithms and sensors, is on component rig tests - particularly where tests are continued to failure, and on prototype rotorcraft.

2.2 Where tests to failure are instituted specifically for health monitoring purposes components with naturally produced defects, are preferable to deliberately manufactured (seeded) faults.

2.3 Care should be taken to obtain adequate data at the commencement of test and immediately after each component disassembly if these cannot be avoided.

2.4 Continuous data should be recorded wherever possible, but where not so, representative 'snap-shots' of data at regular intervals should be retained for analysis. Archiving of such recorded data is essential for the development and validation of algorithms.

2.5 Sensor and cable installations should be monitored carefully for reliability and maintainability in addition to performance evaluation in the test rotorcraft environment.

2.6 Algorithms should be evaluated against data from earlier models of rotorcraft where appropriate.

2.7 In parallel with algorithm development in the tests described, criteria and preliminary threshold values should be established for caution and warning status indications.

2.8 In establishing caution and warnings, criteria and threshold values the differences in load and environmental conditions between rig tests and the in-service rotorcraft should be considered.

3 SOFTWARE

3.1 The procedures for testing and validation of airborne software are detailed in interim DEF STAN 00-31 (The Development of Safety Critical Software for Airborne Systems).

4 HUM SYSTEM VALIDATION

4.1 System integration rigs and dynamic system models should be used at an early stage of HUM system validation wherever possible, using recorded rig test data or other rotorcraft data where appropriate.

4.2 Fully integrated system validation should be undertaken on prototype rotorcraft or on Ground Test Vehicles.

4.3 A period of HUM system evaluation on in-service rotorcraft will be advantageous. Where monitored components are subject to fixed Time Between Overhaul (TBO) maintenance, evidence of component condition at overhaul should be used to validate the HUM algorithms and sensor installations. Also the HUM system provisions should be used to reduce the number of sample inspections required for TBO extension.

LEAFLET 727/5

HEALTH AND USAGE MONITORING SYSTEMS

DEMONSTRATION OF COMPLIANCE

1 Demonstration of compliance the requirements of Chapter 727 can be assisted by:

(i) ensuring that a HUM system features in all rotorcraft specification and cost plan documents produced by the Design Authority, including preliminary specifications, and that the scope of the monitoring provisions is defined.

(ii) ensuring that in all design workshare statements relating to collaborative projects the responsibilities for HUM systems, reliability, maintainability, and component monitoring functions are clearly defined.

(iii) ensuring that in any safety assessments carried out (e.g,. Failure Modes and Effects Analysis) specific requirements for health monitoring and usage monitoring necessary to achieve desired levels of safety and mission availability are addressed by the proposed monitoring provisions.

(iv) ensuring that a Development plan for HUM is produced and implemented which satisfies the requirements for developing the algorithms, the sensor installations, and the HUM system. The development plan should also make provision for interface activities necessary for definition of the ground station, and for any early adjustments to caution and warning levels that may be necessary in service.

(v) providing evidence to support the selection and implementation of algorithms adopted.

CHAPTER 730

STRENGTH OF PRESSURISED AIR DUCTS AND PIPES

1 INTRODUCTION

1.1 The requirements of this chapter shall apply to all pressurised air ducts and pipes which are classified as Grade A parts in accordance with the requirements of Chapter 400.

1.2 The requirements for static and fatigue strength are intended to address only those issues which are specific to the strength of pressurised air ducts and pipes and shall be interpreted in the context of the more general requirements of Chapters 200 and 201.

2 STATIC STRENGTH

2.1 DESIGN CONDITIONS

2.1.1 The limit pressure shall be the maximum internal pressure that is likely to arise during one lifetime of usage. When assessing this maximum pressure, account shall be taken of the effectiveness of any control valves and/or relief valves, and the maximum pressure permitted by the setting tolerance of any such valves.

2.1.2 Account must be taken of strains superimposed on the walls of the duct or pipe due to such effects as changes in temperature, the constraints of joints and attachments to the aircraft structure and inertia loading. For the purposes of static design these superimposed strains may be combined and expressed as an equivalent factor on pressure, fp_s, which is applied to the limit pressure. When this approach is followed the Proof and Ultimate factors must be applied to this increased pressure (Limit x fp_s).

2.1.3 The proof pressure shall be at least 1.125 times the Limit pressure (or, where appropriate, Limit x fp_s) unless it is shown that under no circumstances can the limit pressure (or Limit x fp_s) be exceeded. In these latter circumstances a Proof factor of 1.0 is acceptable.

2.1.4 The ultimate pressure shall be at least 1.33 times the Proof pressure.

2.2 TEST CONDITIONS

2.2.1 A pressure test to demonstrate the ultimate static strength shall be done at 1.33 times the Proof pressure on a complete system or on sections which together represent the complete system. The effects of the superimposed strains must be represented. This can normally be done by increasing the test pressure by the factor fp_s.

3 FATIGUE STRENGTH

3.1 DESIGN CONDITIONS

3.1.1 The fatigue pressure spectrum shall be made up of those cycles of pressure that are most likely to occur during one lifetime of usage. When assessing these typical pressures, account shall be taken of temperature effects, any surge/shock pressures and the effectiveness of any control valves and/or relief valves.

3.1.2 Account must be taken of normal levels of superimposed strain due to influences such as those identified in para 2.1.2 above. For design purposes these strains may be combined and expressed as an equivalent factor on pressure, fp_f, that is applied to the pressure spectrum.

3.2 TEST CONDITIONS

3.2.1 A cyclic pressure test shall be done on a complete system or on sections which together represent the complete system. The effects of the superimposed strains (para 3.1.2 above) must be represented. This can normally be done by increasing the test pressure by the factor fp_f.

4 PRODUCTION TESTS

4.1 A proving test shall be done on each production air duct or pipe or on each complete system. The test shall be done to the Limit pressure increased as necessary by any factor that may be necessary to represent the thermal or inertia components of the superimposed strains.

5 DISCHARGE OR LEAKAGE

5.1 Pressurised air ducts or pipes may contain air at high temperature and high pressure. Adequate precautions shall, therefore, be taken to ensure that the discharge of air from relief valves or leakage from components cannot cause the failure of other structure, equipment, or services.

5.2 Pressurised air ducts and pipes shall, where practical, be routed away from vulnerable components such as electrical wiring, hydraulic and fuel pipes. Where this is not practical consideration shall be given to the provision of protective shielding between the air duct or pipe and adjacent components. The need for protection systems which would detect failures and shut off the air supply to the failed duct, such as pressure detectors, or heat detectors in vulnerable areas, shall also be considered. The consequences of failure of any system as a result of the impingement of hot air from a duct or pipe shall be taken into account when deciding on a degree of protection to be incorporated.

5.3 Provision shall be made to facilitate leak checking of pressurised air duct and pipe systems.

6 IN-SERVICE INSPECTION

6.1 Pressurised air ducting or piping in Aircraft is necessarily thin walled, and if unprotected, may be susceptible to external damage which will invalidate its safe fatigue life. Moreover, complex shapes and welded joints are prone to unpredictable fatigue failures. Therefore, any pressurised air duct or pipe which is susceptible to leakage from such causes as the settling of joints, damage by impact, adverse environment or abnormalities in manufacture and/or maintenance or is of complex shape or welded construction, shall satisfy the requirements for ‘inspection-dependent’ structure in Chapter 201 para 3. (See Chapters 800 and 802 for Accessibility, and Ease of Inspection, respectively).

CHAPTER 731

CONDITIONING SYSTEMS

1 INTRODUCTION

1.1 The requirements of this chapter apply to any Conditioning System (CS) installed to condition occupied compartments and specified rotorcraft components.

1.2 A conditioned environment may be achieved by use of Air Cycle Refrigeration, Vapour Cycle Refrigeration (see Note), Ram Air, Compartment Air, Thermo Electric Refrigeration, Combustion Heater, Engine Exhaust, Engine Bleed Air, Electric or other means approved by the Rotorcraft Project Director.

Note: Chlorofluorocarbons (CFCs) shall only be used with the approval of the Rotorcraft Project Director when there are no other suitable compounds or methods available. See The Montreal Protocol on substances that deplete the ozone layer.

2 REQUIREMENTS

2.1 The conditioning system shall be designed to preserve the comfort and performance of the crew and passengers from adverse effects of thermal stress, and to ensure efficient functioning of all equipment under various climatic and flight induced conditions likely to be encountered by the rotorcraft. The rotorcraft manufacturer shall make provision for the complete range of aircrew clothing and equipment prescribed for use in the rotorcraft and for the complete range of rotorcraft activities.

2.2 Consideration shall be given to the effects of, and possible presence of, over temperature, whether due to leakage or system malfunction. Where fluid systems are used, consideration shall also be given to fluid corrosive properties, freezing and boiling, expansion and contraction, heat transfer properties, consequences of leakage including loss of fluid, system capacity and control in respect of the environmental control media. Indication of system malfunction and/or over temperature shall be provided.

3 VENTILATION

3.1 The ventilation airflow of the occupied compartments and equipment compartments during normal system operation shall not introduce contamination but shall be adequate to remove any contaminants that may be present.

100 per cent of the airflow to the cockpit shall be fresh air. A minimum of 50 per cent of the airflow to other occupied compartments shall be fresh air.

Note: Fresh air in this context means 'air drawn in from outside the rotorcraft, i.e. not recirculated,

3.1.1 The ventilation system shall provide airflow for aircrew and passengers which shall not be less than:-

a) 0.25 kg/min. per person in normal operation.

b) 0.20 kg/min. per person in the event of the loss of one source of supply.

The distribution of air to compartments shall ensure that there are no areas where the air movement velocities are less than 3m/min.

3.2 Cabin air distribution shall maintain crew comfort and minimise occupant heat stress. Cabin airflow shall be directed in such a manner that the crew's exhalation is prevented from coming in contact with transparencies. If an air blast type defogging and defrosting system is used, controls shall be installed to permit incoming air for the cockpit to be directed to the windshield front panel and side panels. The air velocity shall not be such as to cause discomfort and the airflow shall not be directed into occupants' eyes. The velocity of air at head level in crew stations shall be maintained below 0.5 m/sec. This velocity shall be measured at the exposed skin, (on the face, when no protective clothing is worn), of each flight crew member when seated. Where protective clothing is worn, these velocities may be considerably exceeded.

3.3 The ventilating system controls shall be convenient to the pilot or flight crew.

3.4 Where individual air ventilators are provided, they shall be manually operated by controls convenient to the occupant. Each outlet shall have control of volume and flow direction of air delivered and shall be water tight in the closed position.

3.5 Air intakes essential to the operation of the air supply system shall be suitably protected against blockage from ice, dirt, etc., and shall be located so as not to introduce contamination into the ventilating system.

3.6 Compartments susceptible to contamination from smoke, gun gas and fuel vapours etc., shall be sealed from adjacent compartments.

3.7 If an exhaust heat exchanger is used for heating ventilating air, means shall be provided to preclude the harmful contamination of the ventilating air, e.g., provision of a secondary heat exchanger.

3.8 Galley and toilet areas shall be separately vented with fresh air.

4 TEMPERATURE

4.1 The rotorcraft manufacturer shall aim to maintain a mean skin temperature of 33°C for each crew member. Achievement of this mean temperature shall take into account all relevant factors such as the temperature of the rotorcraft skin, the cockpit or cabin wall insulation, the quantity and temperature of cooling air supplied, the amount of solar radiation, the type of crew clothing and the crew members' rate of work. The cabin or compartment temperatures will be less than the mean skin temperature (33°C) as shown in Fig 1.

4.1.1 The average temperature of occupied compartments shall be maintained within the limits of curves 1 and 2 of Fig 1. The Wet Bulb Globe Temperature (WBGT) in the vicinity of the head and shoulder levels of crew stations shall not be greater than 32°C.

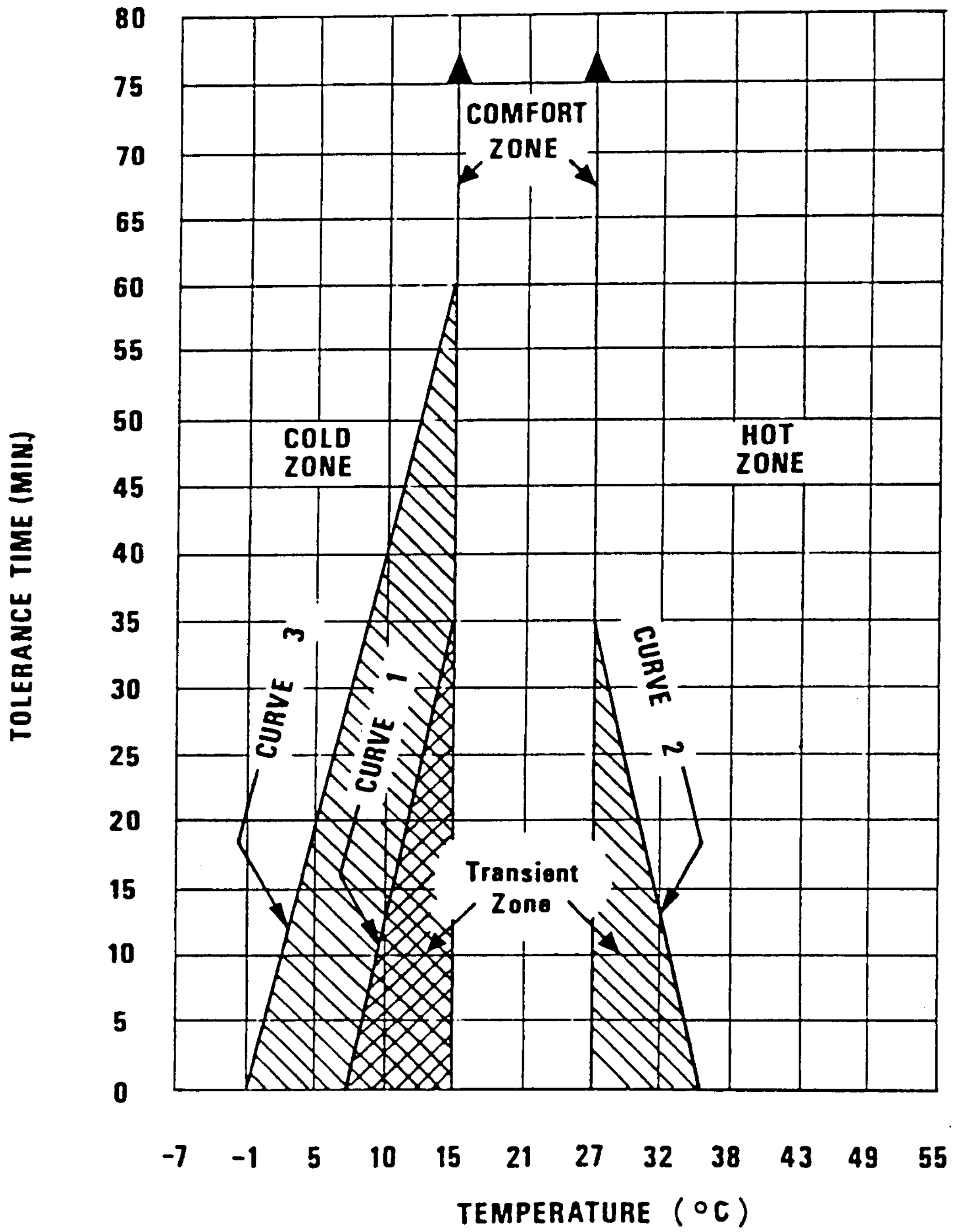

FIG 1 THERMAL REQUIREMENTS FOR OCCUPIED COMPARTMENTS

$$WBGT = 0.7T_{wb} + 0.2T_{bg} + 0.1T_{db}$$

Where T_{wb} = wet bulb temperature

T_{bg} = black globe temperature from a 5cm sphere, and

T_{db} = dry bulb temperature

For ambient environments which exceed the performance limits of the cooling system but are within the operations limits, the pilot envelope temperatures of crew stations shall be maintained within the thermal tolerances specified by the Rotorcraft Project Director.

4.2 The temperature in occupied compartments shall be so controlled that during steady state conditions and with a single failure, the air temperature entering that compartment shall not be greater than 93°C.

4.3 The design and construction of the heating system shall be such that the temperature of the air delivered from the heat exchanger or combustion heater under any conditions of flight shall not exceed 170°C measured at a point in the ventilating air duct 500 mm downstream of the heat exchanger or combustion heater.

4.4 No surface which can be touched by the flight crew or occupants shall exceed a temperature of +60°C or fall below -15°C when the conditioning system has reached its stabilized operating condition.

4.5 Temperature achieved under normal and failure conditions shall not cause material used in the construction of the rotor-craft to deteriorate or to change its physical properties such that its operation is impaired.

5 POWER SUPPLIES

5.1 Means shall be provided to limit the maximum power off-take demand from the power source(s). When necessary power for propulsion shall be given priority.

6 CONTROLS

6.1 Means shall be provided to enable the pilot/flight crew to control the temperature within the occupied compartments. Where applicable the pilot/flight crew shall have separate control of the cockpit/flight crew compartment. These controls shall be readily accessible to crew members.

6.2 Where separate control is provided for other components, this shall be independent of the facility for the flight crew to control conditions in the flight crew compartment.

6.3 Automatic temperature control systems shall have either a duplicated or a back-up control system.

6.4 The controls for starting the combustion heater system in operation shall be simple and the shut off control on the heating system shall consist of a single control for shutting off all the system.

6.5 A bleed air heating system shall have a single control to completely shut off the flow of bleed air to the heating system. An indication of over temperature conditions shall be provided.

6.6 The design and location of the controls and indicators shall be subject to rotorcraft mock-up, inspection, and approval by the Rotorcraft Project Director.

6.7 Electrical heating devices shall have automatic overheat cut out in addition to temperature control.

7 INDICATION

7.1 Indication shall be provided to alert the crew of unsafe system operating conditions, and to enable them to take appropriate corrective action. Systems, controls and associated monitoring and warning means shall be designed to minimise crew errors which could create additional hazards.

7.2 Warning and monitoring devices shall be in accordance with Chapter 105, para 12 and/or the Rotorcraft Specification.

8 HUMIDITY

8.1 Humidity of cockpit/flight deck or cabin shall be in accordance with Chapter 101, para 2. For instruments/displays the humidity limits shall be in accordance with Chapter 724, para 4.4 and for avionic equipment shall be in accordance with Chapter 725, para 6.4.

8.2 The system shall be designed to prevent discharging air with entrained moisture onto windshields or into pressure suits, occupied compartments, electronic equipment compartments or forced cooled electric equipment. All air delivered to the compartments or equipment by any means, including ram air, shall meet these requirements, except where ram air is used as an emergency backup.

9 NOISE CONTROL

9.1 Acoustical noise of the air conditioning system when combined with acoustical noise from all other sources shall not be setae than the maximum acceptable levels of Chapter 108.

10 GROUND OPERATION

10.1 Conditioning shall be provided to meet the rotorcraft requirements when the main engines are not running. For example, by A.P.U. or ground servicing equipment.

10.2 When external sources are required, the conditioning system shall be compatible with existing GSE cooling units.

11 EQUIPMENT CONDITIONING

11.1 An environment suitable for the limitations of installed equipment shall be in accordance with Chapter 706, para 9, Chapter 709, paras 5 and 8, Chapter 710, para 3.2, Chapter 724, paras 4.3 and 4.4 and Chapter 725 paras 6.3 and 6.4.

11.2 Equipment temperature limits shall be observed. Where there is a danger of atmospheric condensation freezing and so impairing equipment operation, adequate drainage, ventilation or heating shall be provided.

11.3 Where closed loop or re-circulating fluid systems for cooling of Avionic equipment, optical units, power supplies, weapons, etc., is fitted, this shall be in accordance with individual equipment specification. If an inflammable fluid is used, fire protection shall be provided in accordance with Chapter 712.

12 INSTALLATION

12.1 GENERAL

12.1.1 Installation of the rotorcraft conditioning system shall be designed to operate in accordance with the Rotorcraft Specification, and to withstand conditions occurring in service.

12.1.2 Due consideration shall be given in design of an installation to the effects of thermal expansion and contraction.

12.1.3 All materials and working fluids used in the construction and operation of a heating or cooling system shall be suitable for the working temperatures involved.

12.2 HEAT EXCHANGERS

12.2.1 Heat exchangers shall be constructed of corrosion-resistant materials suitable for temperatures and pressures encountered.

12.2.2 Heat exchangers which dissipate heat from the bleed air or compressed ram air into fuel or coolants other than water, shall be so designed that a single structural failure will not result in leakage of fuel or coolant fluid into the air supply.

12.2.3 When fitted, each exhaust heat exchanger shall be constructed and installed to withstand each vibration, inertia and other loads to which it would be subjected in operation and shall be suitable for continued operation at high temperatures. It shall be resistant to corrosion from exhaust gases.

12.2.4 No heat exchanger shall be installed in such a way as to permit an accumulation of flammable liquids or vapours either in service or as a result of a malfunction or leak.

12.3 SHUT OFF VALVE

12.3.1 A shut-off valve shall be installed which will permit the crew to shut-off the conditioned air supplied by each sub-system for general cabin conditioning. For any degree of emergency the valve shall be closed in less than two seconds.

12.3.2 Provision shall be incorporated to close the shut-off valve in the event of an overheat condition or control system failure. The temperature setting and response time shall be selected to protect the components from damage or service life degradation but to preclude shutdowns during transients occurring in normal operation. The designer shall determine the best location for sensing the overheat condition.

12.4 FLOW RATE ADJUSTMENT

12.4.1 The flow control system shall allow for increasing or decreasing the airflow schedule within the limits stated in the Rotorcraft Specification.

12.5 MAXIMUM AND MINIMUM FLOW

12.5.1 The flow control system shall ensure that sub-system flow will not exceed the value specified in the Rotorcraft Specification and the minimum flow rate will always be met or exceeded for all ground and flight conditions.

12.6 AIRFLOW CONTROL SYSTEM ENVIRONMENTAL CONDITIONS

12.6.1 The effects of humidity, freezing of condensate in line, high and low energy starting, local heating and vibration, and pressure and thermal shock shall be carefully considered in component design and shall not deleteriously affect their operation or service life. Control devices including orifices shall be designed to prevent malfunctions due to freezing of moisture, corrosion, or contamination within the devices.

12.7 MOISTURE CONTROL

12.7.1 The sub-assembly shall be constructed so that all lines and ducting are located or fitted with proper insulation, draining and venting to preclude moisture accumulation.

12.8 WATER SEPARATOR

12.8.1 Water separators, when installed for moisture control, shall have adequate drainage or a by-pass valve shall be provided where an automatic means of preventing icing of the separator is used.

13 THERMAL INSULATION

13.1 Consideration shall be given to the use of thermal insulation in conjunction with heating equipment and ducting in order to prevent excessive heat loss (for fire precautions see Chapter 712).

13.2 Insulating material shall not be hygroscopic.

14 FIRE PRECAUTIONS

14.1 Fire precautions shall be in accordance with Chapter 712.

15 NUCLEAR, BIOLOGICAL AND CHEMICAL (NBC) REQUIREMENTS

15.1 Where required, satisfactory air filtration of radioactive particles and chemical and biological warfare agents shall be provided in accordance with the Rotorcraft Specification requirements (see Chapter 717).

16 FAILURE EFFECT AND MODE ANALYSIS

16.1 A failure effect and mode analysis shall be prepared for the conditioning system and associated components.

17 TESTS

17.1 The functioning of the system in the aeroplane shall be in accordance with Chapter 1007.

18 CROSS REFERENCES

18.1 A number of requirements directly related to Conditioning Systems (CS) appear elsewhere in this publication. The most important are listed in Table 1.

TABLE 1

LIST OF OTHER IMPORTANT REQUIREMENTS

CHAPTER	PARAGRAPH	SUBJECT
101	1	(Temperature limits for design purposes.
	2	(Humidity Conditions
103	2	Operational colouring and marking.
104	5	View and clear vision.
105	-	Crew stations - general requirements.
107	5	Pilots cockpit - control and instruments.
108	-	Internal noise.
112	-	Reduction of vulnerability to battle damage.
203	-	Control systems - mechanical components.
206	-	Design to resist birdstrike damage.
307	2	Crash landing and ditching.
400	-	Detail design and strength of materials.
401	-	Detail data for metallic materials.
402	-	Processes and working of materials.
404	-	Marking of rotorcraft parts.
405	-	Exfoliation corrosion of aluminium alloys.
406	-	Stress corrosion cracking.
407	-	Precautions against corrosion and deterioration.
408	-	Plastics materials.
600	3	General requirements and definitions.
702	12	Fuel systems.
706	-	Electrical installations.
708	-	Bonding and screening.
709	8	Gun installations.
710	-	Armament installations.
711	-	Ice protection.
712	10	Fire precautions.
724	-	Instrumentation/Display installations.
725	-	Avionic Equipment installations.
806	7	Marking and notices.
1007	3,4,5	Flight tests - installations and structures. (Conditioning Systems).

LEAFLET 731/0

CONDITIONING SYSTEMS

REFERENCE PAGE

British Standards

C10	..	..	..	Coupling dimensions for aircraft Ground Air-Conditioning Connectors.
R.A.E. TECHNICAL NOTE No. 2196		..	..	Cabin Air Conditioning in high speed flight.
R.A.E. TECHNICAL NOTE No. 68304		..	..	Cabin air requirements for crew comfort in Military Aircraft.

British Civil Airworthiness Requirements

CHAP D6-11	..	..	Ventilation and pressurisation of crew and passenger compartments.

JAR's

JAR 25-859	..	..	Combustion heater fire protection.
JAR 25-1125	..	..	Exhaust heat exchangers.

Misc

APOSH STD 161-8	..	..
DRAWING AND 10375		..

US Military Specifications

MIL-A-83116 (USAF)		..	Air Conditioning system.
MIL-B-81365 (WP)	..	..	Bleed Air System - General Spec.
MIL-C-7762	..	..	Compass - Installation.
MIL-D-7890 (AS)	..	..	Design & Instln - Anti G Suits in jets.
MIL-E-7080	..	..	Installation of Electrical Equipment.
MIL-E-18927 (A)	..	..	Environmental Control Systems.
MIL-H-5484	..	..	Heater - Combustion Type.

MIL-I-8670 (AS)	..	..	Installation fixed guns & Equip. in naval aircraft.
MIL-I-8673 (Aero)	..	..	Installation & Test - flexible weapon system.
MIL-M-8609	..	..	Motors direct current.
MIL-N-18325 (Aer)	..	..	Heating & ventilating systems.
MIL-O-27210	..	..	Oxygen - aviators breathing liquid & gas.
MIL-P-6992 (Aer)	..	..	Pumps Assy's fuel electric aircraft heater.
MIL-T-18606 (Aer) 1	..	..	Test procedures - cabin Press & Air Cond.
MIL-T-23103 (AS) Amdt 2		..	Thermal performance Evaluation Electronic Equipment.
MIL-STD-210	..	..	Climatic information.
MIL-STD-454	..	..	General requirements for Electronic Equipment.
MIL-STD-1472	..	..	Human engineering design criteria.

Nato Standards

STANAG 3208 (ED4)		..	Air Conditioning Connections
STANAG 3611 GGS	..	..	(Characteristics of Compressed (Breathing Air.

LEAFLET 731/1

CONDITIONING SYSTEMS

HEATING REQUIREMENTS

1 INTRODUCTION

1.1 This leaflet provides details of some of the current heating methods available and gives guidance on their use.

1.2 Sufficient heating capacity should be provided for occupied compartments and for compartments housing cold sensitive equipment to achieve an even distribution of temperature throughout the compartment and having the capability of stabilising within five minutes of starting or as stated in the Rotorcraft Specification.

1.3 When the heating system used for de-icing the wings and tail surface is combined with the cabin heating system, that pertaining to the thermal de-icing should conform to the requirement of Chapter 711.

2 HEATING EQUIPMENT (Combustion)

2.1 The combustion heater and associated fuel pump must be approved and meet the fire protection requirements of Chapter 712.

2.2 The combustion heating system should be designed and constructed so that it will ignite and operate continuously under any conditions of flight from sea level to the service ceiling of the rotorcraft or to such limitations as the Rotorcraft Project Director may specify.

2.3 Provision should be made to ventilate adequately and continuously the compartments in which combustion-type heaters are installed in order to prevent the accumulation of fuel vapours in the event of fuel leakage. The ventilation rate, which should be related to the volume of the compartment as well as the capacity of the heater, should exceed 0.076 kg of air per minute per kilowatt rated thermal capacity of the heater.

3 HEATING EQUIPMENT (Engine Exhaust)

3.1 Suitable means should be provided to permit the flow of sufficient air through the heat exchanger at all times to prevent premature failure due to overheating. The valves used to control the flow of heated air should be of such construction and ruggedness as to preclude warpage or leakage for any reason during the life of the heating system.

3.2 If a secondary air-to-air heat exchanger is used, means should be provided for regulating the flow of the air being used in the secondary exchanger. The design and construction of primary exhaust-to-air heat exchanger should be such that vibration or expansion and contraction due to rapid heating and cooling should not cause cracking of the heat exchanger or the exhaust stack.

4 HEATING EQUIPMENT (Engine Bleed Air)

4.1 Temperature regulation of cabin inlet air may be accomplished by a suitable air-to-air heat exchanger, although other methods may be acceptable if approved by the Rotorcraft Project Director.

4.2 On multi-engined rotorcraft, the conditioning system should be capable of maintaining operation within design limits with loss of air supply from one of the engines. In the case of a duct or component failure, a means should be provided which will prevent air extraction from the engine at a flow rate greater than that allowed by the engine manufacturer.

5 HEATING EQUIPMENT (Electric)

5.1 Electric heaters may take the form of radiant heaters or heating mats.

5.2 Overheat protection should be provided for all electric heaters. This should be in addition to the primary temperature control systems.

5.3 Surface temperatures exceeding 250°C will only be permitted where possible contamination from ignitable vapours does not exist.

5.4 Where radiant heaters are located in air ducts, flow detectors or temperature limiters (i.e. thermostats or thermal fuses) should be provided to prevent over-heating.

LEAFLET 731/2

CONDITIONING SYSTEMS

COOLING REQUIREMENTS

1 INTRODUCTION

1.1 This leaflet provides details of some of the current cooling methods available and gives guidance on their use.

1.2 Sufficient cooling capacity should be provided for cockpits/flight decks and compartments and for compartments housing heat sensitive equipment.

1.3 The source of cooling may be air cycle, vapour cycle, thermoelectric or any other method approved by the Rotorcraft Project Director.

2 COOLING REQUIREMENTS

2.1 When external ground cooling is required by the Rotorcraft Specification, the hose coupling should be located for easy access by ground personnel.

2.2 Sun shields may be provided for overhead cabin transparencies. These should be readily removable, should be of transparent infra-red reflecting type and should reduce the transmission of incident infra-red radiant energy by at least 60 per cent.

2.3 Where sufficient airflow is not available during ground and low speed flight operations, a fan or bleed air ejector, which is automatically shut off when not required, should be used for inducing airflow through the coolant circuit.

2.4 Automatic temperature control is required for all cooling circuits. Each circuit should provide an over temperature indication to the aircrew.

2.5 The cooling capacity required should be determined with respect to:-

(i) Heat input to the compartment. Possible sources are, solar heat, high rotorcraft skin temperature, operating requirements, leakage from adjacent compartments and crew members body heat.

(ii) The nature and pattern of cooling air distribution.

(iii) Crew clothing and the crew work rate entailed.

LEAFLET 731/3

CONDITIONING SYSTEMS

CONTAMINATION

1 INTRODUCTION

1.1 This leaflet details some of the methods to avoid contamination of the atmosphere in occupied compartments.

1.2 The system should prevent the concentration of harmful or irritating substances in the occupied compartment atmosphere.

2 CONTAMINATION REQUIREMENTS

2.1 The rotorcraft manufacturer should consider the adverse effect of smoke concentration with regard to vision and the viewing and operation of essential flight controls when locating inlet and extraction devices.

2.2 The air intake(s) should be so arranged that in normal or any fault conditions, contaminated air will not be supplied to occupied compartments or to other regions to such an extent as to create a hazard.

2.3 The air supply to occupied compartments should be ducted when passing through compartments that are inaccessible in flight.

2.4 Precautions should be taken to preclude the contamination of air in occupied compartments resulting from the operation of the rotorcraft in normal and emergency conditions. In particular:-

(i) Carbon monoxide concentrations should not exceed 50 parts per million by volume for any period exceeding 5 minutes.

(ii) Carbon dioxide concentrations should not exceed 30,000 parts per million by volume for any period exceeding 5 minutes.

(iii) Hazardous concentrations of fire extinguishing agents should not be liable to occur either as a result of intentional use of a fire extinguishing system or portable extinguishers, or as a result of any failure which might lead to unintentional discharge of the extinguishant.

(iv) Systems employing fluids liable to give off noxious or toxic vapours or substances (e.g. some de-icing and hydraulic fluids) should not be installed in such a manner as to risk hazardous contamination of the cabin air either by leakage or by use.

(v) No materials which give off noxious fumes when heated should be used in such a way that they may become heated in normal or failure conditions to the extent that the cabin air would become dangerously contaminated.

(vi) Electrical equipment should be so constructed and installed that in the event of failure, no dangerous quantities of toxic or noxious products will be distributed in the crew compartment.

2.5 No failure having a probability of occurrence greater than 1 x 10^{-7} per flying hour should result in leakage into the air supply (e.g. from a cabin air heat exchanger), which would hazard the rotorcraft or its occupants.

2.6 In normal operation or after a failure in the power-plant, it should be shown that dangerous concentrations of harmful substances cannot occur (or in the case of power-plant failure can be prevented from occurring) in occupied compartments, e.g, where the cabin air supply is obtained from a direct tapping on a turbine engine.

2.7 Where a recirculating ventilating system could cause hazardous contamination of compartment air in either normal or fault conditions, it should be possible to stop the recirculating system and still show compliance.

2.8 Where the air supply is used, for example, to cool devices, it should be free from any contamination which could adversely affect their functioning.

LEAFLET 731/4

CONDITIONING SYSTEMS

CONTROLS

1 INTRODUCTION

1.1 This leaflet amplifies the control requirements of Chapter 731, para 6.

2 CONTROL CONSIDERATIONS

2.1 Independent control of compartments other than crew compartments is not required if all of the following conditions are met:

(i) The total volume of the flight crew and passenger compartments is $25m^3$ or less.

(ii) The air inlets and passages for air to flow between flight crew and passenger compartments are arranged to provide compartment temperatures within 3°C of each other and adequate ventilation to occupants in both compartments.

(iii) The temperature and ventilation controls are accessible to the flight crew.

2.2 The controls for the ventilation system should be of such design and construction that a minimum of leakage results when the system is closed.

2.3 Means should be provided for operating the heating system at partial output. Suitable, concise, clear instructions for the operation of the heating system should also be incorporated in a nameplate or mounted adjacent to the heating system controls. If the ventilating system controls are combined with the heating controls, suitable instructions for the operating of this system also should be incorporated.

2.4 Combustion heating systems should incorporate a suitable automatic control located in the fuel line as near to the fuel take-off from the main fuel system as possible, to stop the flow of fuel if the pressure in the fuel line downstream of this point is reduced by leakage or by being broken.

2.5 For any system conditions causing flow disturbances, flow overshoot or undershoot should be not greater than ten per cent of the value existing immediately before the disturbance and the flow should be within five per cent of the final control level within two cycles or ten seconds, whichever is less, following completion of the conditions causing the disturbance.

2.6 The temperature selector for automatic cabin temperature control should be designed for the range 5°C above and below the comfort zone. The aim should be to maintain the temperature in each compartment within ±2°C

CHAPTER 732

ROTOR SYSTEMS

1 INTRODUCTION

1.1 The requirements of this chapter cover the design and testing to be met by Rotorcraft Rotor Systems and apply equally to all configurations of main and tail Rotor Systems unless stated otherwise.

1.2 DEFINITION

1.2.1 The Rotor System shall be considered to include all parts, excluding engines, shafts and gearboxes necessary to provide a Rotorcraft with controlled thrust for lift, manoeuvring, forward flight and for countering the effects of torque reaction. This definition includes helicopter main rotor and tail rotors and prop-rotors of tilt wing or tilt rotor aircraft. The System shall comprise the rotor hub assemblies (to include hydraulic and electrical systems for powered blade folding), blades, means of transmitting movement to provide blade pitch control (including pitch control shafts and/or mechanisms housed with gearboxes), lubrication systems and electrical connections.

1.3 SCOPE

1.3.1 The System is bounded by connections to the rotor drive system via the rotor hub and the rotor drive shaft where appropriate. Blade pitch control mechanisms are bounded by connections from the control system servo actuators to the blades for powered controls and from gearbox mounted controls to the blades for manual control mechanisms. Electrical connections comprise those for bonding, electric blade folding, ice protection and any connections for health and usage monitoring systems. Hydraulic pressure systems comprise installations on the rotor head, such as those used for blade folding. The Chapter does not cover hydraulic and electrical input connections to servo actuators; for these details see Chapters 704 and 706.

1.4 OPERATING CONDITIONS

1.4.1 The Rotor System shall be suitable for and function over, the range of normal operating, environmental and emergency conditions for which the Rotorcraft is designed as defined in the Rotorcraft specification.

2 INSTALLATION INTERFACES

2.1 INTERCHANGEABILITY AND MAINTENANCE

2.1.1 Attachments Points

The attachment points for all items within the Rotor System shall be designed to make their incorrect assembly impossible (see Chapter 100 para 7 and Chapters 804 and 805). Any joint within the rotor system, the failure of which would be catastrophic, should be designed such that any inherent rotation of the joint, if the locking mechanism were to fail, would not cause the joint to loosen or the joint to fail at any rotor speed.

(i) Attachment to Rotor Drive System
The rotor head shall be designed with consideration for its removal and replacement and the ground support equipment to be used.

(ii) Rotor Blade Attachment
Rotor blades, where separable from the rotor head, shall be designed to enable blade removal, replacement, and handling without undue risk to damage. The use of a single fixing shall be prohibited. Rotor blades shall be capable of being replaced in the spread or folded position in all operating conditions.

(iii) Control System Attachment
The control circuit stiffness is important to the helicopter dynamics and stability. The rotor system designer shall specify the allowable range of control system stiffness. The designer shall ensure that no adverse control couplings or instability arises as a result of deflections of the control mechanism under the full range of operating loads. Consideration shall be given to the effects of stiffness variation of controls due to assembly and deterioration in service use.

(iv) Blade Folding Systems Attachment
The effect of any blade folding mechanisms shall be included in the assessment of the static and dynamic characteristics of the Rotor System. Blade folding systems shall ensure that the rotor system maintains a consistent geometry between folding operations such that the rotor track and balance is not affected.

(v) Electrical Installations
All electrical cables within the Rotor System shall be supported in such a manner that damage to cables and structure is prevented.

2.1.2 Accessibility and Ease of Maintenance
The Rotor System shall be designed to minimise routine servicing between specified overhauls (see also paras 3.3.14 and 4.1.11 and Chapters 800, 801, 802, 803 and 804).

3 SYSTEM DESIGN

3.1 SYSTEM ARRANGEMENT

3.1.1 Design of Rotor Angular Motion Capability
An analysis shall be performed of all the likely flight and ground cases, including limit conditions, manoeuvre gusts, torque applications either deliberately applied by the pilot or suffered as a result of single failures, to determine the ranges of flap, lag and pitch or torsion motion which the rotor will be required to accommodate.

3.1.2 Avoidance of Rotor Interaction

The operating planes of the rotors shall not interact such that loss of transmission synchronisation could result in contact between rotors. Consideration shall be given to all possible combinations of blade angles and wind conditions that might occur.

3.1.3 Phasing of Rotors

See Chapter 705 paragraph 3.1.6.

3.1.4 Rotor Clearance

(i) Structure Clearance

Adequate clearance shall be provided between any Rotor System and all other parts of the Rotorcraft by considering the combined effects of foreseeable manoeuvring and gust loads, such that the possibility of the blades striking any part of the Rotorcraft shall be consistent with the requirements set out in the Rotorcraft Specification. Particular attention shall be paid to wind conditions during rotor starting and stopping operation and when applicable helicopter motion, such as from deck motion, during shipboard operation when the rotor speeds are low.

(ii) Ground Clearance

Droop stops or other means shall be provided to ensure that the rotors do not contact the landing surface during a normal take off or landing as specified in the Rotorcraft spec.

(iii) Propeller Clearance

For compound type Rotorcraft utilising conventional propellers for forward flight adequate clearance shall be provided between the propeller or propellers and the rotors to satisfy those conditions described in para 3.1.4 (i).

Note: Propeller to structure clearance is outside the scope of this chapter. (Refer to Vol 1 Leaflet 311/1 para 3.1, BCAR G5-1 para 6 and G5-1 App para 3).

3.1.5 Blade Folding Arrangement

It shall be impossible for main and tail rotor blades to contact the Rotorcraft structure during or after blade folding operations over the full range of conditions applicable to blade fold as set out in the Rotorcraft Specification. Rotorcraft provided with tail fold installations shall have means to prevent tail rotor blades striking main rotor blades during blade fold operations for automatic blade fold systems. A positive indication shall be provided to the pilot when it is safe to commence folding and when the operation is complete.

To prevent windmilling of a tail rotor in gust conditions when the pylon is folded there shall be means to lock the tail rotor in a predetermined position (see also paras 5.6, 5.7 and 6.1.11 and Chapter 100 para 17, Chapter 107 Table 4 item 2 and Chapter 722).

3.1.6 Incorporation of other Functions
The mounting of accessories on, or driving accessories from a Rotor System shall be permitted only if it does not affect the system operation adversely or constitute a hazard due to any failure of the accessories or their connections to the Rotor System.

3.1.7 Rotorcraft Weight and Weight Distribution
The Rotor System shall function at all practicable weights from the Rotorcraft Design Minimum Weight to the Design Maximum Weight when the centre of gravity of the Rotorcraft is in the most adverse position compatible with the weight assumed.

3.1.8 Vibration
The rotor system shall be designed to ensure that the vibratory forces and moments produced by the rotor are such that, in conjunction with the likely airframe response, the vibration requirements of Chapter 501 and the Rotorcraft application can be satisfied.

3.1.9 Structural Distortion, Flutter and Ground and Air Resonance
The Rotor System and its associated mountings shall be designed such that the likely variation in stiffness during the life of the Rotorcraft will not have an adverse effect of the potential Rotorcraft instabilities. See also Chapter 500.

3.1.10 Rotor System Noise
Rotor system design shall consider minimization of noise transmitted to the aircraft interior or to the aircraft external surroundings. Consideration shall include effect on crew fatigue and work load, general nuisance or annoyance to external observers, as well as stealth aspects. Parameters that shall be considered will be rotational speed, direction of rotation, blade detailed geometric design and rotor to rotor, rotor to airframe interaction with an aim of balancing the lowest noise levels with conflicting requirements of weight and performance. (For requirements relating to Rotorcraft internal noise control see Chapter 108 and Chapter 501 para 3).

3.1.11 Control System Arrangements
The Rotor System control elements shall be designed to satisfy the appropriate requirements of Chapter 203.

3.1.12 Reliability
Rotor System reliability programme shall be in accordance with Def Stan 00-40 and 00-41 or the Rotorcraft Specification where requirements differ.

3.1.13 Operation Following Rotor Drive System Failure
The Rotor System shall continue to function satisfactorily following non-simultaneous failure of any combination of power units and their drive systems up to associated freewheels; to permit rapid entry into autorotation and allow a controlled descent to a landing.

3.2 SYSTEM DYNAMICS

3.2.1 An analysis is to be carried out on the Rotor System Dynamics to determine the modes of vibration and subsequent loads arising. It shall also demonstrate safe operation of the system, including freedom from adverse couplings and instabilities. The model used in the analysis is to include or take account of; the fuselage, the control circuit and the drive system as well as the rotor itself.

3.2.2 A further analysis shall be carried out to determine the sensitivity of the rotor system dynamics to possible variations in the parameters of the rotor, such as control circuit stiffness, blade chordwise CG, gearbox mounting stiffness, damper characteristics etc. Account is to be taken of variations occurring during service life due to such phenomena as: bearing wear, backlash, lag damper degradation, elastomeric bearing creep and stiffness changes, balance changes due to moisture absorption of the rotor blades etc.

3.2.3 The dynamic model of the Rotor System shall be validated by ground and flight test. The model shall then be modified as required to enable predictions of failure cases to be adequately simulated in support of the helicopter safety analysis.

3.3 SYSTEM SAFETY

3.3.1 Safety Assessment

A safety assessment of the Rotor System shall be carried out. The analysis shall include:

(i) Possible modes of failure

(ii) Possible multiple failures and undetected failures.

(iii) The resulting effects on the Rotorcraft and aircrew.

(iv) Aircrew and maintenance crew warning cues.

(v) Events and errors that include, but are not limited to, external events, flight crew errors, maintenance errors etc.

The extent, depth and form of the analysis shall be stated in the Rotorcraft Specification. However; for failure modes with effects that are hazardous or catastrophic, it shall identify the compensating provisions that minimise probability of occurrence (e.g. establishment of safe fatigue lives, enhanced QA procedure, health monitoring, qualification test evidence etc.).

3.3.2 Protection from Sudden overload

(i) The design of the rotor system shall consider the likely modes of failure of components due to sudden overtorque produced by the rotor drive system and the use of components where failure results in rupture rather than permanent set should be avoided unless it can be shown that such an overtorque is extremely remote.

(ii) The display of suitable markings or placards to the aircrew shall be sufficient means of complying with paragraph 3.3.2(i) if the inherent characteristics of the Rotorcraft and instrumentation are such that the pilot can safely be left to provide the protection.

3.3.3 Rotor Overspeed
The Rotor System shall be shown to withstand centrifugal loads corresponding to overspeed of a minimum of 120% of the rotor max steady power-off rotor RPM without failure.

3.3.4 Electrical Bonding for Lightning Discharge Protection and Electrical Screening

(i) Adequate electrical bonding shall be provided during the design of the Rotor System. All conducting parts which may intentionally or unintentionally carry a proportion of lightning strike currents shall be provided with low impedance bonding back to the aircraft structure. Components which may be subject to damage from arc attachment shall incorporate appropriate surface protection. Electrical components and wiring within the Rotor System shall be designed so that any failure resulting from a lightning strike shall occur in a non-hazardous manner and such that the effects of a lightning strike shall not propagate into other aircraft electrical systems. Rotor System components shall be designed so that full functioning is not impaired, for a period not less than the maximum Rotorcraft flight duration, in the event of such components being subjected to lightning currents.

(ii) Bonding leads fitted to Rotorcraft Systems shall have sufficient strength to withstand the forces applied due to rotation, air loads, vibration and additional loads produced by the presence of ice accretion. Such flexible bonding leads shall be installed so that movement occurs through twisting rather than bending.

3.3.5 Climatic Conditions
For climatic conditions under which a Rotor System shall operate see Chapter 101.

3.3.6 Ice Protection
Areas where moisture can collect and cause control restriction or jam any parts of the Rotor System in icing conditions shall be eliminated or adequately drained. These requirements shall be applied to the Rotorcraft non-operating condition and all operating attitudes where the effects of centrifugal force on moisture and ice shall also be taken into account (see also Chapter 711 and paragraph 3.3.7(i)).

3.3.7 Protection from Contaminants

Rotor System components shall be designed to resists the effects of external contamination.

(i) Moisture

Components shall be designed to withstand continuous exposure, in the configuration as installed in the Rotorcraft and either operating or non-operating, to rain, sleet, snow, slush, salt sea-spray etc. As a design aim moisture ingress due to centrifugal force effects is to be prevented and the design should ensure that no moisture can enter components during washing operations. Satisfactory functioning shall be established by testing to BS 3G.100* for Waterproofnesss and Salt Mist Test Severity. Where drains and drain holes are used to remove moisture the blockage of a single drain or drain hole shall not prevent adequate drainage (see para 3.3.5 for freezing conditions).

*To be progressively superseded by DEF STAN 00-35.

(ii) Mould Growth

Components shall be inherently resistant to mould growth and microbiological attack (see also Def Stan 00-29).

(iii) Dust and Sand

Rotor System components shall be designed to resist or be protected from internal or external erosion caused by airborne particles of dust, sand etc. Def Stan 07-55 (Part 2) Section 4 specifies test methods for the ingress of dust and sand (see also Chapter 101 para 7).

(iv) Fluids

Components shall function correctly and not be damaged as a result of being externally contaminated (leakage, spillage, splashing, handling) by hydraulic fluid, oils, fuels with or without anti-static additives, de-icing fluid, cleaning fluids etc. approved for use with the Rotorcraft. BS 3G.100:Part 2* gives details of contaminating fluids and tests to be carried out on components. Information concerning chemical attacks is provided by DEF STAN 00-50.

*To be progressively superseded by DEF STAN 00-35.

(v) Exposure to Sunlight

Materials and finishes subject to degradation in sunlight shall be kept to a minimum. This shall include, but not be limited to, rubber, certain plastics, lacquers, pigmented paints and some forms of safety glass. Testing of materials shall be carried out to Def Stan 07-55 (Part 2).

3.3.8 Corrosion and Deterioration

See Chapter 407 for requirements relating to corrosion and deterioration.

3.3.9 Shock

The Rotor System in the non-operating condition, may be subjected to relatively infrequent non-repetitive shocks such as those that might occur during handling or maintenance operations. Shocks may also occur as the Rotorcraft is being towed on the ground. The System and its components shall be resistant to these shocks whose duration and peak acceleration shall be stated in the Rotorcraft Specification. Consideration should also be given to the use of monitoring devices.

3.3.10 Bird Strikes

Requirements for Rotor System bird strike resistance are given in Chapter 206.

3.3.11 Ice Strikes

Requirements for Rotor System ice strike resistance shall be as stated in the Rotorcraft Specification. (See also Chapter 711 para 2.2.2).

3.3.12 Battle Damage

Battle damage tolerance requirements shall be in accordance with Chapter 112 and as stated in the Rotorcraft Specification.

3.3.13 Safety of Lubricated Parts

(i) Each Lubricated assembly, provided with an oil reservoir, shall have a suitable level indicator or contents gauging means that shall be easily visible and readable over the life of the Rotorcraft system.

(ii) Following a loss of lubricant from an assembly, the assembly shall continue to operate safely for a period to be defined in the Rotorcraft specification, and as demonstrated by rig test. In determining the acceptable period for which safe operation is to be demonstrated the method of detection of the lubricant loss and the consequences of a failure shall be taken into account to ensure that a catastrophe is highly improbable. Failure Mode Effect Analysis (FMEA) and safety assessments shall be carried out. Health monitoring may be used to detect the loss of lubricant or the onset of failure such as heat build-up in the bearings, see para 3.3.14.

3.3.14 System Monitoring

(a) Health and Usage Monitoring

Consideration shall be given to the application of Health and Usage Monitoring Systems (HUMS) to support the achievement of the design requirements in respect of components especially as follows:

(i) For which reliance on damage tolerant characteristics (i.e. dual load paths, crack growth rate, progressive debris release, ballistic tolerance etc.) is impractical.

(ii) For which a safe life philosophy has been decided, in order to ensure that the usage of the component agrees with the assumed load/duty cycle assumption.

In determining the HUM functions to be provided consideration shall be given to Chapter 727.

(b) Accident Data
The parameters required to establish the operating status or condition of the Rotor System for Accident Data Recording purposes shall be determined by noting the requirements of Chapter 727 and the mandatory requirements relating to civil Rotorcraft. Knowledge gained from Rotor System related occurrences and accidents to both military and civil Rotorcraft shall be implemented.

3.3.15 Assessment of Safety Devices
Where reliance is placed on safety devices, instrumentation, early warning devices, maintenance checks etc., to limit the effects of a failure, the safety analysis shall cover the safety system failure in combination with the basic Rotor System failure.

3.3.16 Provision of Instruments
Due allowance shall be made for the accuracy and potential reading errors of cockpit indicators necessary for safe operation of the Rotorcraft. The overall limits of accuracy required of indications for enabling flight crew to control the operation of the Rotor Systems shall also be stated so that the suitability of the instruments as installed may be assessed. Regard shall be given to possible inaccuracies in reading the instruments and the need to delineate normal and abnormal values.

3.4 SYSTEM OPERATION LIMITS
3.4.1 Normal Service Use
Rotor System operating limitations shall be determined by having regard to the values demonstrated during the type tests, due allowance having been made for the overall accuracy limits of instrumentation declared for use in service and shall conform to the following:

(a) Rotor Speed
Operational maximum and minimum RPM limits for power-off and power-on steady and transient conditions shall be established for the design of the Rotorcraft and specifically the rotor components. These speeds shall be displayed to the pilot and design RPM values shall be related to these operational speed limits. The operational RPM limits shall be chosen carefully on the basis of the anticipated handling behaviour such that the limits will not be exceeded during practical flight operations. For power-off conditions rotor speed behaviour must be considered for manoeuvres, gusts and especially

during engine off landings. For power-on cases, manoeuvres, gusts, rapid power changes, post engine failure manoeuvres and ease and accuracy of setting datum RPM shall be considered.

(i) The maximum operational transient power-off RPM limit shall be no greater than 95% of the lesser of the maximum design RPM or the maximum power-off demonstrated during type tests.

(ii) The maximum steady power-off RPM limit shall be set high enough that autorotation can be achieved without retarding or redatuming the engines of the aircraft. It shall be set high enough that normal manoeuvres (excluding avoidance manoeuvres or final flare prior to engine off landing) can be carried out without exceeding the limit.

(iii) Maximum steady power-on RPM shall be no greater than the lesser of the maximum mean value demonstrated in the type test or the design maximum power-on RPM.

(iv) Minimum steady power-on RPM shall be no lower than the greater of the minimum mean value demonstrated during the type test or the design minimum power-on RPM.

(v) The steady power-on RPM range shall be wide enough to accommodate all normal power-on manoeuvres without infringing the power-on transient range. The transient range is to deal with emergency or exceptional circumstances.

(vi) Load cases for the design of the rotor shall take into account the design maximum and minimum power-on Rpm values and the maximum power-off RPM. In addition the maximum and minimum transient power-on rotor speeds and the minimum transient rotor speed power-off declared for operation shall be considered for specific emergency cases.

(viii) For tilt-rotor aircraft, allowable rotor speed ranges may be required to the specified separately for hover/conversion and cruise flight regimes.

(b) Torque
Each declared torque limitation shall be based on the mean values recorded during the appropriate tests.

4 COMPONENT DESIGN GENERAL REQUIREMENTS

4.1 COMPONENT DESIGN CRITERIA AND CONSIDERATIONS

4.1.1 Rotor System components shall be designed in accordance with the strength, deformation and detail design requirements of Chapters 200 and 400.

4.1.2 Limit Design Cases

(a) The design cases for the Rotor System shall include all practical combinations of rotor thrust, moment, torque, blade flap and lag movement, control loads and rotor speeds within the Design Specification flight envelope of the aircraft. These loads must be compatible with the cases taken to design the non rotating components of the airframe but are not necessarily the same conditions. An independent analysis to determine the critical condition for rotor components shall be performed. For proof and ultimate strength factors see Chapter 200 para 3.1 (see also Chapter 200 para 1.9).

(b) Symmetrical Gust Loads

(i) The Rotorcraft is assumed to be flying in a trimmed unaccelerated flight condition corresponding to any point on or within the symmetric flight envelope.

(ii) In this condition the Rotorcraft encounters a gust velocity 35 fps from any direction, upwards, downwards, sideways or intermediate.

(iii) This gust intensity shall be considered as sharp-edged, i.e. the air velocity is suddenly and uniformly increased over the rotor in the direction of the gust. (The assumption is thus made of unit alleviating factor for this gust velocity).

(iv) Both the power-on and power-off states shall be considered.

(c) Gust Gradient Distance

(i) For individual blade stressing it shall be assumed that a vertical gust intensity of 50 fps increases linearly from zero to the maximum value while the Rotorcraft travels a distance of 100 feet.

(d) Gust Requirement for Instrument Flight
Any increase in gust velocity for instrument flight shall be agreed with the Rotorcraft Project Director. The requirements of paragraph 4.1.2b and c above are only intended to cover flight in clear air.

(ii) For design purposes the worst combination of rotor speed and forward speed shall be considered, in conjunction with this gust gradient.

4.1.3 Fatigue

All components shall have a fatigue life at least equal to that stated in the Rotorcraft Specification. The usage spectrum to be used for substantiation (i.e. aircraft weight, altitude, airspeed, normal acceleration, rotor speed, manoeuvre type and time allocated for each condition) shall be stated in the Rotorcraft Specification. A once per flight load shall be part of the fatigue spectrum and picketing, ground handling, engine start and rotor brake application shall be included as necessary.

4.1.4 Stress Concentration

Care shall be taken in component detail design to avoid or reduce stress concentrations, particularly in areas subjected to vibratory loading.

4.1.5 Retention of Fasteners

For details concerning locking of threaded fasteners see Chapter 400 para 7. Quick release fasteners such as those used for rotor blade to head fastening (see para 2.1.1(ii)) shall conform with the requirements of Chapter 400 para 7.9.

4.1.6 Tolerances and Static-Dynamic Clearances

Attention shall be paid to the combinations of component maximum and minimum tolerances including the effects of wear tolerances to ensure adequate operating clearances are maintained within the System and include allowance for temperature and component defection under load. The Rotorcraft Specification shall state clearances to be maintained for blade pitch control mechanisms. The full tightening of threaded fasteners without thread binding shall be obtained by making allowance for the combination of component adverse tolerances.

4.1.7 ‘Thin Wall’ Requirements

Component wall thickness shall be controlled by design and inspection to prevent wall thicknesses below drawing tolerances remaining undetected. DEF STAN 05-61 (Part 12) gives details of the procedures to be followed.

4.1.8 Identification of Finite Life Components

Any critical component shall be identified with a serial number. In addition the Rotorcraft designer shall draw up a recommended list of unlimited life components of sufficient value and importance to be given individual serial numbers. This list shall be agreed by the Rotorcraft Project Director.

4.1.9 Control of Grade A -Vital Parts

See Chapter 400 para 2.2.2.

4.1.10 Control of Mass Distribution

Consideration shall be given to the control of mass, moments and dynamic balance within suitable limits, to permit direct replacement. Drawings shall specify these limits and indicate how adjustments are to be made.

4.1.11 Component Handling
When required, slinging facilities shall be provided in accordance with Chapter 801, paras 4.3.1, 4.3.2 and 4.4.1.

5 ROTOR HEAD ASSEMBLIES

5.1 DESIGN CONSIDERATION

5.1.1 Rotor head configurations are many and various, ranging from fully articulated to fully rigid. In all cases the Rotorcraft Design Authority shall pay particular attention to the following:

(i) Reduction of Parts
Reducing to a minimum the number of parts, the failure of which would be catastrophic.

(ii) Damage Tolerant Design
Ensuring that the design maximises tolerance to likely defects or damage. The rate of growth of such defects or damage should be sufficiently slow such that it will allow discovery during maintenance inspection. Alternative load paths should be provided whenever possible.

(iii) Fatigue Warning Devices
Where a part is not damage tolerant then it should be possible to detect impending failure by health or usage monitoring.

(iv) In the event that a part cannot satisfy paras 5.1.1(ii) or (iii) it shall be designed to safe life principles.

(v) Fretting
Means shall be taken to reduce to a minimum the fretting that can occur between mating surfaces.

(vi) Temperature Effects
Where joints of differing materials (e.g. composite/metallic) require specified fits, the effects of extreme temperatures and differential thermal expansion shall be taken into consideration.

(vii) Environmental Conditions
Environmental conditions shall be taken into consideration when selecting materials (e.g. corrosion caused by damaged protective coatings, moisture ingress or moisture absorption causing differential expansion between differing materials).

5.2 HINGED ASSEMBLIES

5.2.1 Mounting of Elastomeric Bearings

(i) Hinged assemblies employing elastomeric bearings shall be mounted so that the elastomer is not subjected to harmful tensile loads.

(ii) Where elastomeric bearing are "lifed on condition" the working surface of the bearing, i.e. the edge of the rubber and metal sandwich, should be easily inspectable.

5.2.2 'Indexing' of Elastomeric Bearings

Where applicable elastomeric bearings shall be 'indexed' to a minimum loading position when the blades are stationary.

5.2.3 Hinge 'Stiffness'

Consideration shall be given to the effect of bearing wear on hinge stiffness due to its influence on the dynamics of the system (see also para 3.1.9).

5.3 ROTOR ARTICULATION LIMIT STOPS

5.3.1 Damage Prevention

Where necessary to prevent damage, including blade contact with the airframe or ground, means shall be provided to limit blade motion.

Note: Gust conditions, deck motion or ground handling shall also be considered when complying with this requirement. See also para 3.1.4.

5.3.2 Stop Clearance

If the means of achieving the requirements of para 5.3.1 is the use of droop stops they shall be designed such that they will not be contacted in flight. The safe operation of the stops during starting and stopping the rotor shall not require special pilot attention, either to ensure engagement or to prevent high loads in the stops. Attention shall be paid to the operation of stops in icing conditions (see Leaflet 732/1).

5.3.3 Droop Stop Failure

Where centrifugally operated droop stops are employed the design of the rotor head shall be such that following failure of one or more of the droop stops in zero wind and with the Rotorcraft standing on a level surface the rotor blades will not contact any other part of the Rotorcraft.

5.4 LEAD/LAG DAMPERS

5.4.1 General Design

The reliability of each damper shall be such that the possibility of an instability occurring is highly improbable. This will depend upon the tolerance of the Rotor System to loss or partial loss of a damper or dampers.

5.4.2 Dampers

The design and maintenance of dampers should ensure that the requirements of para 5.4.1 are met. If this is not possible and reliable cues do not result for an impending damper failure this shall be detected by health monitoring. In the calculation of reliability, the reliability of the health monitoring devices shall also be taken into account.

5.5 BLADE PITCH INDICATOR

5.5.1 Pitch Range

Means shall be provided to check the control movement setting of the Rotor System when the Rotorcraft is on the ground, either with fixed equipment or the use of ground equipment. It is desirable that this should be independent of aircraft attitude.

5.6 AUTOMATIC BLADE FOLDING

5.6.1 Operation Requirements

Blade folding requirements and operational limits for blade folding shall be stated in the Rotorcraft Specification.

5.6.2 Power Malfunction

The system design shall permit manual folding of the blades in the event of a power malfunction.

5.6.3 Pitch Locks

Where pitch locks are employed as part of the folding sequence, provision shall be made to ensure that the rotors cannot be driven with the pitch locks in place. The whole folding system shall be disabled once the rotor blades are spread.

5.7 MANUAL BLADE FOLDING

5.7.1 Blade Handling

The procedure used to hold the rotor blades during manual folding and picketing operations shall be such as to minimise damage or that any damage likely to be caused during folding and picketing by excessive loads, impact of folding equipment etc. will not endanger the Rotorcraft.

5.7.2 Pitch Locks

Where pitch locks, control locks or any ground equipment is employed as part of the folding and picketing procedure, provision shall be made to ensure that the rotors cannot be driven with the locks or ground equipment in place if to do so would endanger the Rotorcraft.

5.8 ROTOR HEAD BALANCING

5.8.1 General

(i) Means shall be provided to enable the balance of the rotor head to be adjusted to within specified limits.

(ii) Means should be provided to permit the on aircraft dynamic balance of main and tail rotors.

(iii) The possibility of masking any developing faults by repetitive balance corrections shall be considered. This is in addition to the balancing of individual blades.

5.9 SLINGING

5.9.1 Where it is required to sling a Rotorcraft via the rotor hub the requirements of Chapter 801, para 4 and associated sub-paras shall apply. Such requirements shall be defined in the Rotorcraft Specification and shall include any requirements to carry the Rotorcraft as underslung beneath another aircraft if appropriate.

5.10.1 The requirements for blade signature, in the microwave visible and infra-red frequencies shall be stated in the Rotorcraft Specification. Means of achieving agreed signature requirements shall be compatible with protection of the blade from erosion, corrosion, humidity and ultra-violet radiation unless stated otherwise in the Rotorcraft Specification.

6 ROTOR BLADES

6.1 DESIGN CONSIDERATIONS

6.1.1 Static and Dynamic Balance

(i) Rotor blades shall be designed to contain means for static and dynamic balancing.

(ii) When provision for adjusting the blade balance is provided it shall not be possible to add weights incorrectly within the physical adjustment range and cause catastrophic instabilities in the blade or rotor, either on the ground or in flight, if operated in such an unbalanced condition (see also Chapter 500 para 3.4).

6.1.2 Measurement of Blade Track

(i) Consideration should be given to means for adjustment of blade track, including the method to be adopted for correcting the individual blade lift differences whilst maintaining the design rotor pitch range.

(ii) Means for in-flight measurement of blade track shall be provided, either remotely or by in-built means such as reflective flags or transducers.

(iii) If means are provided for adjusting the aerodynamic balance of the blade by means of tabs or wedges or some other device:

(a) if the operator is allowed to adjust the setting, the design of the means of adjustment must consider its use in the field such that adjustment is straight forward for the operator.

(b) the trimming device must be capable of fine and progressive adjustment.

(c) the settings of the trimming devices must not be disturbed by flight loads including the loads that arise during blade stall conditions in manoeuvres.

Note: trailing edge tab devices have been known to change angle during blade stall encounters, requiring subsequent tracking activity.

6.1.3 Service Life
The minimum safe fatigue life of a rotor blade shall be stated in the Rotorcraft Specification. The resistance or otherwise of the blade to environmental effects such as humidity, salt spray and sunlight shall be as defined in the Rotorcraft Specification.

6.1.4 Tree and Wire Strike
The severity of a tree or wire strike to be resisted by the rotor blades and the means of compliance, shall be stated in the Rotorcraft Specification.

6.1.5 Repairability
When designing rotor blades, repair procedures shall be considered that combine ease of repair and the maintaining of aerodynamic and dynamic balance characteristics. Due to the nature of their usual construction, blade trailing edges are most susceptible to damage; thus particular attention should be paid to their easy repair. The life of rotor blades following any standard repairs shall be determined by testing and quoted in the repair manual.

6.1.6 Deformation
The blades shall be so designed that they will maintain their aerodynamic contour and balance characteristics throughout the permissible flight envelope. Consideration shall be given to the effects of variation of ambient conditions, e.g. pressure, temperature and humidity.

6.1.7 Venting and Water Accumulation

(i) Venting
There shall be means to vent the internal pressure of a blade. This requirement shall not apply to sealed rotor blades capable of withstanding the maximum pressure differentials expected in service.

(ii) Water Accumulation
Rotor blades shall be designed to prevent water accumulation in any part of the blade.

6.1.8 Effect of Engine Exhaust Gases Impingement
Consideration shall be given to the effects of possible engine and auxiliary power unit exhaust gases impingement on the blades during ground operation (folded and spread) and in flight. Testing shall be carried out to establish that the blade materials and construction are not adversely affected by the temperatures achieved on the blade under operating conditions. This requirement shall also apply during blade folding and parking operations.

6.1.9 Effect of Thermal Ice Protection System
Consideration shall be given to the effects of thermal ice protection systems on rotor blades. Testing shall be carried out to establish that blade materials and construction are not adversely affected by the temperatures achieved on the blade under operating conditions (see also Chapter 711).

6.1.10 Health and Usage Monitoring
See para 3.3.14.

6.1.11 Blade Handling
If there is a requirement for handling blades during blade folding operations handling positions shall be clearly marked. For handling and transporting blades once removed from the Rotorcraft, position shall be provided that are clearly defined and appropriately marked. All markings shall be in accordance with operational camouflage requirements.

7 TESTING

7.1 PROTOTYPE AND RIG TESTS

7.1.1 The achievement of acceptable Rotor System design and development shall be shown by a programme of tests. These tests are of vital importance and a detailed schedule of the proposed tests shall be stated in the Rotorcraft Specification. Consideration shall be given to the development history of the Rotor System when reviewing the results of these tests.

7.2 PRODUCTION TESTS

7.2.1 A programme of tests shall be carried out to reveal, as far as is practicable, any fault of manufacture or assembly. A detailed schedule of the tests proposed for each component or assembly and components requiring a production quality test to be periodically carried out when such components have already passed their production tests, to check that manufacturing standards are maintained, shall be stated in the Rotorcraft Specification. The Rotorcraft Design Authority shall state the requirements for test schedules, frequency and conditions for such tests.

7.3 MONITORING PROVISIONS

7.3.1 Consideration shall be given on the critical rotor system components for the use of a health and usage monitoring system.

7.3.2 It shall be demonstrated by tests that those fault detection devices intended to give early warning of an impending failure which could lead to catastrophe, give adequate warning of the fault before it has progressed to a dangerous extent.

8 DEMONSTRATION OF COMPLIANCE

8.1 Compliance with the requirements of this Chapter shall be demonstrated by either Design Records (DR), Mock Up (MU), Analysis (A), Component Test (CT), Ground Test Vehicle (GTV), Flight Test (FT), Production Quality Tets (PQT) or a combination of these means in accordance with Table 1*, unless otherwise stated in the Rotorcraft Specification.

*Alternatives are indicated by /.

Note: Where test data is required it will normally relate to production standard components of the Project Rotorcraft. However, some requirements may be satisfied by data obtained from tests with other components if indicated in the Rotorcraft Specification.

9 CROSS REFERENCE TO OTHER RELEVANT CHAPTERS

9.1 A number of requirements related to Rotor Systems appear elsewhere in this publication. These are listed in Table 2.

TABLE 1

DEMONSTRATION OF COMPLIANCE REQUIREMENTS (see par 8.1)

CHAPTER PARAGRAPH	METHOD OF COMPLIANCE DEMONSTRATION (See par 8.1 for explanation of codes)
1.4	DR and A and CT/GTV and FT
2.1.1	A and DR
2.1.1 (i)	DR
2.1.1 (ii)	MU and DR
2.1.1 (iii)	A and CT and DR
2.1.1 (iv)	A and CT and DR
2.1.1 (v)	DR and GTV/FT
2.1.2	MU and GTV/FT
3.1.1	DR and A
3.1.4 (i)	A supported by FT
3.1.4 (ii)	A supported by GTV/FT
3.1.4 (iii)	A supported by GTV/FT
3.1.5	DR, GTV and MU/FT
3.1.6	DR
3.1.7	DR, A and FT
3.1.8	A/DR supported by GTV/FT

CHAPTER PARAGRAPH	METHOD OF COMPLIANCE DEMONSTRATION (See par 8.1 for explanation of codes)
3.1.9	DR/A
3.1.10	A/DR supported by GTV and FT
3.1.11	CT and GTV/FT
3.1.12	CT and GTV/FT
3.1.13	A and FT
3.2.1	A supported by FT
3.2.2	A
3.2.3	A, CT, GTV and FT
3.3.1	A and DR
3.3.2	A and DR
3.3.3	DR and GTV/FT
3.3.4	A and CT
3.3.5	A
3.3.6	A and DR supported by GTV/FT
3.3.7	DR and FT
3.3.8	DR and GTV and FT
3.3.9	DR/CT/GTV/FT
3.3.10	DR/CT/GTV/FT
3.3.11	DR/A/CT
3.3.12	A and CT
3.3.13	A and CT
3.3.14	A and CT
3.3.15	DR and A
3.3.16	DR
3.4.1	DR supported by CT/GTV/FT
4.1.2	DR and A and CT and FT
4.1.3	DR and A and CT and FT
4.1.4	DR and A
4.1.6	DR and A
4.1.7	DR and A
4.1.8	DR and A
4.1.9	DR
4.1.10	DR
4.1.11	DR
5.1.1 (i)	DR and A
5.1.1 (ii)	DR and A/CT
5.1.1 (iii)	DR
5.1.1 (iv)	DR and A
5.1.1 (v)	DR and CT and GTV
5.1.1 (vi)	A and CT
5.1.1 (vii)	DR/A and CT
5.2.1	DR
5.2.2	DR

CHAPTER PARAGRAPH	METHOD OF COMPLIANCE DEMONSTRATION (See par 8.1 for explanation of codes)
5.2.3	A and CT/GTV
5.3.1	DR and GTV/FT
5.3.2	DR and GTV/FT
5.3.3	DR/A supported by GTV/FT
5.4.1	DR/A supported by MU/GTV
5.4.2	DR and PQT and GTV/FT
5.5.1	DR
5.6.1	DR and A
5.6.2	DR and GTV
5.6.3	DR supported by GTV/FT
5.7.1	A supported by GTV/FT
5.7.2	DR supported by GTV/FT
5.8.1	DR
5.9.1	DR
5.10.1	DR,CT and FT
6.1.1	DR/A
6.1.2	CT/GTV/FT
6.1.3	A supported by CT/GTV/FT
6.1.4	A supported by CT
6.1.5	A and CT
6.1.6	A/CT
6.1.8	DR/A supported by FT
6.1.9	DR/A DR supported by GTV and FT
6.1.10	CT
6.1.11	DR

TABLE 2

LIST OF OTHER IMPORTANT REQUIREMENTS (see para 9.1)

CHAPTER	PARAGRAPH	SUBJECT
100	7,17	General Requirements
101	-	Operation in Various Climatic Conditions
107	Table 4 Item 2	Pilots Cockpit - Controls and Instruments
108	-	Internal Noise
112	-	Reduction of Vulnerability to Battle Damage
200	-	Static Strength and Deformation
203	-	Control Systems - Mechanical Components
206	-	Design to Resist Birdstrike Damage
400	-	General Detail Design
407	-	Precautions against Corrosion and Deterioration
500	-	Aero-Elasticity
501	3	Vibration and Internal Noise
704	-	Hydraulic Systems
705	3.1.6	Transmission Systems
706	-	Electrical Installations
711	-	Ice Protection
722	-	Folding Components
727	-	Health and Usage Monitoring Systems
800	-	General Maintenance Requirements
801	-	Transport, Handling and Storage
802	-	Routine Servicing
803	-	Repairs
804	-	Replacement of Components
805	-	Interchangeability
900	7.2,7.5,7.6,7.7	General Handling Flight Test Requirements

LEAFLET 732/0

ROTOR SYSTEM

REFERENCE PAGE

Civil Aviation Authority

British Civil Airworthiness Requirements (BCAR)	.. ..	Section G Rotorcraft

Federal Aviation Regulations (FAR)

Part 29 - Airworthiness Standards	..	Transport Category Rotorcraft

US Military Specification

MIL-STD-1294A	.. ..	Acoustical Noise Limits in Helicopters

Defence Standards

00-29	.. ..	Fungal Contamination Affecting the Design of Military Material
00-50	.. ..	Guide to Chemical Environmental Contaminants and Corrosion Affecting the Design of Military Material
05-61 (Part 12)	.. ..	Control of Wall Thickness in Design and Manufacture of Flying Control Systems
07-55 (Part 2) Section 4	.. ..	Environmental Testing of Service Material

RAE Technical Memos

FS(F) 632	.. ..	Lightning Protection Requirements for Aircraft

British Standards

3G.100:Part 2	.. ..	General Requirements for Equipment in Aircraft (Gradually being superseded by DEF STAN 00-970 and DEF STAN 00-35)

LEAFLET 732/1

ROTOR ARTICULATION LIMITS

1 INTRODUCTION

1.1 This Leaflet contains recommendations on rotor head design with respect to blade angular movement and limit stops.

2 DRAG STOPS

2.1 Drag stops should be incorporated in every drag hinge assembly. The freedom of drag movement should be determined by the extreme positions in operation.

2.2 Failure of a drag stop should not lead to consequential failure of other parts of the Rotor System which would cause a hazard when the Rotorcraft is operated within its normal operational envelope.

2.3. Where stops are centrifugally controlled to provide more clearance in flight than when the rotor is stationary they should be designed such that excessive pilot effort is not required to achieve engagement and disengagement, even after degradation in service. Consideration should be given to the consequences of the stops failing to engage or disengage.

2.4 The design of centrifugally operated drag stops should consider the effects of the Rotorcraft starting when the rotor is below 0°C and the drag stops failing to disengage. This consideration should also apply during rotor shut down following ice accretion in flight to establish the consequences of the drag stops failing to engage.

2.5 In the majority of articulated tail rotors there are no drag hinges therefore drag stops are unnecessary.

3 FLAPPING STOPS

3.1 MAIN ROTOR

3.1.1 Where flapping stops are provided on a main rotor head they must allow for the full range of collective control plus some portion of the cyclic control to be applied simultaneously, throughout the allowable rotor RPM range, without the blade stops being impacted. It must also be possible to perform usual helicopter manoeuvres in conditions of moderate turbulence without impacting the flapping stops. This is to ensure that undue pilot attention is not required to prevent damage to the stops during normal operation. Flight testing shall demonstrate the clearance margins existing for flight within the flight envelope and attention should be given to the security of droop stops throughout the service life of the Rotorcraft.

It shall be demonstrated by analysis supported by flight test that in extreme or limit manoeuvres blade stop contact is unlikely to occur. In cases where enough clearance cannot be provided to deal with extreme or unusual manoeuvres or gusts and stop impact is likely the resulting loads must be quantified and substantiated. They shall be shown to be repeatable and be included in the fatigue spectrum for the affected components. The affect of stop impact on the assumed fatigue strengths must also be addressed.

3.1.2 Where stops are centrifugally controlled to provide more clearance in flight than when the rotor is stationary, they shall be designed such that excessive pilot effort is not required to achieve engagement, even after degradation in service. Consideration should be given to the consequences of the stops failing to engage or disengage.

3.1.3 The static and fatigue strength of flap limiting stops and the rotor system in general shall consider the loads arising from the stops being impacted at extremes of the possible pilot inputs at the extremes of the aircraft operating envelope.

3.1.4 The design of centrifugally operated droop stops should consider the effects of the Rotorcraft starting when the rotor is below 0°C and the droop stops failing to disengage. This consideration should also apply during rotor shut down following ice accretion in flight to establish the consequences of the droop stops failing to engage.

3.2 TAIL ROTORS

3.2.1 Although it is not desirable to strike the flapping stops during flight, it is permissible to allow this to occur if the effect can be accurately quantified and used in the evaluation of the design loads.

Note: Because the pilot is unlikely to receive any direct cue that tail rotor flap stop contact is occurring, careful attention to quantifying such loads and the condition under which they occur must be paid so that the loads may be conservatively included in the fatigue spectrum. Careful attention must be paid to the occurrence of deliberate sideslip manoeuvres in which tail rotor stop contact is most likely to occur.

4 GUST LOCKS

4.1 When a helicopter is stationary for long periods it is prudent to prevent damage to the rotor by the use of various ‘gust locks’ and other picketing devices. Ground equipment should also be provided to prevent tail rotors flapping in the wind and impacting heavily onto the flapping stops if this would cause damage to the rotor, unless the rotor has sufficient inherent stiffness in the flapping plane.

PART 7 APPENDIX No.2
INSTALLATIONS
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 700: PROPULSION SYSTEM INSTALLATIONS

700	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-HDBK-336-1	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, GENERAL CRITERIA, VOL.1
	MIL-HDBK-336-3	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, ENGINE, VOL. 111
	MIL-STD-1290	LIGHT FIXED AND ROTARY WING AIRCRAFT CRASHWORTHINESS
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION FOR AEROSPACE SYSTEMS
	MIL-T-5091	TRANSMISSION, POWER, CONSTANT RATIO: GENERAL SPECIFICATION (AIRCRAFT USE)
	MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-F-8615	FUEL SYSTEM COMPONENTS: GENERAL SPECIFICATION FOR
	MIL-C-8678	COOLING REQUIREMENTS OF POWER PLANT INSTALLATIONS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-P-8686	POWER UNITS; AIRCRAFT AUXILIARY, GAS TURBINE TYPE, GENERAL SPECIFICATION FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-H-8775	HYDRAULIC SYSTEM COMPONENTS, AIRCRAFT AND MISSILES, GENERAL SPECIFICATION FOR
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT (GENERAL SPECIFICATION FOR)
	MIL-I-18802	FUEL AND OIL LINES, AIRCRAFT, INSTALLATION OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT
	MIL-P-85573	POWER UNIT, AIRCRAFT, AUXILIARY, GAS TURBINE, GENERAL SPECIFICATION FOR

CONTROLLED DISTRIBUTION:

	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS.VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

700 1.	MIL-STD-2069	PARA: 5.2.3
	MIL-E-8593A	PARA: 1.1, 3.1.2
	MIL-S-8686	PARA: 3
	MIL-F-83300	PARA: 3.1.10.2.1
	MIL-P-85573	PARA: 3

2. GENERAL

700 2.	MIL-STD-250D	PARA: 5.2.2
	MIL-STD-1290	PARA: 5.1.7
	MIL-STD-1472C	PARA: 4.5, 4.7, 4.9, 5.1

700 2.	MIL-B-5087B	PARA: 3.3.4
	MIL-E-8593A	PARA: 3.1.2.6, 3.1.2.10, 3.3.1.1.4, 3.3.8.10, 3.5, 3.7
	MIL-S-8698	PARA: 3.6.4, 3.6.5.2.4.3
	MIL-F-18372	PARA: 3.1.1.2
	MIL-D-23222A	PARA: 3.3.5.2.5.1, 3.10, 3.14.2

3. ENVIRONMENTAL

700 3.	MIL-E-8593A	PARA: 3.2.5
	MIL-D-23222A	PARA: 3.10

4. PROPULSION SYSTEM FEATURES

700 4.	MIL-T-5091	PARA: 4.1.1
	MIL-E-8593A	PARA: 3.2.1.4.11, 3.2.1.10, 3.3.8.9, 3.7.10, 3.7.11, 3.7.12, 3.7.14
	MIL-S-8698	PARA: 3.6.6

5. RELATED SYSTEM FEATURES

700 5.	MIL-E-8593A	PARA: 3.3.6.5, 3.7.2.3.2

6. OPERATIONAL

700 6.	MIL-E-8593A	PARA: 3.2.5.8.1, 3.3.8, 3.7.10.3
	MIL-D-23222A	PARA: 3.10.7.4

7. MAINTENANCE AND SERVICING

700 7.	MIL-STD-1472C	PARA: 5.9
	MIL-I-8500D	PARA: 3.3.1.(b)
	MIL-E-8593A	PARA: 3.2.1.3, 3.2.4, 3.5.1, 3.5.1.1, 3.7.3.4, 3.7.7.4.3
	MIL-F-8615D	PARA: 3
	MIL-C-8678	PARA: 3
	MIL-T-8679	PARA: 3.2.9.1.2
	MIL-H-8775D	PARA: 3
	MIL-F-17874	PARA: 3.3.10.3
	MIL-F-18372	PARA: 3.1.1.2.3
	MIL-I-18802A	PARA: 3.5
	MIL-D-23222A	PARA: 3.17

8. REDUCTION OF VULNERABILITY TO BATTLE DAMAGE

700 8.	MIL-HDBK-336-1	
	MIL-HDBK-336-3	
	MIL-STD-2069	PARA: 4, 5
	MIL-E-8593A	PARA: 3.3.6.7

9. CROSS REFERENCE TO OTHER CHAPTERS

700 9.

CHAPTER 701: REFUELLING AND DEFUELLING SYSTEMS

701	MIL-HDBK-221	FIRE PROTECTION DESIGN HANDBOOK FOR U.S. NAVY AIRCRAFT POWERED BY TURBINE ENGINES
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
	MIL-F-8615	FUEL SYSTEM COMPONENTS: GENERAL SPECIFICATION FOR
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-A-19736	AIR REFUELLING SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-C-38373	CAP, FLUID TANK FILLER
	MIL-C-83413	CONNECTORS AND ASSEMBLIES, ELECTRICAL, AIRCRAFT GROUNDING DISCHARGERS, AND RECEPTACLES, ETC

CONTROLLED DISTRIBUTION:

	MIL-F-87154	FUEL SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS.VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

701 1.	MIL-F-17874B	PARA: 3.3.9
	MIL-D-23222A	PARA: 3.10.13

2. FUELLING/DEFUELLING METHODS

701 2.	MIL-F-17874B	PARA: 3.3.9.2

3. OPEN ORIFICE FUELLING

701 3.	MIL-B-5087B	PARA: 3.3.4
	MIL-F-8615D	PARA: 3.6.4
	MIL-F-17874B	PARA: 3.3.9.2, 3.3.9.3.1, 3.3.9.4
	MIL-C-38373B	PARA: 3.5

4. PRESSURE REFUELLING AND DEFUELLING

701 4.	MIL-F-8615D	PARA: 3.6.3
	MIL-F-17874B	PARA: 3.3.9

5. FIRE PRECAUTIONS

701 5.	MIL-HDBK-221	PARA: 2.5
	MIL-C-83413	

REFERENCES

CHAPTER 702: FUEL SYSTEMS

702	MIL-STD-210	CLIMATIC INFORMATION TO DETERMINE DESIGN AND TEST FOR MILITARY SYSTEMS AND EQUIPMENT
	MIL-HDBK-336-1	SURVIVABILITY, AIRCRAFT, NONNUCLEAR GENERAL CRITERIA, VOL. 1
	MIL-HDBK-336-2	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, AIRFRAME,. VOL. 11
	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-STD-1290	LIGHT FIXED AND ROTARY WING AIRCRAFT CRASH RESISTANCE
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
	MIL-W-5088	WIRING, AEROSPACE VEHICLES
	MIL-P-5238	PUMP, CENTRIFUGAL, FUEL BOOSTER, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-T-5578	TANK, FUEL, AIRCRAFT SELF SEALING
	MIL-T-6396	TANKS, AIRCRAFT PROPULSION FLUID SYSTEM, INTERNAL, REMOVABLE, NON-SELF-SEALING
	MIL-I-7566	INDICATOR SYSTEM
	MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
	MIL-F-8615	FUEL SYSTEM COMPONENTS: GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-T-18847	TANKS, FUEL, AIRCRAFT, AUXILIARY EXTERNAL, DESIGN AND INSTALLATION OF
	MIL-A-19736	AIR REFUELLING SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-V-25023	VALVE, FUEL DRAIN, SELF LOCKING
	MIL-T-27422	TANK, FUEL, CRASH-RESISTANT, AIRCRAFT
	MIL-C-38373	CAP, FLUID TANK FILLER

CONTROLLED DISTRIBUTION:

	MIL-F-87154	FUEL SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS.VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

702 1.	MIL-F-17874B	PARA: 1.1, 3.3.1

2. GENERAL

702 2.	MIL-HDBK-336-1	
	MIL-HDBK-336-3	
	MIL-STD-1290	PARA: 3.32
	MIL-STD-2069	PARA: 5.3.3.4
	MIL-3-5087B	PARA: 3.3.4
	MIL-F-8615D	PARA: 3.6.8, 3.7.6
	MIL-A-19736	

702 2. MIL-F-17874B PARA: 3.1, 3.2, 3.3.4, 3.3.6, 3.3.9.4, 3.5, 4.17
MIL-C-38373B PARA: 3.5, 3.6

3. FUEL SYSTEM ANALYSIS AND TEST
702 3. MIL-STD-210
MIL-STD-810
MIL-T-6396
MIL-F-8615
MIL-T-8679
MIL-F-17874
MIL-F-18847
MIL-A-19736
MIL-D-23222

4. FUEL SYSTEM LIGHTNING PROTECTION
702 4. MIL-B-5087
MIL-W-5088

5. FUEL FLOW
702 5. MIL-P-5238
MIL-F-17874

6. FLOW BETWEEN INTERCONNECTED TANKS
702 6. MIL-F-17874

7. UNUSABLE FUEL
702 7. MIL-F-17874

8. FUEL SYSTEM HOT WEATHER
702 8. MIL-D-2322

9. CRASHWORTHINESS
702 9. MIL--STD-1290
MIL-T-27422

10. REDUCTION IN VULNERABILITY TO BATTLE DAMAGE
702 10. MIL-HDBK-336-1
MIL-HDBK-336-2
MIL-STD-2069

11. FUEL TANKS
702 11. MIL-T-5578
MIL-T-6396

702 11. MIL-T-18847
MILT-27422

12. FUEL TANK INSTALLATION
702 12. MIL-T-5578
MIL-T-6396
MIL-I-8500
MIL-T-18847
MIL-T-27422

13. FUEL TANK STRENGTH REQUIREMENTS
702 13. MIL-T-6396
MIL-S-8698

14. ENGINE REFRIGERANT INJECTION SYSTEM
702 14. MIL-T-6396

15. FUEL TANK EXPANSION SPACE
702 15. MIL-T-6395
MIL-T-18847

16. FUEL TANK TEMPERATURE
702 16. MIL-STD-210
MIL-F-17874
MIL-D-23222

17. FUEL TANK SUMP AND FUEL SYSTEM DRAINS
702 17. MIL-V-25023

18, FUEL TANK VENTS
702 18. MIL-F-17874

19. FUEL TANK OUTLET
702 19. MIL-F-17874

20. FUEL TANK FILLER CONNECTION
702 20. MIL-F-8615
MIL-F-17874
MIL-C-38373

21. PREVENTION OF INCORRECT ASSEMBLY

702 21. MIL-D-1472
MIL-I-8500
MIL-F-8615

22. NON-RETURN VALVES

702 22. MIL-F-8615

23. FUEL SYSTEM LINES AND FITTINGS

702 23. MIL-F-17874
MIL-A-19736

24. FUEL SYSTEM L.P. AND CROSS FEEL VALVES

702 24. MIL-F-8615

25. FUEL FILTER CONTAMINATION

702 25. MIL-F-8615
MIL-F-17874

26. FUEL PUMPS

702 26. MIL-F-8615

27. FUEL JETTISONING SYSTEM

702 27. MIL-F-17874
MIL-A-19736

28. AUXILIARY POWER UNIT FUEL SYSTEM

702 28. MIL-F-8615

29. FUEL SYSTEM INSTRUMENTATION

702 29. MIL-I-7566
MIL-F-17874

30. FUEL TANK TESTS

702 30. MIL-T-6396
MIL-F-8615
MIL-T-8679
MIL-F-17874
MIL-D-23222

31. FUEL CONTENTS TEST
702 31. MIL-T-6396
MIL-I-7566
MIL-F-8615
MIL-T-8679
MIL-F-17874
MIL-D-23222

32. CROSS REFERENCES
702 32.

TABLE 1 LIST OF OTHER IMPORTANT REQUIREMENTS

CHAPTER 703: PNEUMATIC SYSTEMS

703	MIL-HDBK-221	FIRE PROTECTION DESIGN HANDBOOK FOR U.S. NAVY AIRCRAFT POWERED BY TURBINE ENGINES
	MIL-HDBK-336-1	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, GENERA CRITERIA VOL. 1
	MIL-HDBK-336-2	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, AIRFRAME VOL. 2
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-P-5518	PNEUMATIC SYSTEM, AIRCRAFT, DESIGN, INSTALLATION AND DATA REQUIREMENTS FOR
	MIL-T-5522	TEST REQUIREMENTS AND METHODS FOR AIRCRAFT HYDRAULIC AND EMERGENCY PNEUMATIC SYSTEMS
	MIL-P-8564	PNEUMATIC SYSTEM COMPONENTS, AERONAUTICAL, GENERAL SPECIFICATION FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-D-8804	DE-ICING SYSTEM, PNEUMATIC BOOT, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-B-81365	BLEED AIR SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-R-81367	RAIN REMOVAL SYSTEMS, AIRCRAFT WINDSHIELD, JET AIR BLAST

CONTROLLED DISTRIBUTION:

	MIL-P-87210	PNEUMATIC POWER SYSTEMS, HIGH PRESSURE
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS.VOL II - ROTARY WING AIRCRAFT

1. INTRODUCTION

703 1.	MIL-HDBK-221	PARA: 2.7, 2.8, 2.10
	MIL-STD-2069	PARA: 5.3.3.8, 5.3.3.11
	MIL-T-5522D	PARA: 1.1
	MIL-P-8564D	PARA: 3.1
	MIL-E-008593A	PARA: 3.7.8.2

2. GENERAL

703 2.	MIL-P-5518	
	MIL-P-8564D	PARA: 3.3

3. TEMPERATURE EFFECTS

703 3.	MIL-P-8564D	PARA: 3.3.2

4. SAFETY AND VULNERABILITY

703 4.	MIL-HDBK-336-1	
	MIL-HDBK-336-2	
	MIL-P-5518	
	MIL-P-8564D	PARA: 3.3.7

5. DEFINITIONS

703 5.

6.	PRESSURES	
703 6.	MIL-P-5518	
	MIL-P-8564D	PARA: 3.3.13, 3.3.14
7.	STRENGTH	
703 7.	MIL-P-5518	
	MIL-P-8564D	PARA: 3.3.11
8.	STRENGTH TESTS	
703 8.	MIL-P-5518	
	MIL-T-5522D	PARA: 3
	MIL-P-8564D	PARA: 4.5.3, 4.5.9
	MIL-D-23222A	PARA: 3.12.4
9.	FUNCTIONING, ENDURANCE AND ENVIRONMENTAL TESTS	
703 9.	MIL-P-5518	
	MIL-T-5522D	PARA: 3
	MIL-P-8564D	PARA: 4.5.4, 4.5.5, 4.5.6, 4.5.7, 4.5.8
	MIL-D-23222A	PARA: 3.12.4
	MIL-B-81365	PARA: 4.2.7
	MIL-R-81367	PARA: 4
10.	PRODUCTION TESTS	
703 10.	MIL-P-5518	
	MIL-P-8564D	PARA: 4

CHAPTER 704: HYDRAULIC SYSTEMS

704	MIL-STD-210	CLIMATIC INFORMATION TO DETERMINE DESIGN AND TEST FOR MILITARY SYSTEMS AND EQUIPMENT
	MIL-HDBK-221	FIRE PROTECTION DESIGN HANDBOOK FOR U.S. NAVY AIRCRAFT POWERED BY TURBINE ENGINES
	MIL-HDBK-336-1	SURVIVABILITY, AIRCRAFT, NONNUCLEAR GENERAL CRITERIA - VOL. 1
	MIL-HDBK-336-2	SURVIVABILITY, AIRCRAFT, NONNUCLEAR AIRFRAME - VOL 2
	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-B-5087	BONDING, ELECTRICAL AND LIGHTNING PROTECTION FOR AEROSPACE SYSTEMS
	MIL-W-5088	WIRING, AEROSPACE VEHICLES
	MIL-H-5440	HYDRAULIC SYSTEMS, AIRCRAFT, TYPES I AND II, DESIGN AND INSTALLATION REQUIREMENTS FOR
	MIL-F-5504	FILTERS AND FILTER ELEMENTS, FLUID PRESSURE HYDRAULIC MICRONIC TYPE
	MIL-R-5520	RESERVOIRS, AIRCRAFT, HYDRAULIC, NON-SEPARATED TYPE
	MIL-T-5522	TEST REQUIREMENTS AND METHODS FOR AIRCRAFT HYDRAULIC AND EMERGENCY PNEUMATIC SYSTEMS
	MIL-V-5524	VALVES, CHECK, HYDRAULIC, AIRCRAFT TYPE I SYSTEMS
	MIL-C-7413	COUPLINGS, QUICK DISCONNECT, AUTOMATIC SHUTOFF, GENERAL SPECIFICATION FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-H-8775	HYDRAULIC SYSTEM COMPONENTS, AIRCRAFT AND MISSILES, GENERAL SPECIFICATION FOR
	MIL-F-8815	FILTER AND FILTER ELEMENTS, FLUID PRESSURE, HYDRAULIC LINE, 15 MICRON ABSOLUTE AND 5 MICRON ABSOLUTE, TYPE II SYSTEMS
	MIL-H-8890	HYDRAULIC COMPONENTS, TYPE III, GENERAL SPECIFICATION FOR
	MIL-H-8891	HYDRAULIC SYSTEMS, MANNED FLIGHT VEHICLES, TYPE II DESIGN, INSTALLATIONS AND DATA REQUIREMENTS FOR GENERAL SPECIFICATION FOR
	MIL-R-8931	RESERVOIRS: AIRCRAFT AND MISSILE, HYDRAULIC, SEPARATED TYPE
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	MIL-F-83660	FILTER ELEMENT, FLUID PRESSURE, HYDRAULIC, AIRCRAFT DISPOSABLE, 5 MICRON ABSOLUTE
	MIL-F-83861	FILTER ELEMENTS, NONCLEANABLE, FLUID PRESSURE, HYDRAULIC LINE, 5 AND 15 MICRON ABSOLUTE, TYPE II SYSTEMS, GENERAL SPECIFICATION FOR
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	MIL-V-85245	VALVE, RELIEF, HYDRAULIC, HIGH RESPONSE, TYPE II SYSTEMS, GENERAL SPECIFICATION FOR

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704 1.	MIL-H-5440G	PARA: 1.1
	MIL-H-8775D	PARA: 1.1
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704.10 MIL-H-8890
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	MIL-HDBK-221	FIRE PROTECTION DESIGN HANDBOOK FOR U.S. NAVY AIRCRAFT POWERED BY TURBINE ENGINES
	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-STD-2175	CASTINGS, CLASSIFICATION AND INSPECTION OF
	MIL-T-5955	TRANSMISSION SYSTEMS, VTOL-VSTOL, GENERAL REQUIREMENTS FOR
	MIL-T-7101	TRANSMISSION; POWER, CONSTANT SPEED, GENERAL SPECIFICATION (AIRCRAFT USE)
	MIL-S-7470	SHAFT, POWER TRANSMISSION, AIRCRAFT ACCESSORY, GENERAL SPECIFICATION FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-E-8970	ENGINES AND RELATED PROPULSION AND POWER EQUIPMENT, AIRCRAFT, ACCEPTANCE TESTS OF, SAMPLING PLAN FOR, STATISTICAL
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83870	FILTER ELEMENT, FLUID PRESSURE, TRANSMISSION, AIRCRAFT, DISPOSABLE, 20 MICRON ABSOLUTE

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS.VOL II - ROTARY WING AIRCRAFT

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	MIL-T-7101A	PARA:
	MIL-E-8593A	PARA: 3.1, 3.1.2.7

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	MIL-HDBK-241	DESIGN GUIDE FOR ELECTROMAGNETIC INTERFERENCE (EMI) REDUCTION IN POWER SUPPLIES
	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-HDBK-336-1	SURVIVABILITY, AIRCRAFT, NONNUCLEAR, GENERAL CRITERIA - VOL 1.
	MIL-HDBK-336-2	URVIVABILITY, AIRCRAFT, NONNUCLEAR, AIRFRAME - VOLE. 2
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	MIL-STD-705	GENERATOR SETS, ENGINE DRIVEN, METHODS OF TEST AND INSTRUCTIONS
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	MIL-STD-810	ENVIRONMENTAL TEST METHODS
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	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
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	MIL-G-21480	GENERATOR SYSTEM, 400 HERTZ ALTERNATING CURRENT, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-R-23761	REGULATORS, VOLTAGE, AND CONTROL PANELS, AIRCRAFT DIRECT CURRENT GENERATOR, GENERAL SPECIFICATION FOR
	MIL-E-24021	ELECTRIC POWER MONITORS, EXTERNAL, AIRCRAFT
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	MIL-E-85583	ELECTRIC POWER GENERATING CHANNEL, VARIABLE INPUT SPEED, ALTERNATING CURRENT, 400 Hz, AIRCRAFT; GENERAL SPECIFICATION FOR

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	MIL-STD-280	DEFINITIONS OF ITEM LEVELS, ITEM EXCHANGEABILITY, MODELS AND RELATED TERMS
	MIL-STD-454	STANDARD GENERAL REQUIREMENTS FOR ELECTRONIC EQUIPMENT
	MIL-STD-461	ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR THE CONTROL OF ELECTROMAGNETIC INTERFERENCE
	MIL-STD-462	ELECTROMAGNETIC INTERFERENCE, CHARACTERISTICS, MEASUREMENT OF
	MIL-STD-469	RADAR ENGINEERING DESIGN REQUIREMENTS, ELECTROMAGNETIC COMPATIBILITY
	MIL-STD-704	AIRCRAFT ELECTRIC POWER CHARACTERISTICS
	MIL-STD-781	RELIABILITY TESTING FOR ENGINEERING DEVELOPMENT, QUALIFICATION, AND PRODUCTION
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	MIL-STD-877	ANTENNA SUBSYSTEMS, AIRBORNE CRITERIA FOR DESIGN AND LOCATION OF
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	MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
	MIL-STD-2164	ENVIRONMENTAL STRESS SCREENING PROCESS FOR ELECTRONIC EQUIPMENT
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
	MIL-W-5088	WIRING, AEROSPACE VEHICLE
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	MIL-E-7080	ELECTRIC EQUIPMENT, AIRCRAFT, SELECTION AND INSTALLATION OF
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	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
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	MIL-A-25730	ANTENNA SUBSYSTEMS FOR AIRBORNE IDENTIFICATION AND NAVIGATION EQUIPMENTS, GENERAL SPECIFICATION FOR

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	MIL-STD-241	DESIGN GUIDE FOR ELECTROMAGNETIC INTERFERENCE (EMI) REDUCTION IN POWER SUPPLIES
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	MIL-HDBK-274	ELECTRICAL GROUNDING FOR AIRCRAFT SAFETY
	MIL-HDBK-419	GROUND BONDING, AND SHIELDING FOR ELECTRONIC EQUIPMENTS AND FACILITIES BASIC THEORY AND APPLICATIONS
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	MIL-STD-877	ANTENNA SUBSYSTEMS, AIRBORNE CRITERIA FOR DESIGN AND LOCATION OF
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
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	MIL-1-18802	FUEL AND OIL LINES, AIRCRAFT, INSTALLATION OF
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REFERENCES

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709	MIL-HDBK-221	FIRE PROTECTION DESIGN HANDBOOK FOR U.S. NAVY AIRCRAFT POWERED BY TURBINE ENGINES
	MIL-HDBK-244	GUIDE TO AIRCRAFTS/STORES COMPATIBILITY
	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-461	ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY CONTROL OF ELECTROMAGNETIC INTERFERENCE, REQUIREMENTS FOR THE
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	MIL-E-25499	ELECTRICAL SYSTEMS, AIRCRAFT, DESIGN AND INSTALLATION OF, GENERAL SPECIFICATION FOR

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	MIL-F-87168	FIRE AND EXPLOSION HAZARD PROTECTION SYSTEMS, GENERAL SPECIFICATION FOR

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PART 8

MAINTENANCE

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CHAPTER 800

GENERAL MAINTENANCE REQUIREMENTS

1 INTRODUCTION

1.1 The requirements of this Part aim at the simplification of inspection, servicing, replacement of components and repair by Service personnel in order to reduce the time, effort and cost spent in maintaining the rotorcraft in an airworthy condition, thereby increasing its operational effectiveness.

1.2 The rotorcraft and its equipment shall be designed to reduce to a minimum the need for servicing or inspection other than the need to replenish consumable stores. (see para 2 below)

1.3 Routine servicing, adjustments, lubrication, repairs and replacements of component parts shall be possible with ease, using, as far as possible, standard tools and equipment and with the minimum of manhours and without need for access equipment. (see para 2 below)

2 DESIGNING FOR MAINTENANCE

2.1 During the early design stages of rotorcraft special attention shall be given to ensure:

(i) that the engine and its equipment can be replaced easily and quickly.

(ii) that inspection, maintenance and repair of parts of a rotorcraft structure and installations can easily be accomplished.

(iii) easy removal and replacement of damaged sections of a rotorcraft structure and systems.

2.2 In addition consideration shall be given to the following requirements (the list is not exhaustive).

2.2.1 Reduction of complexity by:

(i) providing adequate accessibility, workspace and work clearance.

(ii) providing for interchangeability of similar materials, components and parts

(iii) use of standard parts (see Chap 100 para 2 and Chap 400 para 3).

(iv) logically sequenced maintenance tasks

2.2.2 Reducing the frequency of servicing and maintenance activities by using:

(i) fail safe or damage tolerance design features in essential services.

(ii) components that require little or no preventive maintenance

(iii) tolerances that allow for use and wear in service

(iv) adequate corrosion prevention and control features (see Chap 407)

2.2.3 Reducing Maintenance Downtime by designing for

(i) rapid and positive detection of a malfunction

(ii) ease of fault rectification

(iii) rapid and positive adjustment and calibration

(iv) rapid testing or checking of rectification

(v) easy repair or replacement of components and parts that are vulnerable to normal operating hazards and battle damage avoiding time consuming and tedious processes. (see para 5.1 below)

2.2.4 Reducing of maintenance support costs by limiting

(i) number and variety of tools and support equipment

(ii) the need for special tools and support equipment

(iii) the need for 3rd and 4th line maintenance (see leaflet 800/1)

(iv) the need for extensive or complex maintenance data

2.2.5 Reducing the possibility of maintenance error by eliminating:

(i) the possibility of incorrect assembly, installation or connection (see Chap 100 para 7)

(ii) dirty, awkward and tedious job elements

(iii) ambiguity in maintenance labelling, coding and technical data

3 HANDLING

3.1 Requirements for transport, handling and storage are given in Chapter 801 and the design shall permit the use of standard equipment wherever possible. Where standard equipment or tools cannot be employed, the special items shall be included with the design of the rotorcraft.

4 ROUTINE SERVICING AND REPAIR

4.1 Requirements for routine servicing are given in Chapter 802 but in the general design, consideration shall be given to the possibility of reducing repair work arising from the normal operating hazards of the rotorcraft type, (e.g. birdstrike, heavy or wheels up landings, tail scrapes). Those portions of the structure likely to suffer such damage should be quickly and easily replaceable.

4.2 Sound proofing, heat shielding and furnishings shall be arranged for easy access to hidden components requiring maintenance.

4.3 For Naval rotorcraft a means by which the alignment of the fuselage can be checked should be provided. Any fittings necessary for this purposes may be removable.

5 REPLACEMENTS AND INTERCHANGEABILITY (see also Chapters 804 and 805)

5.1 The basic requirements for ease of replacement are:

(i) Accessibility of joints, connectors and fittings

(ii) Interchangeability of components

(iii) Minimum use of special tools and equipment

6 ACCESSIBILITY

6.1 Where practical access shall be provided to permit inspection (including inspection of principal structural elements and control systems), replacement of parts, adjustment and lubrication as necessary for continued airworthiness. Where is it not practical to provide direct visual access, non-destructive inspection aids may be used to inspect structural elements if it can be shown that the inspection is effective.

6.2 Where practical access shall be provided through hinged cowls or panels. Engine cowls shall be hinged to the airframe.

7 STOWAGE

7.1 Stowages, easily accessible to servicing crews, shall be provided for Travelling Servicing Notes and forms, Form F700, cross servicing guide, pitot head covers, static vent plugs, safety and warning pennants, turn round tools, undercarriage locks and special to type servicing couplings.

8 CONDITION INSPECTION EQUIPMENT

8.1 The lives of many rotorcraft components are influenced by the duty cycles that those components experience. Although statistical analyses are used to evaluate safe lives, a more accurate assessment of the life expended can be made through the use of monitoring equipment, (e.g. elapsed time indicators, magnetic chip detectors, vibration analyses, etc). Consideration shall be given to the incorporation of such monitors in the rotorcraft design and, where possible, they should be easily read in situ and from the ground.

9 FAULT DIAGNOSIS

9.1 To assist in fault diagnosis, rotorcraft system designs shall include where appropriate, facilities for built in test equipment (BITE) and the use of accessible self diagnostic maintenance panels to indicate, to an agreed level, the location of system faults. Where ground test equipment is required for maintenance of complex systems, an automatic test facility shall be considered.

LEAFLET 800/1

GENERAL MAINTENANCE REQUIREMENTS

NOTES ON THE ORGANISATION OF SERVICING IN

THE ROYAL NAVY, THE ARMY AND THE ROYAL AIR FORCE

1 INTRODUCTION

1.1 As an aid to an appreciation of the requirements of Part 8, some understanding of the maintenance organisation in the Services is desirable.

1.2 The policy of the Air Force Department of MOD is to establish a small specialist project team for each new major project. This project team will be controlled by the Central Servicing Development Establishment (CSDE) and will be located at the Contractor's works as soon as practicable in the early design stages. In addition to advising on maintenance policy, the team will make recommendations on behalf of the Chief Engineer (RAF), MOD, where changes in design are considered desirable to facilitate economic servicing and to improve reliability.

1.3 The Navy Department of MOD operates a similar scheme with a team from the Naval Air Technical Evaluation Centre (NATEC). This team is called the Special Maintenance Party (SMP).

2 ROTORCRAFT PREVENTATIVE MAINTENANCE

2.1 Preventative maintenance is systematic and work is undertaken in order to reduce the probability of failure. For Royal Air Force rotorcraft it includes 3 types of servicing, (i.e. flight servicing, routine servicing and out of phase servicing).

2.2 For Royal Air Force rotorcraft, the provisions of Air Publication (AP)100A-01 Leaflet 311 apply for rotorcraft preventative maintenance procedures.

2.3 For Royal Navy rotorcraft, a system of flight servicing and routine servicing is operated. Routine servicing operations are of various frequencies, either calendar based or flying hour based and may be advanced as required to provide flexibility. Full details are given in Air Publication (AP)100N-0140.

2.4 For Army rotorcraft the provisions of Electrical and Mechanical Engineering Regulation (EMER) A 100 apply for rotorcraft preventative maintenance procedures.

3 LINES AND DEPTHS OF MAINTENANCE

3.1 Maintenance activities within the three Services are assigned to lines and depths of maintenance.

3.2 FIRST LINE

3.2.1 The maintenance organisation immediately responsible for both the preparation for operation and the initial diagnosis of defects of complete rotorcraft and weapons systems.

3.3 SECOND LINE

3.3.1 The maintenance organisation under the control of:

(i) for the Royal Navy, the commanding officer of a station, ship or unit,

(ii) for the Army, the officer commanding a field workshop (aircraft),

(iii) for the Royal Air Force, the commanding officer of a station established to provide support for those types of rotorcraft and weapons systems operated at the station, ship, unit or workshops, but excluding the organisation within first line.

3.4 THIRD LINE

3.4.1 The maintenance organisation within the Services, but excluding the organisation within first and second line.

3.5 FOURTH LINE

3.5.1 The industrial maintenance organisation providing repair, modification and reconditioning under contract.

3.6 DEPTH A

3.6.1 That maintenance which is directly concerned with preparing end items for use, and keeping them in day-today order. It may include such operations as functional checking, replenishment, flight or daily servicing, diagnosis of defects, simple rectification by replacement or adjustment, rearming and role changing.

3.7 DEPTH B

3.7.1 That maintenance which is required on end items and assemblies which are temporarily unserviceable or which require servicing. This may include calendar, operation or operating hour based servicing, embodiment of prescribed modifications, bay servicing of assemblies, and rectification of defects beyond depth A.

3.8 DEPTH C

3.8.1 That maintenance which is the repair, partial reconditioning and modification requiring special skills, special equipment or relatively infrequently used capability which is not economic to be provided generally, but which is short of complete strip, reconditioning and assembly.

3.9 DEPTH D

3.9.1 That maintenance which is full reconditioning, major conversion, major rework, or such repair as involves work of this depth.

4 SERVICING SCHEDULES

4.1 The requirement for rotorcraft preventative maintenance is implemented by the preparation, issue and use of servicing schedules for each type and, if necessary, mark of rotorcraft. Each schedule specifies the work to be undertaken and frequency of its arising, and is approved by the concerned rotorcraft engineering authority.

4.2 For the operation and amendment of Royal Air Force rotorcraft schedules, the provisions of Air Publication (AP) 100B-01 Order 0561 apply.

5 TOOLS AND EQUIPMENT

5.1 It is a requirement that as much as possible of 1st and 2nd line servicing of Army and RAF rotorcraft should be within the capacity of tradesmen using only general purpose tools from within the Army and RAF tool inventories.

5.2 In the Royal Navy and RAF, standard tool kits for tradesmen are no longer issued. Tools for maintenance of Naval and RAF rotorcraft are issued under a system of Tool Control. Tool outfits are provided for each type of rotorcraft and for certain workshops and maintenance tasks.

5.3 Where use of special tools cannot be avoided they should be simple and robust and capable of use by men wearing NBC protective clothing.

6 ROTORCRAFT STRUCTURAL INTEGRITY

6.1 The RAF policy and organisation for the maintenance of the structural integrity of rotorcraft is detailed in AP 100A-01 Leaflets 315 (Structural Integrity of RAF Aircraft), 316 (Structural Integrity Examination of RAF Aircraft) and 320 (Aircraft Corrosion Control in the Royal Air Force).

CHAPTER 801

TRANSPORT, HANDLING AND STORAGE

1 TRANSPORT OF COMPONENTS

1.1 So that, as far as possible, rotorcraft components may be packed and transported by normal transport facilities, the requirements of paras. 1.2 to 1.6 inclusive shall be met on the following basis:

(i) for rotorcraft up to 34,020kg all-up weight, the requirements shall be met in full, and

(ii) for rotorcraft over 34,020kg all-up weight, the requirements shall be considered individually in the light of the role and construction of the rotorcraft.

1.2 It shall be possible to sub-divide the rotorcraft into components as indicated in Table 1 and the individual components including packing cases, shall not exceed the size and weight limitations given in Table 2. Case wall thickness shall be taken to be:

(i) 100mm for sides and ends, and

(ii) 150mm for top and bottom.

Where it is not possible to design the airframe with suitable breakdown or transport joints the maximum facilities shall be provided for on-side "fly-in" repairs.

1.3 All components shall be designed to be conveniently stored and handled without elaborate supporting crates; jury struts should be used where necessary to render components self-supporting.

1.4 Components should have no projecting portions liable to be damaged and brackets in this category should be readily removable for packing and transport.

1.5 Component parts of main and tail rotor units shall be packed vertically or flat in their cases and diagonal packing to comply with the limiting dimensions will only be accepted in exceptional circumstances. It should be possible to store main and tail rotor blades with the chord vertical.

1.6 As far as possible components shall incorporate facilities for handling and securing them in storage and transport containers so that the containers do not have to provide complicated parts to give the necessary support.

2 GROUND HANDLING

2.1 The rotorcraft shall be designed with a view to the fullest possible use being made of standard R.A.F. or Naval equipment, tools and ground support equipment (GSE) as appropriate, for servicing, as, for example:

(i) moving the rotorcraft on the ground,

(ii) filling tanks (see Chapter 701)

(iii) routine overhauls of the engine(s) and airframe,

(iv) removing and handling an engine change unit,

(v) removing and handling an auxiliary airborne power plant, if fitted,

(vi) jacking-up the rotorcraft, and

(vii) re-arming.

2.2 Typical items of G.S.E. which the Contractor may be required to supply are as follows:

(i) rotorcraft gantry for engine changing,

(ii) ladders for reaching the engine(s), tanks or other parts,

(iii) jacks to lift and support the rotorcraft,

(iv) tail trestles,

(v) tail towing and steering arms,

(vi) tackle for handling auxiliary airborne power plants, if fitted, and

(vii) slings (see para 4).

2.3 Particulars of standard equipment are obtainable, on written application, from the Rotorcraft Project Director.

2.4 When Special to Type (STT) GSE is required, consideration shall be given to:

(i) ease of use in any operational environment.

(ii) reliability.

(iii) operation, handling and maintenance by personnel wearing NBC protective clothing.

3 JACKING

3.1 It shall be possible to raise the rotorcraft on three primary jacking points for the following purposes:

(i) as a routine measure for the testing, removal and replacement of all retractable mechanisms and their associated equipment, and for those servicing operations which include rigging checks and adjustments, and

(ii) as part of the recovery operations following an incident.

3.2 It shall be possible to jack up the rotorcraft at or near the axles for the purpose of changing an individual wheel having a deflated tyre.

3.3 The rotorcraft in any operational configuration and carrying any authorised combination of stores shall be stable when supported upon jacks at all masses between the empty mass and the maximum take-off mass. The effect of movable masses (e.g. fuel, personnel, equipment and tools) shall also be considered.

3.4 The minimum clear space to be provided under each rotorcraft primary jacking point, to accommodate a tripod jack, shall conform to Fig.1[1]

3.5 Jacking pads shall comply with the requirements of BS C 12 and shall be fitted so that when the rotorcraft is in the normal jacked-up position, the axial centre line of the pad is within 3° of the vertical. The pad may be permanently fixed to the jacking point or may be a detachable fitting.[2]

3.6 The jacking points shall be marked in accordance with the requirements of Defence Standard 05-18.

3.7 If detachable pads are used at primary jacking points, they shall be chosen from the standard pads listed in Defence Standard 17-3. The sockets in the rotorcraft shall be designed so that the pads can be inserted and removed without difficulty. When not in use, detachable jacking pads will not normally be required to be stowed in the rotorcraft.

3.8 It shall be possible to meet the requirements of paras 3.1 and 3.2 using jacks chosen from the standard range and approved by the Rotorcraft Project Director, from whom advice on the current range of standard jacks can be obtained.

4 SLINGING

4.1 RAF ROTORCRAFT

4.1.1 Rotorcraft with a maximum design take-off weight not exceeding 20,412kg shall have provision for the attachment of lifting slings to enable the rotorcraft to be lifted clear of the ground on one hoist.

4.2 NAVAL ROTORCRAFT

4.2.1 Provision shall be made for slinging the complete rotorcraft on one hoist with all folding components both folded and spread, and with jury struts in position. It shall be possible to sling the complete rotorcraft when in the embalmed state.

4.2.2 Rotorcraft with a maximum design take-off weight exceeding 9,072kg shall have provision for slinging by both one and two hoists.

4.2.3 The height of the slinging ring shall be the minimum possible and shall not exceed 4.6m above the ground when in the slinging attitude.

4.2.4 For rotorcraft which can take-off or alight on water, the sling shall be stored at or near the slinging points.

4.3 COMPONENTS

4.3.1 Slinging facilities shall be provided on components too heavy for manhandling.

4.3.2 When required, a detachable hoist for fitting to the rotorcraft shall be provided to permit the installation and changing of heavy components, engines or rotors.

4.3.3 On Naval rotorcraft when the height of the tail pylon including rotor blades exceeds 3.0m above ground level, provision shall be made for the attachment of a lifting frame to the underside of the tail pylon in addition to an overhead sling.

4.4 MARKING OF SLINGING POINTS

4.4.1 All slinging points shall be marked in accordance with the requirements of DEF STAN 05-18.

5 GROUND TOWING (see also Chapter 308)

5.1 Provision shall be made on all rotorcraft for towing and pushing by a standard towing vehicle with a towing arm attached to the auxiliary undercarriage unit or neighbouring strong points. Manoeuvring by these means shall be possible in all turns corresponding to the range of castoring angles called for in the rotorcraft specification (see Chapter 300 para 6) even when undercarriage ground locking devices and/or flying control locks are engaged (see also Chapter 306, para 4 and Chapter 309, para 2.4).

5.2 TOWING POINTS

5.2.1 All towing points shag be readily accessible to ground personnel and shall be marked in accordance with the requirements of DEF STAN 05-18.

5.2.2 On Naval rotorcraft, auxiliary undercarriage towing points shall be located at the axle.

5.2.3 The dimensions of towing points on rotorcraft shall conform to DEF STAN 17-10[3].

5.2.4 Where towing lugs are provided at the main undercarriage, they shall, where possible, lie in a horizontal plane when the rotorcraft is at rest on the ground and shall be located as close to the centre of the main wheel as practicable.

5.3 LOAD LIMITATIONS

5.3.1 All towing arms, with the exception of those for use with shipborne rotorcraft, shall embody a load limiting device to prevent the load transmitted to the rotorcraft from causing structural damage. The operating load shall be tested and shall not be less than $0.15W_\tau$ where W_τ is the maximum design take-off weight.

5.3.2 Provision shall be made to ensure that the driver of the towing vehicle can readily ascertain the maximum permissible towing arm angle.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	11/7	
2	17/1	
3	25/9	3278
	25/10	

TABLE 1

SIZE CATEGORIES FOR ROTORCRAFT

Category	All-up weight	Transport method on ship	Component category (See Table 2)
1	Not exceeding 4,536kg	Between decks	A and B
2	4,536kg to 13,608kg	Between decks and in hold	A, B and C
3	13608kg to 34,020kg	1 crate deck cargo, remainder between decks and in hold	1, D or E; remainder, A, B or C
4	Exceeding 34,020kg	Special arrangements	3 special loads; remainder A, B or C

TABLE 2

SIZE CATEGORIES FOR PACKED COMPONENTS

Component Category	Maximum weight (including case) (tonnes)	Maximum external dimensions of case (metres)			Transportable by	
		Length	Height	Width	Road	Ship
A	3	9.2	2	2	3 tonne high long trailer	Between decks
B	5	4.6	2	2.5	5 tonne flat platform trailer	Between decks
C	5	9.2	3	4.6	5 tonne low loader	Hold
D	6	11.6	3	3.6	6 tonne trailer	Deck
E	15	15.2	3	3.6	20 tonne trailer	Deck

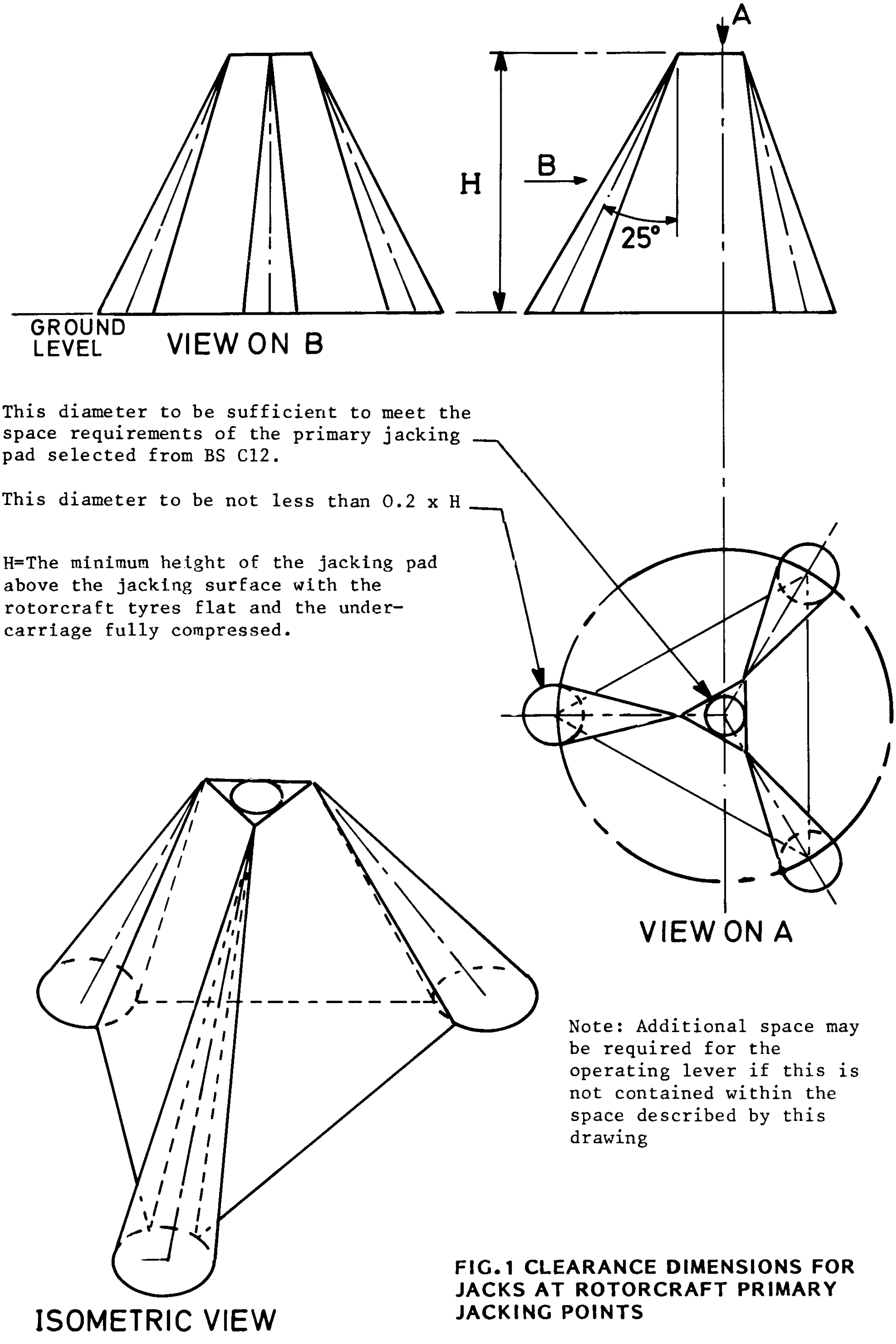

FIG.1 CLEARANCE DIMENSIONS FOR JACKS AT ROTORCRAFT PRIMARY JACKING POINTS

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TRANSPORT, HANDLING AND STORAGE

REFERENCE PAGE

Specifications

AD2/SRD/902/P	..	..	Lifting tackle

Defence Standards

05-18	..	..	Symbol markings for servicing and safety/hazard points on aircraft.
17-1	..	..	Aircraft lifting slings - Aperture of terminal ring or link.
17-3	..	..	Adaptors, aircraft jacking points (detachable jacking pads).
17-10	..	..	Towing attachments for aircraft.

British Standards

C12	..	..	Form of contact surface of aircraft jacking pads.
C15	..	..	Coupling dimensions for aircraft-to-tractor tow bar connections.

CHAPTER 802

ROUTINE SERVICING

1 GENERAL

1.1 Routine servicing as conducted by Service Units consists of inspection and operations necessary to keep the rotorcraft in a serviceable and operable condition and facilities to assist such work shall be provided.

1.2 It should be remembered that an operation easily performed in a well-lighted and heated workshop may prove difficult in the open and under adverse climatic conditions with limited equipment, and wearing NBC protective clothing. Similarly, operation from hardened aircraft shelters (HAS) or under deployed conditions in the field may impose other restrictions.

1.3 The design shall cater for adequate servicing with the least expenditure of manhours (see also Chapter 800). The following servicing operations:

(i) flight servicing,

(ii) replenishment and draining of consumable supplies,

(iii) re-arming,

(iv) removal, replacement and functional testing of all avionic equipments,

shall be possible, wearing the appropriate clothing, under the maximum and minimum ground temperature conditions given in Chapter 101 and in all winds of up to 28m/s (55Kn). No tools shall be necessary for carrying out the operations (i) and (ii) above and where possible, without great increase in weight or complication, this requirement shall also apply to operations (iii) and (iv) above.[1]

1.4 On naval rotorcraft it shall be possible to carry out the operations set down under para 1.3 when the rotorcraft is secured, rotors folded, to the deck of an aircraft carrier rolling 15° either side of the vertical.

1.5 On transport rotorcraft, servicing should be possible with the minimum need to enter the cabins.

1.6 TURN ROUND

1.6.1 The turn round time, i.e., the time taken to replenish all consumable stores and to make a visual Between-Flight Servicing, shall be as short as possible. The Contractor shall demonstrate that the time does not exceed that given in the Staff Requirement.

1.6.2 On naval rotorcraft, the demonstration shall be made with rotors folded, when necessary, but the time taken in folding and spreading is not included in the turn round time.

1.6.3 The demonstration shall be made using standard Service equipment with personnel wearing clothing appropriate to the cold weather conditions laid down in para 1.3 and wearing NBC suits as appropriate but making the test under sheltered temperate conditions. The Contractor shall at an early stage agree with the appropriate Service authority (Central Servicing Development Establishment for RAF rotorcraft and Naval Air Technical Evaluation Centre for naval rotorcraft) the number of personnel, the equipment and clothing used in the demonstration.

1.6.4 Servicing shall be possible without damage to the rotorcraft's finish.

2 INSPECTION

2.1 Parts which require frequent inspection or replacement shall be easily accessible and fully visible to a tradesman working on them. If direct access is not available then suitable inspection panels shall be provided. Such panels in pressure cabins shall be self sealing and shall not require the application of sealing compound to make an airtight joint.

2.2 On naval rotorcraft, it shall be possible to carry out all Flight Inspections without the use of ladders and all joints and moving parts of all folding mechanisms shall be readily accessible for inspection and/or servicing.

3 LUBRICATION

3.1 Where sealed or self lubricating bearings are not used, lubrication of all parts of the controls shall be provided for.

3.2 To permit the use of standard lubricating guns specified in DEF STAN 49-2, all lubricating nipples shall be designed and installed in accordance with the appropriate requirements of BS Specification SP 115 to 117[2] or SBAC Standard AS 44401 to 44403[3]. Straight nipples to BS Specification SP 115 or SBAC Standard AS 44401 should be used whenever possible.

3.3 The rotorcraft shall be designed to use the lubricants listed in DEF STAN 01-5 wherever possible and as few different kinds as possible shall be used.

4 GASEOUS SYSTEMS

4.1 Provision shall be made for charging the gaseous systems in situ.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	11/2	3294
2	17/8	
3	-	3766

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ROUTINE SERVICING

REFERENCE PAGE

<u>MOD (PE) Specifications</u>

D Eng R D 2089	..	..	Aero-engine and propeller and engine accessory tools development procedure.
D Eng R D 2081	..	..	Specification of tools for servicing aero-engines and aero-engine accessories and propellers.

<u>Defence Standards</u>

01-5	..	..	Fuels, lubricants and associated products
16-17	..	..	Air supply and test connections for aircraft.
49-2	..	..	Lubricating guns, hand and guns fluid, direct delivery.

CHAPTER 803

REPAIRS

1 GENERAL

1.1 Consideration shall be given in the design of the airframe to ease of repair in service. Particular cure shall be taken to avoid conditions where defects or damage of a minor nature cannot be repaired by Service units but require major replacements or the use of special equipment. All repairs shall be designed with a view to their being done by Service personnel and, as far as possible, without need to dismantle the surrounding structure. Thus, types of construction using special techniques are not acceptable if in Service, repair of minor damage is thereby precluded.

2 STRENGTH

2.1 Repairs shall comply with all relevant design requirements for the rotorcraft as a whole. The design of the repairs shall, in general, be such that the reserve factor of the repaired member is not lower than 1.2 or that of the undamaged member, whichever is the less. It is however, undesirable to repair one member in such a manner that its strength is relatively much below that of the surrounding structure. If the application of a particular repair would considerably reduce the reserve factor of the member, care shall be taken to ensure that a combination of two or more repairs could not reduce the reserve factor below the safe limit. Alternatively, a warning shall be included drawing attention to, and prohibiting the application of, the dangerous combination. If compliance with these requirements is difficult to attain, the matter shall be referred to the Director, RAE.

3 MATERIAL

3.1 Standard materials and methods of repair shall be employed where possible.

4 COMPOSITE MATERIALS

4.1 Composite structural repair is in a developing stage, but because of the need to use specialist equipment, environmental conditions, material handling difficulties and the need for careful lay up, most repairs are likely to be undertaken at 2nd, 3rd and 4th line.

4.2 Repair operations should be of a flush or patch configuration and spreaders and compatible fasteners shall be used where applicable on rotorcraft panels and cowlings. All repairs shall comply with the relevant design requirements for the whole rotorcraft.

4.3 Factors that shall be considered for the design of composite repairs are:

(i) Environmental conditions to ensure temperature, humidity and cleanliness are compatible with the repair.

(ii) the dressing of composite materials will produce highly conductive fibres, therefore, electrical equipment may need to be screened. The fibres also act as an irritant.

(iii) the need for an NDT inspection before and after the repair to ensure integrity of the surrounding structure and the repair.

(iv) the provision of a scrim cloth and or an adhesive film as an insulator because galvanic action will occur between composite materials and non ferrous metals.

(v) provision for accessibility, replacement of parts, and inspection in composite material enclosed areas.

(vi) greases, oils, chemicals and fuels can have a detrimental affect on composite material resins.

(vii) the provision of an anti fretting strip to the underside of panels and, cowlings prior to fitment ('Limpet suction cups' may be used for removal if restriction is felt).

(viii) the need to maintain lightning strike and EMC protection where applicable.

CHAPTER 804

REPLACEMENT OF COMPONENTS

1 DISMANTLING AND ERECTION

1.1 The construction of component parts of the rotorcraft shall be such as to permit them to be rapidly dismantled and erected by Service Units. Any equipment for this purpose other than standard Service items shall be supplied for use with the rotorcraft.

1.2 Components liable to damage such as landing gear, rotors, ailerons, pitot head, aerials, etc., shall be easily replaced.

1.3 Where it is necessary to break the fuselage to remove an engine, fuel tanks, or other components, consideration shall be given to providing:

(i) a quick release mechanism or a minimum number of bolts, and

(ii) quick release couplings for all pipelines and controls and plug and socket joints for electric cables at, or accessible to, the break joint in the fuselage.

2 ENGINE INSTALLATION

2.1 The engine installation shall be in accordance with the requirements of Chapter 700.

3 PIPELINES

3.1 Attention shall be given to the provision of easy means of replacement of pipelines. The grouping together of pipelines for ease of access shall not significantly increase the vulnerability of the rotorcraft to battle damage.

4 INSTRUMENTS

4.1 The instrument panel design and location shall make it easy to facilitate removal and installation of the instrument.

5 CONTROL CABLES OR RODS

5.1 Cables must be so arranged that they are easy to identify, adjust and to replace. Turnbuckles, other adjustment points, fairleads and other points in a system which might cause difficulty when replacing a cable, should be placed in positions where they are readily visible and accessible, an inspection panel being fitted if necessary. Where a cable passes round a pulley, joints may be inserted in the run of the cable to facilitate replacement of that portion which travels round the pulley.

5.2 Any shear connection in a flying control or engine control run, failure of which would lead to a loss of either control or motive power shall be either:

(i) an acceptable permanently locked connection, or

(ii) be provided with a secondary means of retention such that, once the item is placed in position, the secondary retaining device becomes automatically effective in preventing it from dropping out of position even though the usual retaining device may have been omitted. This secondary device should be automatic in operation and should not depend upon maintenance personnel remembering to carry out a separate action such as the bending of locking tabs or the fitting of locking wire. Secondary means of retention which depend upon friction or springs are usually acceptable.

5.3 Standard control rod eye ends shall be used in accordance with Chapter 400 para 3.

6 ELECTRICAL AND AVIONIC INSTALLATION

6.1 Electrical cables should be marked at frequent intervals with a separate code identity for each cable so that they are easy to identify and replace. Consideration shall be given to the provision of redundancy and separate routeing to reduce essential system vulnerability to battle damage.

6.2 Where appropriate, avionic and electrical systems shall be designed with line replaceable units permitting rectification of a system defect by fitting a replacement unit; the unserviceable unit then being repaired off the rotorcraft.

6.3 Avionic and electrical component replacement units likely to be carried by one man shall not exceed 27Kg (60lb), and ideally should not exceed 18Kg (401b). All units exceeding 27Kg (60lb) shall be fitted with suitable slinging points.

CHAPTER 805

INTERCHANGEABILITY

1 INTRODUCTION

1.1 This Chapter provides interchangeability requirements for production rotorcraft and their accessories. On prototype rotorcraft, interchangeability is not a requirement, but the provision of complete interchangeability shall receive full consideration in the design of such rotorcraft. See also DEF STAN 05-123, Chapter 201.

2 BASIC REQUIREMENT

2.1 All items (components, assemblies or sub-assemblies) of airframes, accessories and spares shall be designed and manufactured to ensure that they are capable of being replaced without degrading their functional capability so that any item may be replaced with a stock item of the same standard unless subsequent authorised modification action has affected or nullified the item that is in question. wherever practical complete assemblies shall not be 'handed'. The procedures for granting exemption from full interchangeability are contained in DEF STAN 05-123, Chapter 201.

3 DEFINITIONS

3.1 FULLY INTERCHANGEABLE

3.1.1 A fully interchangeable item is one which shall be capable of being installed without alteration being necessary to the item, its associated interfaces or its counterparts other than by adjustment or replacement of shims, serrated washers, tab washers, seals or expendable locking devices (wire locking, split pins or tab washers). The item when installed shall be capable of meeting all the requirements of the original item it is replacing in all characteristics (physical and functional).

3.1.2 The Rotorcraft Specification will state where on a particular project a fully interchangeable item is required to be capable of being changed within a specified timescale.

3.1.3 Full interchangeability of an assembly does not normally imply that all details and components of that assembly are considered in themselves to be fully interchangeable.

3.2 REPLACEABLE

3.2.1 A replaceable item has similar characteristics to a fully interchangeable item except that certain defined features may be subject to alteration to facilitate its installation.

Note: Spares for these items may be supplied with trim allowances and/or in the undrilled condition but this must be stated on the spares drawing.

3.3 PHYSICAL CHARACTERISTICS AND FUNCTIONAL CHARACTERISTICS

3.3.1 The physical and functional characteristics shall be as defined in DEF STAN 05-123, Chapter 201.

4 LIMITS

4.1 Limits shall be clearly stated on the drawings for all dimensions affecting assembly or functioning of the parts.

5 JIGS, TEMPLATES AND GAUGES

5.1 In the construction of all parts, the Contractor shall provide and use such jigs, templates and gauges as are necessary to meet the interchangeability requirements, and shall maintain them so that the approved initial standard is not deviated from during their period of use.

6 DIMENSIONING OF DRAWINGS

6.1 In dimensioning the drawings of parts, the individual dimensions shall be given from a chosen datum point or line and rectangular coordinate dimensions shall be employed. It is preferable to choose the datum point or line on the actual part to assist inspection, since an external datum necessitates a special set up.

7 DATA SHEETS

7.1 With each new type of rotorcraft, interchangeability data sheets shall be prepared and included in the Master Record Index (MRI). These sheets shall show the minimum clearance for final covering, and also tolerances upon dimensions and attachment points of all components which will be held as spare parts. The minimum allowable clearances which will ensure that no part of a movable component will foul any adjacent component shall also be shown.

7.2 Where components are further sub-divided, as is the case in large rotorcraft, the above conditions shall also apply at the sub-divisions.

7.3 If shims and bushes are employed at rigid attachments, details shall be given on the data sheets.

8 ATTACHMENT OF MAIN COMPONENTS

8.1 On main components, when rigid attachments necessitating fitted joints are essential, removable parts (e.g., shims and bushes) shall be provided to ensure that no alteration to the main component is necessary to obtain the fit required.

9 HINGES

9.1 DATUM

9.1.1 In an arrangement of hinges one hinge shall be indicated on the drawings as a datum from the centre line of which all other hinges are dimensioned. This datum hinge may be an end hinge if centres are up to 2.3m, but if centres are larger than this a hinge as nearly central as possible shall be used.

9.2 MINIMUM CLEARANCES

9.2.1 Sufficient clearance shall be provided between male and female hinges (other than the datum hinge) to allow for the assembling of each pair in all positions possible, taking adverse limits into consideration.

9.2.2 The datum hinge shall have as small a clearance as possible for location purposes and where freedom of end movement is permissible the minimum clearance of para 9.2.1 need not be provided on this hinge if the limits on the locating dimension are sufficient to cover the full end movement possible under the limits governing the remaining hinges. There must be no danger of fouling other parts of the structure, or of exceeding the minimum clearances allowed between two covered components which are hinged together.

10 PIANO HINGES

10.1 Piano hinges shall not be separated but shall be attached to one component only, leaving one half to be attached to the mating part on assembly. The drawings shall indicate to which part the hinge shall be attached and also convenient bolt holes which shall be used as datum points for assembly.

11 COWLINGS AND PANELS

11.1 Any suitable method may be employed to obtain interchangeability of cowlings, panels, etc. but consideration shall be given during the early stages of design to ensure that the pick-up point on the Rotorcraft and the pick-up point on the removable part are adequately jigged.

12 AERODYNAMIC CHARACTERISTICS

12.1 Care should be taken to minimise variations in contour of any component when change in the aerodynamic characteristics of that component will affect the performance of the Rotorcraft.

12.2 CRITICAL POSITIONING

12.2.1 When the position of one component relative to another is critical (e.g., tail cone and tail pylon) the shape and the positional dimensions of the adjacent surfaces shall be closely specified and tolerances indicated. The jigging and interchangeability arrangements, and the riveting technique, shall be designed to minimise both the risk of departures from the dimensions and the warping or distortion of such components in manufacture.

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INTERCHANGEABILITY

REFERENCE PAGE

MOD (PE) Specifications

D Eng R D 2024	..	..	Interchangeability requirements for engines and accessories

Defence Standards

05-19	..	..	Limits and Fits for engineering

CHAPTER 806

MARKING AND NOTICES

1 INTRODUCTION

1.1 The requirements of this chapter govern the markings and notices necessary to facilitate servicing.

1.2 Markings provided in accordance with the requirements of this chapter shall be shown on the appropriate drawings. Wherever possible adhesive markings shall be used.

2 AIRFRAME NOTICES (see Chapter 103 para 5)

3 RIGGING AND CG DATUM MARKS

3.1 All rotorcraft shall be provided with means for indicating the longitudinal and lateral rigging positions, and for specifying the position of the centre of gravity.

3.2 DATUM POINTS

3.2.1 A "datum point" for use as a point of origin for specifying the position of the centre of gravity shall be chosen conveniently near the cg position, but in any case shall be so positioned that a plumb bob may be dropped from it to the ground without fouling any part of the rotorcraft.

3.2.2 The datum point shall be marked on a metal plate fastened to the primary structure or by other suitable means, and the words "Datum point" permanently marked adjacent to the datum. Preferable the datum point should be positioned on the side of the fuselage.

3.2.3 Two additional datum points shall be provided externally on the side of the fuselage, such that a line joining the datum points will be horizontal when the rotorcraft is in the correct rigging position.

3.3 LONGITUDINAL AND LATERAL RIGGING POSITIONS

3.3.1 The longitudinal and lateral rigging positions shall be specified either in relation to datum points or to suitable members of the primary structure.

3.4 CG POSITION

3.4.1 The axes of reference for the cg position from the point of origin shall be:

(i) a horizontal fore and aft line through the datum point, when the rotorcraft is in the rigging position (the datum line), and

(ii) a vertical line through the datum point and normal to the datum line.

3.4.2 The coordinates of the centre of gravity shall be quoted as positive when aft of the vertical axis and when above the horizontal axis.

3.5 DRAWINGS

3.5.1 The position of the datum point and the location of the surfaces or points for obtaining the longitudinal and lateral rigging positions shall be indicated in the Air Publication Air Publication (AP) for the rotorcraft in meters or centimeters.

4 SERVICING POINTS

4.1 GENERAL

4.1.1 Servicing points shall be marked in accordance with the requirements of DEF STAN 05-18.

4.1.2 The following additional markings shall also be included.

4.2 FUEL

4.2.1 On all rotorcraft required to be pressure refuelled or defuelled, notices shall be placed near the pressure refuelling and defuelling connectors respectively stating:

(i)	Refuelling	Maximum delivery pressure:	3.4 bar
(ii)	Defuelling	Maximum suction pressure:	0.34 bar below atmospheric

5 PIPELINES

5.1 Except as stated in para 5.2, pipe lines, including electrical conduits and tubing, shall be marked in accordance with the requirements of BS Specification M23[1], but the optional colour marking is not to be applied.

5.2 Adhesive tape shall not be used for marking pipes within liquid-filled tanks. Alternative means of marking pipes need not be applied.

6 TRANSPARENT PANELS

6.1 Any markings on transparent panels shall be made in accordance with Chapter 404 para 6.

7 ELECTRICAL CONNECTIONS

7.1 IDENTIFICATION OF CIRCUITS

7.1.1 To facilitate the checking of cable runs and the testing of apparatus and circuits, adequate identification of cable runs, fuses, earth points and apparatus terminals shall be provided. All cables should be marked at frequent intervals with a separate identity for each cable to facilitate rectification and battle damage repair. In addition, each circuit shall be distinguished by the appropriate Circuit Function Letter in accordance with Defence Standard 61-6. The means employed shall be such that the identification is permanent and easily visible.

7.1.2 In rotorcraft with standard power supplies of 115/200 volts, 3 phase AC, warning notices shall be fitted in servicing bays, on fuse carriers and distribution panels. The notices shall be worded "DANGER, 200 VOLTS".

7.2 EXTERNAL CONNECTIONS

7.2.1 External electrical connections shall be marked in accordance with the requirements of DEF STAN 05-18.

8 UNDERCARRIAGE

8.1 UNDERCARRIAGE LEGS

8.1.1 Shock absorber units shall be provided with marks and/or pointers to enable ground personnel to check whether the units are in a safe working condition.

9 STRONG POINTS

9.1 The positions of all strong points provided on the rotorcraft for the purpose of ground handling shall be marked in accordance with the requirements of DEF STAN 05-18.

10 WALKWAYS

10.1 The boundaries of those parts of a rotorcraft that are designed to be used as walkways or platforms, and any areas within these boundaries that may not be walked on shall be marked in accordance with DEF STAN 05-18.

10.2 The floor surfaces of all areas which are likely to become wet in service should have slip resistant properties, wherever practical.

REFERENCES

Reference	ASCC Air Standard	STANAG
1	-	3104

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MARKINGS AND NOTICES

REFERENCE PAGE

Defence Standards

05-18	..	..	Symbol markings for servicing and safety/hazard points on aircraft
16-21	..	..	Marking of hydraulic components.
59-15	..	..	Sleeves, insulation, electrical and sleeves, identification, cable.
61-6	..	..	Aircraft electrical circuit identification.
61-7	..	..	Identification of electrical and electronic systems, wiring and components.

British Standards Specifications

M23	..	..	Identification scheme for aircraft pipe lines.
M24	..	..	Graphical symbols for aircraft hydraulic and pneumatic systems.

PART 8 APPENDIX No. 2
MAINTENANCE
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 800: GENERAL MAINTENANCE REQUIREMENTS

800	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-470	MAINTAINABILITY PROGRAM FOR SYSTEMS AND EQUIPMENT
	MIL-STD-471	MAINTAINABILITY VERIFICATION/DEMONSTRATION/EVALUATION
	MIL-STD-680	CONTRACTOR STANDARDISATION PROGRAM REQUIREMENTS
	MIL-STD-965	PARTS CONTROL PROGRAM
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-1843	RELIABILITY-CENTRED MAINTENANCE FOR AIRCRAFT, ENGINES, AND EQUIPMENT
	MIL-STD-2076	UNIT UNDER TEST, COMPATIBILITY WITH AUTOMATIC TEST EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-STD-2077	TEST PROGRAM SETS, GENERAL REQUIREMENTS FOR
	MIL-STD-2084	GENERAL REQUIREMENTS FOR MAINTAINABILITY OF AVIONIC AND ELECTRONIC SYSTEMS AND EQUIPMENT
	MIL-STD-2124	FLIGHT DATA RECORDER, FUNCTIONAL STANDARDS FOR
	MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
	MIL-STD-2173	RELIABILITY-CENTRED MAINTENANCE REQUIREMENTS FOR NAVAL AIRCRAFT, WEAPONS SYSTEMS, AND SUPPORT EQUIPMENT
	MIL-I-6870	INSPECTION PROGRAM REQUIREMENTS, NON-DESTRUCTIVE, FOR AIRCRAFT AND MISSILE MATERIALS AND PARTS
	MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
	MIL-S-8512	SUPPORT EQUIPMENT, AERONAUTICAL, SPECIAL, GENERAL SPECIFICATION FOR THE DESIGN OF
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-I-8700	INSTALLATION AND TEST OF ELECTRONIC EQUIPMENT IN AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-A-19531	AIRCRAFT: MAINTENANCE AND ENGINEERING INSPECTION REQUIREMENTS
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-T-28800	TEST EQUIPMENT FOR USE WITH ELECTRICAL AND ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR

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SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION

800 1.

2. DESIGNING FOR MAINTENANCE

800 2.

3. HANDLING

800 3.

4. ROUTINE SERVICING AND REPAIR
800 4.

5. REPLACEMENTS AND INTERCHANGEABILITY
800 5. MIL-I-8500

6. ACCESSIBILITY
800 6.

7. STORAGE
800 7.

8. CONDITION INSPECTION EQUIPMENT
800 8.

9. FAULT DIAGNOSIS
800 9.

CHAPTER 801: TRANSPORT, HANDLING AND STORAGE

801	MIL-C-104	CRATES, WOOD; LUMBER AND PLYWOOD SHEATHED, NAILED AND BOLTED
	MIL-HDBK-772	MILITARY PACKAGING CRITERIA
	MIL-STD-794	PARTS AND EQUIPMENT, PROCEDURES FOR PACKAGING
	MIL-STD-805	TOWING FITTINGS AND PROVISIONS FOR MILITARY AIRCRAFT, DESIGN REQUIREMENTS FOR
	MIL-STD-809	ADAPTER, AIRCRAFT, JACKING POINT DESIGN AND INSTALLATION OF
	MIL-STD-1365	GENERAL DESIGN CRITERIA FOR HANDLING EQUIPMENT ASSOCIATED WITH WEAPONS AND WEAPON SYSTEMS
	MIL-STD-1366	TRANSPORTABILITY CRITERIA
	MIL-STD-1367	PACKAGE, HANDLING, STORAGE AND TRANSPORTABILITY PROGRAM REQUIREMENTS FOR SYSTEMS AND EQUIPMENT
	MIL-STD-2073	DOD MATERIAL PROCEDURES FOR DEVELOPMENT AND APPLICATION OF PACKAGING REQUIREMENTS
	MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
	MIL-S-6451	SHIELDS, PROTECTIVE, AIRCRAFT AND MISSILES
	MILM-8090	MOBILITY, TOWED AEROSPACE GROUND EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-S-8698	STRUCTURAL DESIGN REQUIREMENTS, HELICOPTERS
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-A-25175	AIR TRANSPORT, NONTACTICAL, PACKING FOR
	MIL-T-81259	TIE-DOWNS, AIRFRAME DESIGN, REQUIREMENTS FOR

CONTROLLED DISTRIBUTION:

	MIL-STD-1791	DESIGNING FOR INTERNAL AERIAL DELIVERY IN FIXED WING AIRCRAFT
	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

1. TRANSPORT OF COMPONENTS

801 1. MIL-STD-1366
MIL-A-25175
MIL-T-81259

2. GROUND HANDLING

801 2.

3. JACKING

801 3. MIL-STD-809

4. SLINGING

801 4.

5. GROUND TOWING

801 5. MIL-STD-805
MIL-M-8090

REFERENCES

TABLE 1: SIZE CATEGORIES FOR ROTORCRAFT

TABLE 2: SIZE CATEGORIES FOR PACKED COMPONENTS

CHAPTER 802: ROUTINE SERVICING

802	MIL-STD-210	CLIMATIC INFORMATION TO DETERMINE DESIGN AND TEST REQUIREMENTS FOR MILITARY SYSTEMS AND EQUIPMENT
	MIL-HDBK-275	GUIDE FOR SELECTION OF LUBRICANTS, FLUIDS AND COMPOUNDS FOR USE IN FLIGHT VEHICLES AND COMPONENTS
	MIL-STD-470	MAINTAINABILITY PROGRAM FOR SYSTEMS AND EQUIPMENT
	MIL-STD-471	MAINTAINABILITY VERIFICATION/DEMONSTRATION/EVALUATION
	MIL-STD-838	LUBRICATION OF MILITARY EQUIPMENT
	MIL-STD-872	TEST REQUIREMENTS AND PROCEDURES FOR AIRCRAFT EMERGENCY GROUND AND DITCHING ESCAPE PROVISIONS
	MIL-STD-1289	AIRBORNE STORES, GROUND FIT AND COMPATIBILITY TESTS FOR
	MIL-STD-1365	GENERAL DESIGN CRITERIA FOR HANDLING EQUIPMENT ASSOCIATED WITH WEAPONS AND WEAPON SYSTEMS
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-2076	UNIT UNDER TEST, COMPATIBILITY WITH AUTOMATIC TEST EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-STD-2084	MAINTAINABILITY OF AVIONIC AND ELECTRONIC SYSTEMS AND EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-F-3541	FITTINGS, LUBRICATION, GENERAL SPECIFICATION FOR
	MIL-T-5522	TEST REQUIREMENTS AND METHODS FOR AIRCRAFT HYDRAULIC AND EMERGENCY PNEUMATIC SYSTEMS
	MIL-S-6625	SPRAY EQUIPMENT, AIRCRAFT WINDSHIELD ANTI-ICING
	MIL-T-7743	TESTING, STORE SUSPENSION AND RELEASE EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-F-7872	FIRE AND OVERHEAT WARNING SYSTEMS, CONTINUOUS, AIRCRAFT: TEST AND INSTALLATION OF
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-I-8671	INSTALLATION OF DROPPABLE STORES AND ASSOCIATED RELEASE SYSTEMS
	MIL-I-8700	INSTALLATION AND TEST OF ELECTRONIC EQUIPMENT IN AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-A-19531	AIRCRAFT: MAINTENANCE AND ENGINEERING INSPECTION REQUIREMENTS
	MIL-O-19838	OIL SYSTEMS, AIRCRAFT, INSTALLATION AND TEST OF
	MIL-E-22285	EXTINGUISHING SYSTEM, FIRE, AIRCRAFT, HIGH-RATE-DISCHARGE TYPE, INSTALLATION AND TEST OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-23447	FIRE WARNING SYSTEMS, AIRCRAFT, RADIATION SENSING TYPE, TEST AND INSTALLATION OF
	MIL-C-25427	COUPLING ASSEMBLY, HYDRAULIC, SELF-SEALING, QUICK DISCONNECT
	MIL-P-26292	PITOT AND STATIC PRESSURE SYSTEMS, INSTALLATION AND INSPECTION OF
	MIL-T-28800	TEST EQUIPMENT FOR USE WITH ELECTRICAL AND ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-C-38373	CAP, FLUID TANK FILLER
	MIL-R-81589	RAIN REPELLENT FLUID APPLICATION SYSTEM, AIRCRAFT WINDSHIELD

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1. GENERAL

802 1.

2. INSPECTION

802 2.

3. LUBRICATION

802 3 MIL-HDBK-275
MIL-STD-838
MIL-F-3641

4. GASEOUS SYSTEMS

802 4.

REFERENCES

CHAPTER 803: REPAIRS

803	MIL-HDBK-337	ADHESIVE BONDED AEROSPACE STRUCTURE REPAIR
	MIL-STD-470	MAINTAINABILITY PROGRAM FOR SYSTEMS AND EQUIPMENT
	MIL-STD-471	MAINTAINABILITY VERIFICATION/DEMONSTRATION/EVALUATION
	MIL-HDBK-472	MAINTAINABILITY PREDICTION
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-I-6870	INSPECTION PROGRAM REQUIREMENTS, NON-DESTRUCTIVE, FOR AIRCRAFT AND MISSILE MATERIALS AND PARTS
	MIL-A-8867	AIRPLANE STRENGTH AND RIGIDITY, GROUND TESTS
	MIL-P-9400	PLASTIC LAMINATE AND SANDWICH CONSTRUCTION PARTS AND ASSEMBLY, AIRCRAFT STRUCTURAL, PROCESS SPECIFICATION REQUIREMENTS
	MIL-A-83377	ADHESIVE BONDING (STRUCTURAL) FOR AEROSPACE AND OTHER SYSTEM REQUIREMENTS FOR

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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804	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-470	MAINTAINABILITY PROGRAM FOR SYSTEMS AND EQUIPMENT
	MIL-STD-471	MAINTAINABILITY VERIFICATION/DEMONSTRATION/EVALUATION
	MIL-STD-783	LEGENDS FOR USE IN AIRCREW STATIONS AND ON AIRBORNE EQUIPMENT
	MIL-STD-965	PARTS CONTROL PROGRAM
	MIL-STD-1247	MARKINGS, FUNCTIONS AND HAZARD DESIGNATIONS OF HOSE, PIPE, AND TUBE LINES FOR AIRCRAFT, MISSILE, AND SPACE SYSTEMS
	MIL-STD-1353	ELECTRICAL CONNECTORS, PLUG-IN SOCKETS AND ASSOCIATED HARDWARE, SELECTION AND USE OF
	MIL-STD-1367	PACKAGE, HANDLING, STORAGE AND TRANSPORTABILITY PROGRAM REQUIREMENTS FOR SYSTEMS AND EQUIPMENT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-1788	AVIONICS INTERFACE DESIGN STANDARD
	MIL-STD-2076	UNIT UNDER TEST, COMPATIBILITY WITH AUTOMATIC TEST EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-STD-2084	MAINTAINABILITY OF AVIONIC AND ELECTRONIC SYSTEMS AND EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-W-5088	WIRING, AEROSPACE VEHICLE
	MIL-H-5440	HYDRAULIC SYSTEMS, AIRCRAFT, TYPES I AND II, DESIGN AND INSTALLATION REQUIREMENTS FOR
	MIL-I-6115	INSTRUMENT SYSTEMS, PITOT TUBE AND FLUSH STATIC PORT OPERATED, INSTALLATION OF
	MIL-I-7028	INSTRUMENTS AND INSTRUMENT BOARDS, AIRCRAFT, INSTALLATION OF
	MIL-E-7080	ELECTRIC EQUIPMENT, AIRCRAFT, SELECTION AND INSTALLATION OF
	MIL-C-7413	COUPLINGS, QUICK DISCONNECT, AUTOMATIC SHUTOFF, GENERAL SPECIFICATION FOR
	MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-1-8700	INSTALLATION AND TEST OF ELECTRONIC EQUIPMENT IN AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-H-8775	HYDRAULIC SYSTEM COMPONENTS, AIRCRAFT AND MISSILES, GENERAL SPECIFICATION FOR
	MIL-S-8812	STEERING SYSTEM, AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-H-8891	HYDRAULIC SYSTEMS, MANNED FLIGHT VEHICLES, TYPE III, DESIGN, INSTALLATION AND DATA REQUIREMENTS FOR, GENERAL SPECIFICATION FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS-DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-I-18259	INSTALLATION OF WINDOW ANTI-ICING, DEGREASING AND WASHING SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-H-18325	HEATING AND VENTILATING SYSTEMS, AIRCRAFT, GENERAL SPECIFICATION FOR
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	MIL-I-18802	FUEL AND OIL LINES, AIRCRAFT, INSTALLATION OF
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	MIL-O-19838	OIL SYSTEMS, AIRCRAFT, INSTALLATION AND TEST OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-C-25427	COUPLING ASSEMBLY, HYDRAULIC, SELF-SEALING, QUICK DISCONNECT
	MIL-E-25499	ELECTRICAL SYSTEMS, AIRCRAFT, DESIGN AND INSTALLATION OF, GENERAL SPECIFICATION FOR
	MIL-P-26292	PITOT AND STATIC PRESSURE SYSTEMS, INSTALLATION AND INSPECTION OF
	MIL-B-81365	BLEED AIR SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-C-83413	CONNECTORS AND ASSEMBLIES, ELECTRICAL, AIRCRAFT GROUNDING, GENERAL SPECIFICATION FOR

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MIL-L-87139	LANDING GEAR SYSTEMS
MIL-P-87210	PNEUMATIC POWER SYSTEMS, HIGH PRESSURE
SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT

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2. ENGINE INSTALLATION

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3. PIPELINES

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4. INSTRUMENTS

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5. CONTROL CABLES OR RODS

804 5. MIL-I-9490

6. ELECTRICAL AND AVIONIC INSTALLATION

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	MIL-STD-280	DEFINITIONS OF ITEM LEVELS, ITEM EXCHANGEABILITY, MODELS AND RELATED TERMS
	MIL-STD-680	CONTRACTOR STANDARDISATION PROGRAM REQUIREMENTS
	MIL-STD-965	PARTS CONTROL PROGRAM
	MIL-STD-970	STANDARDS AND SPECIFICATIONS, ORDER OF PREFERENCE FOR THE SELECTION OF
	MIL-STD-973	CONFIGURATION MANAGEMENT
	MIL-T-1000	TECHNICAL DATA PACKAGES, GENERAL SPECIFICATION FOR
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-STD-1476	METRIC SYSTEM, APPLICATION IN NEW DESIGN
	MIL-STD-2084	MAINTAINABILITY OF AVIONIC AND ELECTRONIC SYSTEMS AND EQUIPMENT, GENERAL REQUIREMENTS FOR
	MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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MIL-STD-1285	MARKING OF ELECTRICAL AND ELECTRONIC PARTS
MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
MIL-STD-1353	ELECTRICAL CONNECTORS, PLUG-IN SOCKETS AND ASSOCIATED HARDWARE, SELECTION AND USE OF
MIL-STD-2084	GENERAL REQUIREMENTS FOR MAINTAINABILITY OF AVIONIC AND ELECTRONIC SYSTEMS AND EQUIPMENT
MIL-STD-2161	PAINT SCHEMES AND EXTERIOR MARKINGS FOR U.S. NAVY AND MARINE CORPS AIRCRAFT
MIL-W-5044	WALKWAY COMPOUND, NONSLIP AND WALKWAY MATTING, NONSLIP
MIL-W-5088	WIRING, AEROSPACE VEHICLE
MIL-P-6997	PLASTIC; WORKING AND INSTALLATION OF TRANSPARENT SHEET, GENERAL SPECIFICATION FOR
MIL-E-7080	ELECTRIC EQUIPMENT, AIRCRAFT, SELECTION AND INSTALLATION OF
MIL-I-8500	INTERCHANGEABILITY AND REPLACEABILITY OF COMPONENT PARTS FOR AEROSPACE VEHICLES
MIL-E-8593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
MIL-M-13231	MARKING OF ELECTRONIC ITEMS
MIL-N-18307	NOMENCLATURE AND NAMEPLATES FOR AERONAUTICAL ELECTRONIC AND ASSOCIATED EQUIPMENT
MIL-I-18802	FUEL AND OIL LINES, AIRCRAFT, INSTALLATION OF
MIL-E-21981	ELECTRONICS EQUIPMENT, NOMENCLATURE, SERIAL NUMBERS AND IDENTIFICATION PLATES: REQUIREMENTS FOR
MIL-E-25499	ELECTRICAL SYSTEMS, AIRCRAFT, DESIGN AND INSTALLATION OF, GENERAL SPECIFICATION FOR
MIL-C-83413	CONNECTORS AND ASSEMBLIES, ELECTRICAL, AIRCRAFT GROUNDING, GENERAL SPECIFICATION FOR

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SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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2. AIRFRAME NOTICES
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3. RIGGING AND CG DATUM MARKS
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4. SERVICING POINTS
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5. PIPELINES
806 5. MIL-STD-101
MIL-STD-1247

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MIL-E-7080

8. UNDERCARRIAGE
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9. STRONG POINTS
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10. WALKWAYS
806 10. MIL-W-5044

REFERENCES

PART 9

FLIGHT TESTS - HANDLING

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APPENDIX No 1 **FLIGHT TEST - HANDLING FOR MILITARY DERIVATIVE OF CIVIL ROTORCRAFT***
(Note: See relevant para of this Appendix for military derivative requirements relating to particular chapters of Part 9)

APPENDIX No 2 **U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS**

* In Preparation

CHAPTER 900

GENERAL HANDLING FLIGHT TEST REQUIREMENTS

1 INTRODUCTION

1.1 This Part states those tests which shall be made to demonstrate that the rotorcraft can be controlled satisfactorily and is adequately stable in the air, and on the ground, in accordance with the requirements of Part 6.

1.2 Each chapter of this Part contains tests of the rotorcraft under a specific flight condition and is arranged in accordance with a standard layout, the paragraphs being grouped under headings in the following sequence:

(i) Object;

(ii) Relevant design requirements;

(iii) Applicability;

(iv) Equipment;

(v) Loading;

(vi) General test conditions;

(vii) Tests;

(viii) Notes.

2 RELEVANT DESIGN REQUIREMENTS

2.1 Each chapter of this Part repeats for the convenience of flight test personnel the design requirements associated with the flight tests of that chapter. Such requirements are repeated for information only; for design purposes the full requirements must be consulted.

3 APPLICABILITY

3.1 The tests contained in this Part apply to all new types of rotorcraft.

3.2 The tests are to be made in full on a representative sample of the rotorcraft. The tests shall be repeated on other samples of the rotorcraft as agreed between the Rotorcraft Project Director and the Contractor to ensure that variations between individual rotorcraft in the fleet do not adversely affect the demonstrated handling characteristics. However, when more than one loading condition is stipulated, the repeat tests need only be made at the loading at which the behaviour was least satisfactory. When modifications liable to affect the aerodynamic characteristics of the rotorcraft are incorporated, the first rotorcraft so modified shall be tested at the most adverse loadings. In addition, tests at other loadings shall be made, at the discretion of the Rotorcraft Project Director, to ascertain whether the modification is likely to produce more adverse effects at other loadings.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The recommendations for instrumentation for handling trials are discussed in detail in Leaflet 900/2.

4.1.2 All normal flying and engine instruments shall be fitted for the tests of this Part. In addition, particular tests require special instruments, and these are detailed in the appropriate chapters. Obtaining, fitting and calibrating all instruments shall be the Contractor's responsibility.

4.1.3 Unless dispensation is obtained from the Rotorcraft Project Director, test instrumentation capable of continuous recording shall be fitted for the tests of Part 9. Instrumentation recommended for recording data relevant to a particular test is given in para 4 of each subsequent Chapter of this Part. Para 4 specifies these instrumentation requirements by reference to items listed in Tables 1 and 2 of Leaflet 900/2 'Recommended Test Instrumentation' and 'Telemetry Parameters' respectively. Telemetry parameters, as appropriate, are recommended for flight safety reasons and not necessarily to improve trials efficiency.

4.1.4 For some trials, particularly those which constitute a significant flight safety hazard or those where real time analysis would reduce the required flight time significantly (e.g., structural loads during A.F.C.S. runaway trials), the use of telemetry shall also be considered.

4.1.5 An Accident Data Recorder (ADR) shall be fitted in accordance with DEF STAN 05-123 Chapter 240.

4.2 MISCELLANEOUS EQUIPMENT

4.2.1 Tests involving planned flight over water and in particular prolonged hover (e.g., A.F.C.S./sonar coupled modes) shall require operative flotation gear, if standard to the rotorcraft. Such tests on rotorcraft which do not have flotation gear as part of their standard build may only be made following consultation with the Rotorcraft Project Director.

4.2.2 All external equipment likely to affect the aerodynamic characteristics of the rotorcraft shall be either fitted or fully represented by a flightworthy mock-up.

4.2.3 All equipment and systems associated with the satisfactory operation of the rotorcraft in all the flight conditions anticipated for the trial shall be fully operational before the trial commences.

4.2.4 In the interests of safety whenever flight tests of rotorcraft fitted with dummy bombs or other stores which must not be released in flight are required, all electrical wires leading to the carrier release slip, or similar device, shall be disconnected and any cables or rods for operating the mechanical release mechanism or mechanisms, if fitted, shall also be disconnected. However, when carrying stores which may be released safely in flight a jettison system should be fitted.

5 LOADING

5.1 The required loadings are stated in each chapter and are given either in full or as one of the following standard loadings:

(i)	W_{max}	Maximum permissible all up weight or mass at take off C of G to be specified.
(ii)	$W_{max/aft}$	Maximum permissible all up weight or mass at take off with maximum aft C of G.
		The above suitably annotated would be applicable forward (fwd)., left and right.
(iii)	X	The full service loading.
(iv)	X1	The maximum mass with full operational equipment and the furthest forward C of G position at this mass.
(v)	X2	The maximum mass with full operational equipment and the furthest aft C of G position at this mass.
(vi)	X3	The maximum mass with full operational equipment and the furthest left lateral C of G position at this mass.
(vii)	X4	The maximum mass with full operational equipment and the furthest right lateral C of G position at this mass.
(viii)	Y	The forward loading.
(ix)	Y1	That practical loading giving the furthest forward C of G position and the greatest mass obtainable at this C of G, position.
(x)	Y2	That practical loading giving the furthest forward C of G position and the minimum mass obtainable at this C of G position.
(xi)	Z	The aft loading.
(xii)	Z1	That practical loading giving the furthest aft C of G position and the greatest mass obtainable at this C of G position.
(xiii)	Z2	That practical loading giving the furthest aft C of G position and the minimum mass obtainable at this C of G position.

When more than one operational role is specified the most adverse of the above standard loadings shall be used.

5.2 Since the above loading conditions represent limiting cases, it is important that each test, and each part of each test, be made at the loading condition specified. Due allowance may be made for fuel consumed while the rotorcraft is climbing to the test altitude but if subsequent consumption of fuel causes a material change in C of G position, provision shall be made to enable the load carried to be redistributed to compensate for this.

5.3 The mass and C of G position at which each test was made shall be reported. The centre of gravity being quoted in millimetres, in three dimensions, in Cartesian coordinates relative to any convenient set of mutually perpendicular axes intersecting at or near the centre of gravity. In each case the positive directions shall be aft of and upward from the origin.

6 GENERAL TEST CONDITIONS

6.1 The conditions at which the tests shall be made are specified in each chapter. For greater clarity and convenience, test conditions common to a number of tests of a particular chapter have been combined under a separate heading. The test report shall contain a full description of the conditions of each test.

6.2 TERMS, ABBREVIATIONS AND SYMBOLS USED IN PART 9

6.2.1 Controls

(i) Collective Pitch Control. Applies pitch to all main rotor blades evenly and with the appropriate power application, either through a manual throttle or via a coupled power control system, will cause the rotorcraft to climb. The control to operate in its logical sense i.e., upward application of the control lever will cause the rotorcraft to climb.

(ii) Cyclic Pitch Control. Applies pitch cyclically to the main rotor blades to effectively tilt the rotor disc in the desired direction.

Forward displacement of the cyclic control will cause the rotorcraft to pitch nose down and/or produce forward movement. Displacement of the cyclic control to the right will cause the rotorcraft to bank right hand down and/or produce movement in a starboard direction. It follows that aft or left displacement will respond in the appropriate direction.

(iii) Yaw Control. In the case of rotorcraft, directional control may be achieved by varying means. The most common being the variation of pitch to the tail rotor blades. The tail rotor also serves to compensate for main rotor shaft torque and as such has a complexity not normally experienced in aeroplanes. Multi rotor machines may achieve directional control by differential collective pitch on main rotors, and compound rotorcraft or convertiplanes may use asymmetric power or conventional flying controls or combinations of any of the above. Directional control shall be achieved by means of rudder pedals or rudder bar and may be expected to operate in a similar fashion to that accepted for aeroplanes and compound rotorcraft or convertiplanes which utilise rotary wing technology for any part of their flight regime may use asymmetric power or conventional flying controls or combinations of the above to achieve yaw control.

(iv) The pitch and roll control will generally be combined in a conventional control column (single lever type) or control column (wheel type). The yaw control will normally be the rudder bar or the rudder pedals. Power demand from which the rotorcraft will derive its main source of lift will be controlled by means of a collective lever. It is to such systems that the detailed requirements of Part 6 relate.

(v) Where a manufacturer elects to use some other system such as a miniaturised, side-located controller, the control travels and the force levels will, in general, differ from those specified and should be agreed with the Rotorcraft Project Director.

6.2.2 Symbols and Abbreviations

(i) VD ... knots (I.A.S.) the maximum design diving speed.

(ii) VNe ... knots (I.A.S.) the never exceed speed for test purposes speeds in excess of VNe may be specified. For example 10% above VNe will be expressed as 1.1 VNe.

(iii) VNo ... knots (I.A.S) the normal operating limit speed.

(iv) Vy ... knots (I.A.S.) speed for best rate of climb.

(v) M.P.O.G. - minimum pitch (collective) on ground.

(vi) Hover I.G.E. - hover in ground effect.

(vii) Hover O.G.E. - hover out of ground effect.

For the purposes of the tests of this Part, estimated values of the above speeds may be used.

7 TESTS

7.1 All the tests applicable shall be made by the Contractor to the satisfaction of the Rotorcraft Project Director before the rotorcraft is submitted for flight tests at an Experimental Establishment unless written instructions to the contrary are received.

7.2 Before the rotorcraft is submitted for flight tests at an Experimental Establishment the Contractor shall certify that the rotorcraft is safe to be flown by authorised Service pilots and specify the limitations to be observed. (See Leaflet 900/1).

7.3 In many cases it will be necessary in the interests of safety to make preliminary tests at less severe conditions than those prescribed in the following chapters. This is a matter for the discretion of the Contractor and consequently no reference to preliminary tests is made in this Part.

7.4 The tests presented in this Part constitute the major (but not necessarily the whole) basis of handling assessment. While pilot opinion forms one of the principal means of assessing rotorcraft flying qualities, the quantitative results obtained during the handling tests of this Part are necessary to amplify that opinion, and to draw comparisons with the relevant design requirements. To ensure consistent expression and interpretation of pilot opinion, full use shall be made of the revised Cooper-Harper Handling Qualities Rating Scale (see Leaflet 900/3).

7.5 CHOICE OF SERVICE LIMITING SPEED

7.5.1 Operating speed limitations shall be chosen to safeguard the rotorcraft in all areas of operation from inadvertently exceeding any of the criteria which may become the limiting factor, i.e.:

(i) Vibratory stress levels to ensure a useful service life.

(ii) Ultimate stress levels to ensure structural integrity.

(iii) Adequate and safe control characteristics.

7.5.2 Operating speed limitations for VNe and VNo shall be selected as follows:

(i) Never Exceed Speed, VNe. This speed shall be such that the probability of inadvertently exceeding the maximum Design Speed, VD, is remote. This speed shall be chosen to provide the required margin for all possible rotorcraft characteristics and in all likely flight conditions. In the absence of an investigation substantiating another value, VNe shall not exceed 0.9 times VD.

(ii) Normal Operating Limit Speed, VNo. This speed shall be established for all rotorcraft to be sufficiently below VNe to ensure that the probability of exceeding VNe in a moderate upset occurring at VNo is not more than Reasonably Probable. In the absence of an investigation substantiating another value VNo shall not exceed 0.9 V Ne.

(iii) Manoeuvre margins. The flight envelope for VNo and VNe shall be qualified with clear indication of the manoeuvre margin available. The limiting speeds shall be supplemented by a criteria for a sustained roll angle. For normal operations, this should be 30° but at limiting altitudes and VNe, may be reduced to 20°. It shall be demonstrated that there is sufficient control margin available to recover from disturbances which could temporarily exceed the roll angle limits.

7.6 CHOICE OF LIMITING ACCELERATION

7.6.1 With most rotorcraft the probability of imposing excessive positive or negative acceleration in normal operation is relatively low and it is not normal practice to install a visual reading accelerometer. Weapons, roles and low flying regimes may however introduce regular excursions of high acceleration reversals which although not presenting any great risk of structural failure does make an appreciable inroad into the fatigue life of structure and rotating components. Testing shall demonstrate by typical role usage that such excursions can be contained within the normal usage envelope without cockpit reference or a speed/manoeuvre limitation must be declared. Alternatively, a visual reading accelerometer shall be provided although for low level flying this may require head up display or audio warning.

7.6.2 Because of the wide variety of roles which rotorcraft can be called upon to operate, it may follow that a change of usage may alter the established fatigue lives of certain components. It is important that testing covers all major role applications and that limitations arising from such testing are agreed with the Rotorcraft Project Director.

7.7 ROTOR SPEED LIMITATIONS

7.7.1 The maximum and minimum rotor rotational speeds shall be established and a normal operating range declared over which the rotor can be expected to supply lift and control without the necessity for pilot attention beyond normally acceptable cockpit surveillance. There shall within this normal operating range remain sufficient margin to allow excursions towards a Design minimum or maximum rotational rotor speed without risk to controllability or structural integrity.

LEAFLET 900/1

GENERAL HANDLING FLIGHT TEST REQUIREMENTS

FLYING LIMITATIONS

1 INTRODUCTION

1.1 This Leaflet indicates the scope of flying limitations and places on record the methods by which they are established. This Leaflet is for general information and guidance.

2 DEFINITION OF FLYING LIMITATIONS

2.1 Flying limitations are issued in Certificates for Flight Trials (see AvP 25, Chapter 116), C.A. Clearances, Aircrew Manual and Operating Data Manuals. They define the maximum or minimum conditions under which a pilot may fly the rotorcraft so that acceptable margins of safety are maintained over dangerous flight conditions. Limitations should be stated for any characteristic of the rotorcraft or its equipment which is under the control of the pilot and which may affect the safety of the rotorcraft (e.g., flight speed, normal acceleration, engine r.p.m., rotor r.p.m., jet pipe temperature) other operating factors which, when considered in conjunction with the characteristics of the rotorcraft, may lead to flying limitations, should be taken into account (e.g. , take off surface, helipad length, carrier equipment). There are two definite categories of limitations as follows:

(i) Limitations to be observed by Contractors and M.O.D. (P.E), Test Pilots whose ultimate objective is to clear the rotorcraft as safe and satisfactory up to the structural limits or limits beyond which handling difficulties may occur.

(ii) Limitations to be observed by Service pilots in the operational use of the rotorcraft. (It is emphasised that these limitations are appropriate to typical Service pilots and to typical Service flying, and presupposes that the pilot will not be able to concentrate exclusively on the declared limitations). See para 3.3.2.

Limitations for the engines are issued by the M.O.D.(P.E.) (DG Eng) in the operating Data for the engine concerned. There may be rotor r.p.m or torque limitations on rotorcraft. Sometimes special limitations are needed because of the engine installation in the rotorcraft or engine accessories.

2.2 For a Service derivative of a civil rotorcraft, the flying limitations are based on the civil procedures as laid down in the British Civil Airworthiness Requirements (B.C.A.R.) or Joint Airworthiness Requirements (J.A.R.) and account should be taken of the limitations specified in the Flight Manual for the basic civil version of the rotorcraft.

2.3 During the life of a type of rotorcraft, experience for example accidents, may show that certain limitations require to be made temporarily more severe and that is done by imposing "temporary restrictions" issued in Special Flying Instructions (S.F.I.'s). If a longer term restriction is necessary eventually, the flying limitations will be amended accordingly.

3 BASIS FOR FLYING LIMITATIONS

3.1 GENERAL

3.1.1 As a basis for deciding the flying limitations to be stated, consideration should be given to the general conditions listed below and to any special conditions which may apply to the particular rotorcraft. It is essential however to state the minimum possible number of limitations consistent with safety as the pilot will be fully occupied with his normal operational duties; whenever practicable the limitations should be shown on the associated instruments. The limitations should be based on calculations confirmed whenever possible by ground tests and in later stages by flight tests; full consideration should be given to the possible operational configuration of the rotorcraft. The basis for the principal flying limitations is as follows:

(i)	Strength:	Limiting speeds and normal accelerations in clear and turbulent weather associated with certain masses, load distributions and C of G positions. Limitations during take-off and landing. Limitations in low altitude flying due to fatigue resulting from gusts.
(ii)	Stiffness of structure and control systems:	Limitations on speed and normal acceleration to avoid excessive vibration, divergence or undesirable control characteristics.
(iii)	Handling qualities:	Limitations to avoid excessive vibration, blade movement or blade stall. Co-ordination during entry to and recovery from turns. Effect of C of G position, control margins. Weight limitations. Handling during engine off landings including collective lever delay following engine failure. Autorotation. Helipad take off and landing techniques. Yaw control in low speed manoeuvres. Deck landing techniques. Low speed height/velocity limitations.
(iv)	Behaviour of armament and other equipment:	Limitations due to strength and handling with equipment in operational position (e.g., bomb doors open, carriage of external stores, etc.). Limitations on speeds and attitude to ensure adequate clearance between paths of stores and parts of the rotorcraft in any flight regime relevant to the weapon or stores usage.

(v)	Aircrew equipment:	Conditions for the safe use of oxygen equipment, cabin conditioning equipment, etc. Limitations on escape procedures.
(vi)	Airfield operation:	Strength - (see Part 3).
(vii)	Ship operation:	Limitations imposed by deck or helipad operation.
(viii)	System runaway or failure characteristics which could lead to an irrecoverable flight condition or damaging load.	

3.2 DEVELOPMENT ROTORCRAFT

3.2.1 For rotorcraft undergoing Contractors flight trials, the flying limitations are stated in the "Certificate for Flight Trials" prepared under the procedure of AvP 25, Chapter 116. In the early stages such limitations are inevitably based on calculations and on the result of ground testing. In this connection it is important to obtain structural and static resonance clearance (bonk checks) before flying commences or, failing this, to restrict flying to medium speeds and mild manoeuvres and as an additional safeguard; flying under these conditions should only be undertaken by experienced test pilots.

3.2.2 The flying limitations to be observed prior to completion of structural tests should ensure that the loads applied to the structure do not exceed 80% of the unfactored limit load for rotorcraft which resemble a rotorcraft on which there is previous experience. In other cases, a lower figure may be desirable after discussion with the Rotorcraft Project Director.

3.2.3 If during laboratory tests, a premature failure occurs at loads below 80% of the required ultimate load, then the permissible flying limitations should at once be altered in conformity.

3.3 SERVICE ROTORCRAFT

3.3.1 In establishing the flying limitations for a particular rotorcraft, it is necessary to have available:

(i) A knowledge of the intended operational duties of the rotorcraft.

(ii) A statement of the specified design conditions and the respects, if any, in which the rotorcraft fails to meet design requirements.

(iii) The official testing establishment's report on flight tests or, if that is not available,

(iv) The Contractor's report on flight tests.

Note: With regard to (i) it should be borne in mind that the requirements of Chapter 600 (to be issued) define the various flight envelopes for which limitations are required. The pilot may thus have different limitations for the various flight envelopes and failure states.

3.3.2 The cardinal principle which must be observed in choosing a particular limitation is that it should be consistent with the ability of the Service pilot to observe the limitation. In the light of this, the limitation, unless otherwise stated, should be defined as a NORMAL OPERATING limitation such that it:

(i) Can be reached as often as the pilot's task requires without risk, and,

(ii) Can be exceeded occasionally by a small margin without untoward consequences.

3.3.3 In some cases at the discretion of the Rotorcraft Project Director, a second and higher limitation may be quoted in addition to the NORMAL OPERATING limitation. This represents the full design flight case or the highest figure for which all aspects have been investigated in flight or on the ground, and found to be satisfactory. This limitation is never to be exceeded because consequential effects beyond this level are either hazardous or unknown.

4 SAFETY MARGIN

4.1 No general rule can be laid down for each margin of safety to be provided as it will depend on the type of limitation, the probability of the stated margin being exceeded and the consequences if it is exceeded.

4.2 When a rotorcraft is designed for a specific region of flight, it may be necessary to impose strict limitations on its use outside the region other than the more obvious ones of speed, acceleration, vibration etc. For example, a transport rotorcraft might be liable to fatigue troubles if it were flown for long periods at low altitudes due to the effects of increased turbulence and manoeuvres for which it was not designed.

4.3 The major safety limitations associated with any rotorcraft entering Service are normally the limiting Service speed due to strength or vibration considerations and the normal acceleration which may be applied in cyclic, collective and yaw control usage conditions. Their importance justifies detail consideration and this is given in para 5 and 6.

4.4 With regard to the remaining structural limitations (e.g., alighting gear lowering, bomb door opening, etc.), these depend greatly on the design conditions. In general, when it is relatively easy for the pilot to comply with the limitations, the rotorcraft may be cleared up to the appropriate limit conditions. This of course assumes that there is no need to impose a stricter limit due to unexpected troubles due, for example to vibration when the margin imposed would depend on the merits of the case.

4.5 Limitations governed by strength should be based on the strength finally achieved in the design of the rotorcraft as released to the Service, rather than on the specified strength. Thus advantage may be taken of any excess of strength above that specified to provide greater freedom of operational use, or restrictions may be imposed where the rotorcraft fails to achieve the specified strength in all respects.

4.6 Engine limitations will depend on the clearance conditions of the engine Type Record and the extent to which engine or rotor r.p.m., etc., is maintained automatically. The rotorcraft's unique capability to hover for prolonged periods or undertake sideways or rearward flight may lead to temperature limitations in engine bays or jet pipe due to the absence of forced cooling; no general guidance can be given here. It also follows that during low speed manoeuvres such as sideways and rearward flight the rotorcraft may well be operating outside any height velocity clearance which could apply for the forward flight envelope and this may result in a limitation and a level of safety appropriate to that limitation shall be discussed with the Rotorcraft Project Director.

4.7 It will be necessary to specify maximum masses for take-off and landing and it may be necessary to lay down the order of release of stores, disposition of load, use of fuel, etc., to maintain the C of G within specified limits, or to maintain the required safety margin and minimise fatigue life consumption.

4.8 Rotorcraft operations frequently entail movement to and from helipads and platforms in confined spaces or on rigs in open water. In such cases attention has to be paid to the rotorcraft's capability to safely return to the take off point in the event of engine failure or alternatively to fly away with an engine failed. The formulation of a safe take off and approach technique and the definition of the vital technique and the definition of the vital decision points will be an important part of the test programme. Consideration should be given to the effect of landing site altitude, temperature and humidity.

4.9 The design of some rotorcraft may have been based on British Civil Airworthiness Requirements which specify that the Flight Manual for the rotorcraft shall state the manoeuvring speeds and their significance i.e., full applications of tail rotor or cyclic control should be confined to speeds below certain values and these should be included in the limitations prepared under this Leaflet. (Later transport rotorcraft may be based on Joint Airworthiness Requirements).

LEAFLET 900/2

GENERAL HANDLING FLIGHT TEST REQUIREMENTS

TEST INSTRUMENTATION AND TELEMETRY PARAMETERS

1 INTRODUCTION

1.1 The object of this leaflet is to provide guidance with respect to the instrumentation required or recommended for flight tests concerned with handling and flying qualities. In general because handling trials are concerned with dynamic parameters which change very rapidly the instrumentation will involve a large volume of on-board recording. At the same time there is a very real need for good visual reference instruments for the pilot to ensure that manoeuvres can be properly and consistently executed and to give good data upon which qualitative assessments may be based. It must be appreciated that the science of data gathering and retrieval is constantly advancing and contractors will be encouraged to take full advantage of the state of the art.

2 ON BOARD RECORDING

2.1 Accurate records of the main dynamic parameters will be called for from all flight handling tests, these should normally utilise digital magnetic data acquisition systems which lend themselves to computerised analysis. An acceptable alternative would be to use an R.F. telemetry link to a ground based magnetic recording system, providing line of sight can be maintained at all times. The extent or complexity of the instrumentation fit will be dependent upon the knowledge of the rotorcraft's handling characteristics and its state of development. The total requirement for instrumentation should always be the subject of advance discussions with the Rotorcraft Project Director.

3 TELEMETRY

3.1 The use of telemetry for any handling tests of an exploratory nature should be encouraged, for the confidence it gives the crew during excursions into untested areas of the flight envelope and for the way in which the test programme can be advanced rapidly with the minimum number of sorties without compromising flight safety. When coupled with real time analysis, telemetry offers the advantage of allowing thorough probing in problem areas and a unique involvement for specialists while tests are in progress. The overall accuracy of the telemetry should be better than ±5%.

3.2 It should be noted that one of the difficulties encountered with telemetry operations is that the primary radio link may be taken away from Air Traffic Control.

Telemetry facilities should include provision for:

(i) Provision for Air Traffic Control to monitor all radio traffic on the telemetry facility.

(ii) A reliable and instant alert line with Air Traffic Control for emergency.

(iii) A pre-determined procedure whereby Air Traffic Control can perform their function in emergency situations should be laid down. This should include duplex radio facilities so that specialist advice from the telemetry personnel can continue to be available or on call.

4 VIDEO

4.1 Video cameras offer a number of advantages when used to provide a visual record of flight path occurrences and can include a speech channel. Results are immediately available for replay and only minimal technical skill is necessary for efficient operation. The obvious pictorial use of video can be supplemented to advantage by the inclusion of inset data in digital form e.g., timebase, temperature or wind velocity.

5 COCKPIT VISUAL DISPLAYS

5.1 The essential information for most handling work is of necessity available from standard cockpit instruments, but for test work there will be a firm requirement for calibrations. These may call for greater accuracy or sensitivity than that called up as routine for the production rotorcraft. Handling tests will almost certainly require additional cockpit instruments to provide information of control positions or attitude which may not be available or in acceptable form using standard instruments.

6 AUTO OBSERVER PANELS

6.1 These are generally less applicable to handling work and time consuming to analyse. They can however present useful information on slow changing parameters but it should be emphasised that the characteristics of indicators can distort data unless response and damping characteristics are carefully chosen.

7 EQUIPMENT

7.1 A chase vehicle can be used to advantage for airfield low speed handling, equipped with good meteorological instruments for wind speed and temperature and with a good speedometer accurate low speed handling tests can be achieved.

7.2 Kinetheodolites offer a very accurate method for tracking flight path for such tests as rejected take off or approach.

7.3 An alternative to kinetheodolites is available in the form of a 'C' band R.F. distance and velocity measuring system which may be ground based or airborne enabling this data to be recorded along with rotorcraft parametric data.

8 RECOMMENDED TEST INSTRUMENTATION

8.1 General procedures for determining the instrumentation requirements for both contractors and official flight tests are given in DEF-STAN 05-123 Chapter 240. This table provides typical requirements for handling and performance trials. The values quoted below are recommendations only and, depending upon the instrumentation specified and the nature of the trial, adjustments may be made at the discretion of the Rotorcraft Project

Director to the values quoted. Care should be taken not to specify greater accuracy or resolution than is necessary for the trial.

8.2 The following table relates to a similar list contained in DEF-STAN 00-970, Vol 1, Leaflet 900/2, Table 1 which lists 71 instrumentation items associated with aeroplanes, some of which are not applicable to rotorcraft. In order to maintain some degree of commonality, the item numbers for individual parameters have been left unchanged and the additional items concerned with rotorcraft introduced commencing with No. 100. (e.g., Indicated Airspeed remains Item No. 4 whereas Main Rotor Speed has been introduced as Item No. 103).

TABLE 1

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
	GENERAL						
1	Time Base	Sec	Duration of flight + ½ hr				
2	Manual Event Marker	-	-	-	-	16	With external trans-mission facility
3	Crew Speech	-	Duration of flight + ½ hr				
	FLIGHT CONDITIONS						
4	Indicated Airspeed	kt	0 to Vne +20%	Greater of 1 kt or 0.5%	0.5 kt	8	
4A	Ground Speed	kt	0 to max required	Greater of 1 kt or 0.5%	0.5 kt	1	
5	Altitude (pressure)	ft	-1000 to ceiling + 3000 ft	Greater of 20 ft or 0.5 mb	1 mb	8	Ceiling is highest altitude anticipated during test
6	Altitude (radio-altimeter)	ft	0 to 5000 ft	1	5	8	
7	Total Temperature (or Ambient Temp/ O.A.T.)	°C	-60 to +60	1	0.5	8	
8	Angle of Attack*	Deg	-30 to +30	0.25	0.1	16	
9	Pitch Attitude	Deg	±60/±10	1/0.1	0.5/ 0.05	16	
10	Roll Angle	Deg	±90/±10	1/0.1	0.5	16	
11	Sideslip Angle	Deg	±50	0.25	0.1	16	
12	Heading	Deg	0 to 360 & ±10	1 0.1	0.5/ 0.05	16	

TABLE 1 (contd)

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
13	Pitch Rate	Deg/ Sec	±100/±10	1/0.1	0.5/ 0.05	16	
14	Roll Rate	Deg/ Sec	±100/±10	1/0.1	1.5/ 0.05	16	
15	Yaw Rate	Deg/ Sec	±100/±10	1./0.1	0.5/ 0.05	16	
16	Longitudinal Acceleration	g	±1	0.01	0.001	16	All accelerometers mounted as close as possible to the CofG (See Note 4)
17	Lateral Acceleration	g	±1	0.01	0.001	16	
18	Normal Acceleration	g	-4 to +5	0.05	0.005	16	
	AIRFRAME CONFIGURATION AND STATE						
19	Flap Setting*						
20	Landing gear position	Event	Up/down	-	-	4	
21	Airbrake Position	Deg	Full	1	0.1	8	
22	Failure State	Event	-	-	-	-	See Note 1.
23	Brake Parachute*						
24	Fuel Contents	Kg/ lb	Full	20/40	10/20	1	See Note 2.
	CONTROL INPUTS						
25	Stick Position (Pitch)	Deg	Full	To be determined		16	
26	Stick Position (Roll)	Deg	Full	" "	"	16	
27	Yaw Control Pedal Position	Deg	Full	" "	"	16	
28	Stick Force (Pitch)	1bf	±50	" "	"	16	
29	Stick Force (Roll)	1bf	±50	" "	"	16	
30	Yaw Control Pedal Force	1bf	±200	" "	"	16	
31	Pitch Trim Position	Deg	Full	" "	"	16	
32	Roll Trim Position	Deg	Full	" "	"	8	

TABLE 1 (contd)

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
33	Yaw Trim Position	Deg	Full	To be determined		8	
34	Elevator Positions*						
35	Aileron Positions*						
36	Yaw Control Positions*						
37	Port Brake Pressure	N/m^2	Full	" "	"	16	
38	Starboard Brake Pressure	N/m^2	Full	" "	"	16	
	SIGNAL SENSORS						
39	FCS state/mode	Event					
40	ILS deviation (elevation and azimuth)						
41	Flight director demand (elevation and azimuth)						
42A	Stall/spin recovery device actuation*						
42B	Spin recovery device attachment load*						
43A	Reserved						
43B	Stall Warning*						
	ENGINE (EACH)						
44	Throttle Position(s)	Deg	Full	To be determined		16	As applicable to engine under consideration.
45	Rotational Speed(s)	% or rpm	Full	" "	"	32	Power turbine and/or Gas Generator speed
46	JPT or TGT	°C	Full	" "	"	32	
47	Reserved						
48	Intake Position(s)	Deg	Full	" "	"	32	
49	Reserved			" "	"		
50	Thrust Reverser Position*						
51	Fuel Flow	Kg/ hr	Full	" "	"	16	
52	Fuel Temp at Flowmeter	°C	Full	" "	"	1	If required for fuel flow.

TABLE 1 (contd)

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
53 to 55	Reserved						
56	Inlet Guide Vane Position	Deg	Full	To be determined		32	
57	Variable Stator Vane Position	Deg	Full	" "	"	32	
58 to 60	Reserved						
61	Nozzle Angle*						
62	Auxiliary Intake Door Position*						
63 to 67	Reserved						
68	Relight Event	Event	-	-	-	16	
	MISCELLANEOUS SYSTEMS						
69	Brake Temperature (Port and starboard)	°C	0-1000	10	1	1	
70	Tyre Temperature	°C	0-150	10	1	1	Suitable on-board instrumentation may be difficult to provide and it may be necessary to resort to measuring tyre temperature after the manoeuvre using a portable temperature probe inserted into pre-drilled tyres.
71	Nosewheel angle	Deg	Full	To be decided		16	

Note: Items marked thus (*) are aeroplane related parameters which are listed in DEF-STAN 00-970 Volume 1, Chapter 900, Table 1. They have been retained in this section by number and title for reference only, at the present state of the art it is considered that they are unlikely to have relevance to rotorcraft.

TABLE 1 (contd)

ADDITIONAL ROTORCRAFT RELATED INSTRUMENTATION

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
100	Collective Pitch Control position	Deg	Full	15' arc	5'arc	16	
101	Collective Pitch Trim Position	Deg	Full	To be determined		8	
102	Collective Pitch Stick Force	1bf	±100	2	0.5	16	
103	Main Rotor Speed	% or rpm	0-130%	0.3%	±0.1%	8	
104	Input Drive Torque	% or lbf ft	0-max	1.0	±0.2%	16	Max = Full single engine emergency range +10%.
105	Main Rotor Collective Pitch	Deg	Full	15' arc	5' arc	16	
106	Tail Rotor Pitch	Deg	Full	15' arc	5' arc	16	
107	Main Rotor Head Moment	Nm or 1bf ft	Full ±20%	1.5%	0.5%	Max Freq of Interest x10	Two channels re-quired (pitch and roll) Anti aliasing filters required. (See Note 7)
108	Tail Rotor Head Moment	Nm or 1bf ft	Full ±20%	1.5%	0.5%	"	" "
109	Main Rotor Blade Flap Bending	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
110	Main Rotor Blade Lag Bending	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
111	Main Rotor Blade Torsion	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
112	Tail Rotor Blade Flap Bending	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
113	Tail Rotor Blade Lag Bending	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
114	Main Rotor Blade Pitching Moments/ Rotating Control Load	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	" "
115	Tail Rotor Blade Pitching Moments/ Rotating Control Load	Nm or 1bf in	Full ±20%	1.5%	0.5%	"	See Note 7

TABLE 1 (contd)

ITEM NO.	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
116	Main Rotor Blade Pitch Angle	Deg	Full	5' arc	5' arc	150	See Note 8
117	Main Rotor Blade Flap Angle	Deg	Full	5' arc	5' arc	150	"
118	Main Rotor Blade Lag Angle	Deg	Full	5' arc	5' arc	150	"
119	Tail Rotor Blade Pitch Angle	Deg	Full	5' arc	5' arc	250	"
120	Tail Rotor Blade Flap Angle	Deg	Full	5' arc	5' arc	250	"
121	Rotor Brake Temp	°C	0-600	10	1°C	8	
122	Landing Gear Strut Extension/ Compression	in/mm	Full	2%	½%	128	
123	Air Intake Temp	°C	-60/+60	1°C	0.5°C	8	
124	Tail Rotor Drive Shaft Torque	Nm lbf ft	Full ±20%	1.5%	0.5%	128	
125	Fore and Aft Servo Position	mm	Full	1% of full stroke	0.2mm	50	
126	Lateral Servo Position	mm	Full	1% of full stroke	0.2mm	50	
127	Collective Servo Position	mm	Full	1% of full stroke	0.2mm	50	
128	Yaw Servo Position	mm	Full	1% of full stroke	0.2mm	50	
129	Pitch Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
130	Roll Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
131	Collective Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
132	Yaw Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	

TABLE 2

RECOMMENDED TELEMETRY PARAMETERS

The selection listed below is offered for consideration. The final selection should be determined by the requirement of the specific trials programme if capacity is limited.

ITEM	PARAMETER	UNITS	RANGE	REMARKS
	GENERAL			
1	Timebase	-	-	
2	Manual Event	-	-	Available to both Ground and Flight Observers
3	Crew Speech	-	-	"Duplex" RT with ground observers.
	FLIGHT CONDITIONS			
4	Indicated Airspeed	kt	0 to Vne +20%	
5	Pressure Altitude	ft	-1000 to ceiling +3000	
6	Incidence*			
7	Pitch Attitude	Deg	±60/±10	
8	Roll Angle	Deg	±90°	
9	Sideslip Angle	Deg	±50°	
10	Heading	Deg	0 to 360	
11	Pitch Rate	Deg/sec	±100/±10	
12	Roll Rate	Deg/sec	±100/±10	
13	Yaw Rate	Deg/sec	±100/±10	
14	Longitudinal Acceleration	g	±1	) See Note 4 (items 14–16)
15	Lateral Acceleration	g	±1	) See Note 4
16	Normal Acceleration	g	-4 to +15	)
	CONTROL INPUTS			
17	Pitch Control Position	Deg	Full	

TABLE 2 (contd)

ITEM	PARAMETER	UNITS	RANGE	REMARKS
18	Yaw Control Position	Deg	Full	
19	Roll Control Position	Deg	Full	
	ENGINE (EACH)			
20	Throttle Position	Deg	Full	
21	Power Turbine Rotational Speed	%	Full	
22	Turbine Entry Temp	°C	Full	Or equivalent parameter
23	Relight Event	-	-	

TABLE 2 (contd)

RECOMMENDED ADDITIONAL TELEMETRY PARAMETERS FOR ROTORCRAFT

ITEM	PARAMETER	UNITS	RANGE	REMARKS
100	Collective Pitch Control Position	Deg	Full	
101	Main Rotor Speed	%	0.130%	
102	Input Torque	% or lbf ft	0-Max	Max = Single Engine Emergency Torque +10%
	STRESS MONITOR			
103	Main Rotor Head Moment	Nm/lbf ft	Full + 20%	Two channels required (pitch and roll)
104	Tail Rotor Head Moment	Nm/lbf ft	Full + 20%	
105	Main Rotor Blade Torsion	Nm/lbf ft	Full + 20%	) Should be at multiple stations or alternatively at known critical stations if capacity is limited. (items 105–108)
106	Tail Rotor Blade Torsion	Nm/lbf ft	Full + 20%	)
107	Main Rotor Blade Flap Bending	Nm/lbf ft	Full + 20%	)
108	Main Rotor Blade Lag Bending	Nm/lbf ft	Full + 20%	)

TABLE 2 (contd)

ITEM	PARAMETER	UNITS	RANGE	REMARKS
109	Main Rotor Rotating Control Load	N/lb	Full +20%	
110	Tail Rotor Rotating Control Load	N/lb	Full +20%	Or equivalent
111	Tail Rotor Drive Shaft Torque	Nm/lbf ft	Full +20%	

NOTES:

1. Indication of each discrete failure state of primary control and/or stability augmentation systems incorporating revisionary modes is desirable. For auto-pilot/auto-stabiliser runaway trials the magnitude of the signal from the runaway box should also be recorded to ensure that this is appropriate to the failure, under consideration.

2. Data on the disposition of fuel contents is also required: it is acceptable for such data to be derived from fuel usage combined with measurements of total fuel contents.

3. Power input for rotorcraft may involve engine mounted or gearbox mounted torquemeters.

4. Accelerometers should be chosen such that they have an adequately low threshold and high enough sensitivity and yet are not saturated by the vibration present at the mounting point.

5. If there is a device which indicates to the pilot that control load or some other parameter is approaching or exceeding pre determined limits and that he should withdraw from the situation then that data should be recorded.

6. Sufficient data should be recorded to determine the operating loads/modes which occur in the dynamic components of the rotorcraft.

   These should include the following components:

   Blades - Main or tail
   Hubs - Main or tail
   Shafts - Main or tail
   Transmission - Main or tail
   Flying controls - Main or tail - stationary and rotating
   Angular freedom of motion of blades on main and tail rotors.

7. Anti aliasing filters are likely to be required, if spurious data is not to be generated.

Optimum filter characteristic should have roll off at 24 dB/octave, although 18 dB is acceptable. 3 dB point at twice maximum frequency of interest. Phase linear with frequency.

8 Alternative sampling rate can be determined by:

$$\frac{\text{Rotor speed}}{8} \text{ x number of blades}$$

LEAFLET 900/3

GENERAL HANDLING FLIGHT TEST REQUIREMENTS

QUALITATIVE ASSESSMENT

1 INTRODUCTION

1.1 This Leaflet indicates the extent to which pilot opinion may be used for assessing the acceptability of rotorcraft handling characteristics, and places on record criteria for their use. The contents of this Leaflet are intended to provide general information and guidance.

2 ADVISORY INFORMATION

2.1 Qualitative assessment forms a major part of many tests and, as noted in Chapter 900, para 7.4 pilot opinion usually constitutes the principal means of assessing the acceptability of the handling characteristics exhibited.

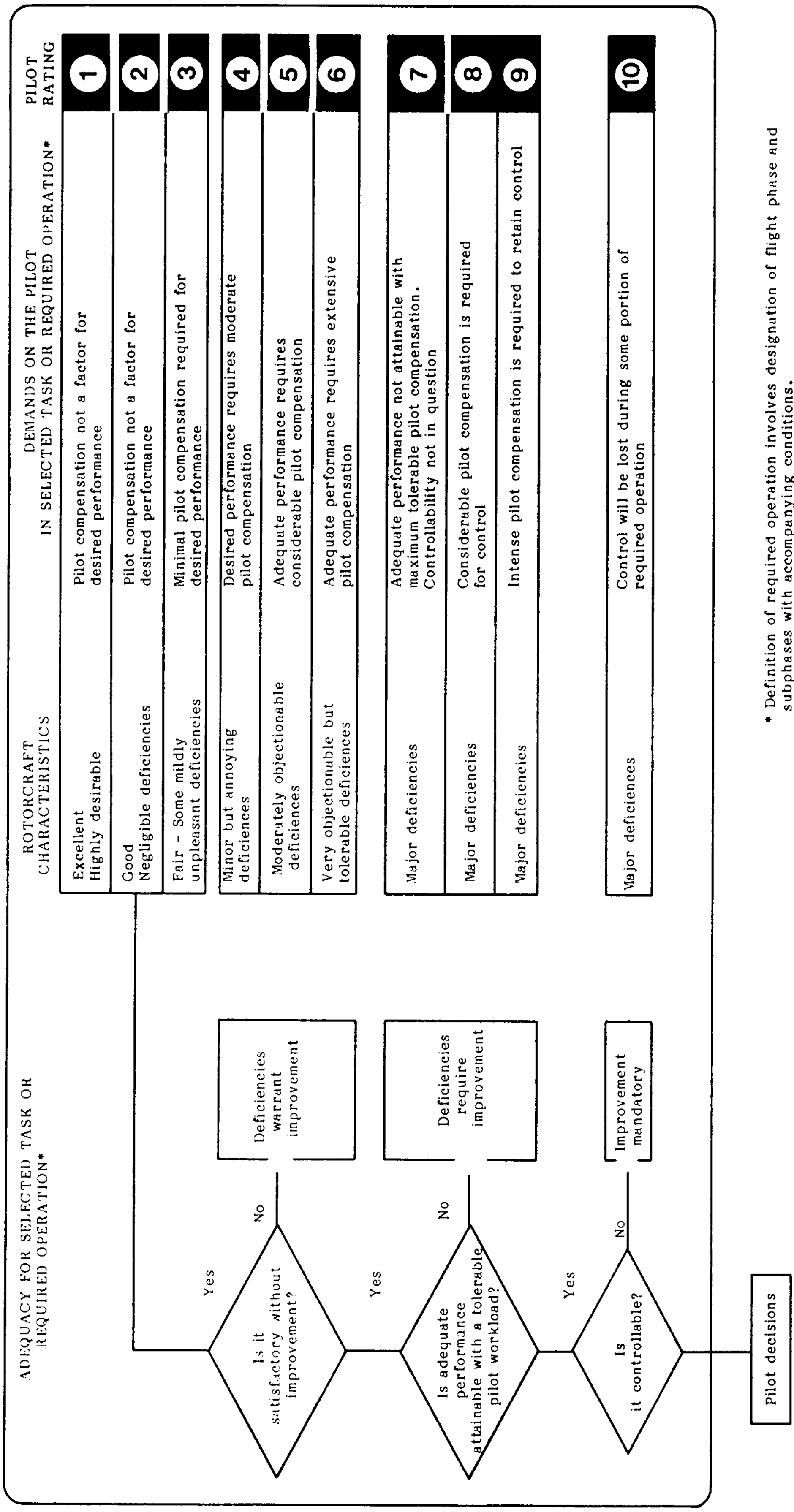
2.2 To avoid the possibility of idiosyncratic judgements, qualitative assessments should be based (in all but minor matters) on the views of at least two pilots. Minority views should not be disregarded, but the reasons for them resolved. Note should be made of the prior experience of the pilots involved in the assessment, in terms of their flying hours in total, on the type or on similar types of rotorcraft and systems, and in particular role(s) involved.

2.3 To facilitate consistent expression and interpretation of pilot opinion full use should be made of the revised Cooper-Harper Handling Qualities Rating Scale. That scale is presented in Fig 1, but Ref 1 gives a fuller explanation of its derivation and use, and of the meaning of the terms in it. Whenever the Cooper-Harper Rating Scale is used the task should be fully defined in terms of the quantities to be achieved. The pilot's assessment of the tendency for pilot induced oscillations to occur, and their classification of the levels of turbulence encountered should preferably be expressed in accordance with the appropriate tables also given in Ref. 1. However, the use of such scales should supplement, and not replace specific pilot comment in respect of the characteristics observed and their significance with regard to Service usage.

2.4 In order to assess the significance of events (such as autopilot runaways) which cause a disturbance to the initial steady flight conditions, it is necessary to take into account, inter alia, the time required for the relevant cues and/or warnings to be recognised by the pilot, and the subsequent time lapse before recovery action is initiated. The choice of appropriate cue recognition, warning and pilot response times must be based, ultimately, on the judgement of the pilots involved in the assessment.

REFERENCES

No.	Author	Title etc.
1	Cooper G.E. and Harper, Jnr R.P.	The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities. AGARD Report 567, April 1969.



* Definition of required operation involves designation of flight phase and subphases with accompanying conditions.

Cooper- Harper Ref. NASA TND.5153

FIG.1 - HANDLING QUALITIES RATING SCALE

LEAFLET 900/4

GENERAL HANDLING FLIGHT TEST REQUIREMENTS

SUBJECTIVE DEFINITION OF TURBULENT AIR STANDARDS

1 INTRODUCTION

1.1 Flight in turbulent air is an important consideration when concerned with any handling, stability or auto pilot assessment.

1.2 There is a clear requirement for a common standard against which turbulence levels can be recognised by test crews and reported. This leaflet offers a turbulent air scale based on that originally contained in AvP 970 Vol. 1 as Leaflet 916/2 and which has been regularly quoted as a basis for turbulence definition in specification documents. (See Table 1).

2 DESIGN DEFINITIONS

2.1 The general requirements and definitions concerning atmospheric disturbances are discussed at length in Leaflet 600/1 (to be issued).

2.2 When related to the day to day activities in a flight programme the definition of turbulence in quantitative terms is difficult and generally impractical, the effect of turbulence in engineering values can usually only be determined in retrospect following analysis of results.

2.3 The turbulence scale is therefore proposed as an aid to reporting for test operations and debrief and is intended to supplement the engineering definitions contained in Leaflet 600/1 (to be issued).

TABLE 1

TURBULENT AIR SCALE

Scale	Definition	Air Conditions
1	-	Flat calm
2 3	Light	Fairly smooth, occasional gentle displacement. Small movements requiring correction if in manual control.
4 5 6	Moderate	Continuous small bumps. Continuous medium bumps. Medium bumps with occasional heavy ones.
7 8	Severe	Continuous heavy bumps. Occasional negative 'g'.
9 10	Extreme	Rotorcraft difficult to control. Rotorcraft lifted bodily several hundreds of feet.

Note: The definitions included in the above trials are intended to relate loosely to those contained in DEF-STAN 00-970 Vol. 1, Leaflet 600/5 Table 2.

CHAPTER 901

GROUND HANDLING

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the rotorcraft can be satisfactorily controlled on the ground at all power settings, and when taxying or ground running in various wind velocities.

1.2 The behaviour of the rotorcraft on the ground is affected by:

(i) Use of power.

(ii) Efficiency and ease of operation of the wheel brakes.

(iii) Taxi/runway surface, profile.

(iv) The mass and centre of gravity of the rotorcraft.

(v) The methods of steering - e.g., nosewheel, differential brakes, tail rotor and/or main rotors.

(vi) The speed of taxying.

(vii) The wind strength and speed component relative to the heading of the rotorcraft.

1.3 Ground handling data shall be obtained at every opportunity during other tests.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2. Chapters 301, 302, 303, 308 and 310.

3 APPLICABILITY

3.1 The following tests are applicable to all classes of rotorcraft except where stated otherwise.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS
4.1.1 No specific instruments are required for these tests.

4.2 TEST INSTRUMENTATION
4.2.1 The parameters which should be recorded during these tests are listed in Leaflet 901/1.

4.3 GROUND EQUIPMENT
4.3.1 The ground equipments which should be used in support of these tests are listed in Leaflet 901/1.

5 LOADING

5.1 The tests shall be made at loadings such that load conditions W_{max}, W_{fwd}, W_{aft}, Y2, and Z2 as defined by Chapter 900 para 5.1 are covered.

5.2 If the rotorcraft is capable of carrying external stores, these loadings shall include the most adverse combinations of stores in relation to:

(i) The ground clearance of the stores.

(ii) The pitch inertia of the rotorcraft.

(iii) The roll inertia of the rotorcraft.

6 GENERAL TEST CONDITIONS

6.1 RUNWAYS AND OPERATING AREAS

6.1.1 The tests shall be made on normal paved dispersal areas. Every effort shall be made to use surfaces which are representative of the Specification requirements in terms of surface roughness and softness including operation on icy surfaces.

6.1.2 Consideration shall also be given to the probability that operations may involve regular or continuous operations from grass, hard sand or snow areas.

6.2 STANDARD OF ROTORCRAFT

6.2.1 The rotorcraft shall be representative of Service standard in terms of engine, torque, ground directional control and brake characteristics.

7 TESTS

7.1 Ground handling characteristics will largely be assessed from pilot qualitative opinion and, for this purpose, the Cooper-Harper Pilot Rating Scale (see Leaflet 900/3) may be used for each of these tests, but for this rating scale to produce meaningful results, the task must be adequately defined, e.g., taxy within ±2 m of runway centreline at 10 kt ±2 kt.

Note: It must be appreciated that once the rotor is engaged, the rotorcraft is running in a flight condition and control movements which would have little or no effect on a rotorcraft can apply significant forces or moments on a running rotor.

7.1.1 The initial phase of ground running tests will be an essential part of the ground resonance testing to which the prototype rotorcraft has to be subjected before the first flight is undertaken (see Chapter 500). These tests, once completed will give confidence that the rotorcraft is free of potentially dangerous resonance characteristics. During subsequent testing for ground handling it is implicit that vigilance for ground resonance should be maintained.

7.2 GROUND RUNNING, AGAINST BRAKES OR WHEEL LOCKS

7.2.1 This test represents the maintenance test case where rotorcraft may be run for prolonged periods on the ground for trouble shooting and system tests. There may also be a requirement to apply some collective pitch as part of power checking and response.

7.2.2 The tests shall be made with wheels unchocked and wheel brakes or locks applied. The following occurrences can be considered as limiting conditions.

(i) Wheel slip.

(ii) Brake slip.

(iii) Fuselage rotation.

(iv) Contact or hammering on rotor blade stops.

(v) Evidence of dangerous proximity of blades to fuselage structure.

(vi) Rotorcraft on point of lift off or oleo full extension due to power application.

7.2.3 ROTOR ACCELERATION, DECELERATION AND BRAKING

(i) The process of accelerating the rotor following rotor brake release or clutch engagement shall be subject to testing. The behaviour of the rotor throughout the whole available rotor speed range with engines running, rotor brake off M.P.O.G. must be tested. The tests shall be made throughout a range of wind strengths, from all quarters up to a maximum required by the specification, if no wind strength is specified, up to 30 knots gusting to 40 knots.

(ii) The above tests will investigate the operation of blade flapping stops, blade sailing characteristics and the effect of small control displacements in both collective and cyclic pitch with a view to providing clear guidance regarding the handling limitations during this regime.

(iii) The behaviour of the rotors and controls during brake applications and free rotor rundown shall be tested to ensure that no structural limitations are exceeded and to ensure that the controls cannot be inadvertently mishandled in such a way as to endanger the rotorcraft.

(iv) Some rotorcraft have the capability to apply negative collective pitch and this feature will be included in these tests.

7.2.4 ROTORS RUNNING

(i) From rotor running M.P.O.G. the maximum collective pitch which can be applied to the point where the rotorcraft is on the point of lift off or full oleo extension shall be established.

(ii) Applications of collective pitch shall be carried out, increasing and decreasing, working progressively from a slow steady rate to a fast application which is representative of that specified ensuring that none of the limiting conditions listed in para 7.2 occur or can be easily exceeded.

(iii) With the rotorcraft running M.P.O.G. the brakes shall be released and the rotorcraft turned in and out of wind using the tail rotor control and wheel steering or castoring as applicable. The ability to stop rotation on any selected heading shall be assessed.

(iv) Testing shall be extended to cover the full range of wind conditions, including gusts, from which ground running wind limitations can be established.

7.3 TAXYING - STRAIGHT AHEAD AND IN GENTLE TURNS

7.3.1 The rotorcraft shall be taxied in a straight path and in gentle turns typical of normal taxying requirements using the appropriate directional controls (for example, nosewheel steering, differential brake, tail rotor application) and cyclic/collective controls to maintain the nosewheel on the ground. The tests shall be made throughout a range of wind strengths, from all quarters up to the maximum required by the specification or, if no wind strength is specified, up to 30 kts (with gusts up to 40 kts). If a nosewheel castoring lock is fitted, these tests shall be made with and without the lock engaged.

7.4 TAXYING - SMALL RADIUS TURNS

7.4.1 The rotorcraft shall be taxied through 360° in a turn of small radius to the left and right using the appropriate directional controls. If a nosewheel castoring lock is fitted, these tests need only be made with the lock disengaged.

7.5 HIGH SPEED TAXYING

7.5.1 In addition to the tests recommended in Leaflet 301/2, the rotorcraft shall be taxied in a straight path at speeds up to 10% in excess of the run on landing speed (Chapter 907 refers) appropriate to the configuration, and any tendency for the nosewheel to shimmy noted.

7.5.2 The adequacy of the brakes to arrest the rotorcraft from high speed shall also be assessed together with brake and tyre temperatures (if either are likely to be critical (also Chapter 1008)), the adequacy of directional control and pilot comfort over rough or uneven surfaces.

NOTE: To comply with the shimmy requirement of Chap.301, para 5, this speed should be increased to 1.15 times the appropriate run on or running take off speed unless theoretical analysis shows the higher speed to be acceptable - see Leaflet 301/1.

LEAFLET 901/1

GROUND HANDLING

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 901, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
9	Pitch attitude
10	Roll angle
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
25	Stick position - pitch
26	Stick position - roll
27	Yaw control pedal position
37	Port brake pressure
38	Starboard brake pressure
44	Throttle position(s)
45	Engine rotational speed(s)
46	JPT/TGT
51	Fuel flow
52 (if required for 51)	Fuel temperature at flow meter
69	Port and starboard brake temperature
70	Tyre temperature

71	Nosewheel angle
100	Collective pitch control position
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
106	Tail rotor pitch
107	Main rotor head moment
123	Engine intake temperature
124	Tail rotor drive shaft torque

1.3 The following additional parameters may also be useful:

24	Fuel contents
28	Stick force pitch
29	Stick force roll
30	Yaw control pedal force

2 GROUND EQUIPMENT

2.1 In addition to the onboard instrumentation, the following ground equipment should also be provided:

(i) A means of measuring the ground speed of the rotorcraft. For general testing this can be achieved by use of a calibrated pace vehicle but in critical areas it may be necessary to use Kinetheodolite equipment. Adaption of Microwave Distance Measuring equipment can be considered as an alternative means of measuring ground speed with the added advantage that a cockpit readout can be made available to the pilot to supplement onboard recording.

(ii) Means for measuring wind speed and direction, e.g., anemometer and wind direction vane.

(iii) Means for measuring ambient temperature.

(iv) Means for measuring surface friction of the runway, e.g., Mu-meter.

And for tests on other than paved runways:

(v) Means for measuring surface hardness, e.g., cone penitrometer, and surface roughness.

CHAPTER 902

TAKE-OFF, HOVER, LOW SPEED MANOEUVRES AND LANDING

1 OBJECT

1.1 This chapter describes the tests to be made to investigate the handling characteristics of the rotorcraft during the take-off, landing and airfield manoeuvring phase of flight.

1.2 The object of these tests is to ensure that control about all three axes is adequate during these manoeuvres while maintaining the required ground clearance by use of the collective pitch control. The ease with which control can be maintained and the acceptability of take-off and landing techniques proposed for service use are to be assessed.

1.3 The tests in this chapter are designed to cover those factors which can affect handling characteristics during take-off, landing, sideways and rearwards flight, spot turns and transition to and from forward flight.

Tests include the effects of:

(i) Mass and centre of gravity position of the rotorcraft.

(ii) Carriage of external stores or underslung loads.

(iii) Flight with doors open-en or removed.

(iv) Relative wind velocity including the effects of gusts.

(v) Density altitude.

(vi) Deficiencies in pilot's field of view.

(vii) Sloping ground.

(viii) Ambient light levels.

(ix) Surface texture and conditions.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Part 6 (In preparation).

3 APPLICABILITY

3.1 The following tests are applicable to all classes of rotorcraft.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 In addition to the normal cockpit instruments a visual readout of collective, cyclic and yaw pedal position shall be included and tail rotor blade angle would also be required where the control system includes any control interlink. If a doppler radar is part of the normal build for the rotorcraft, every effort shall be made to make this operative.

4.2 TEST INSTRUMENTATION

4.2.1 The parameters which should be recorded during these tests are listed in Leaflet 902/1.

4.3 GROUND EQUIPMENT

4.3.1 The ground equipments which should be used in support of these tests are listed in Leaflet 902/1. Most of the instrumentation items necessary for the tests are covered by on board equipment but video recording may be required for flight path and ground clearance assessment.

4.3.2 A vehicle may be used to provide an accurate datum for airfield manoeuvres in lateral and rearwards flight.

5 LOADINGS

5.1 Adequate control margins are essential to rotorcraft operations particularly in low speed manoeuvres and there can be no relaxation in the requirements to investigate this regime of flight comprehensively. Testing shall embrace all the loadings listed in Chapter 900 para 5.1.

5.2 If the rotorcraft is capable of carriage of underslung loads, rescue hoist or sonar role, loadings shall be included to ensure that the effect of such roles falls within the margins determined from the basic range of mass and C of G limits.

6 GENERAL TEST CONDITIONS

6.1 WIND CONDITIONS

6.1.1 Initial tests shall be made in conditions of light wind but can be quickly progressed to cover increasing steady wind strengths until confidence is gained that testing can be extended to include gusts and turbulence up to the maximum required by the rotorcraft specification or as a minimum up to 30 knots (with gusts up to 40 knots turbulence level 6 - see Leaflet 900/4 Table 1).

6.2 ENGINE POWER

6.2.1 Testing shall be carried out using the definitive range of engine power ratings including the effect on power margins due to the use of engine anti icing and cabin conditioning or systems calling for engine air bleeds.

6.2.2 Where the specification states allowance for in service deterioration in the engines power delivery, tests shall demonstrate that this does not impair the take off or low speed manoeuvre handling.

6.3 ALTITUDE AND TEMPERATURE

6.3.1 Tests are required to ensure that control and handling are satisfactory over the temperature/altitude range. This may be achieved by confirming control margins over a range of $W/\sigma\ \omega^2$ conditions which will be gathered as part of the performance programme.

6.3.2 Opportunity must be taken to confirm representative temperature/handling under actual conditions as part of climatic testing. It is appreciated that there may be difficulty achieving a specific test condition but such testing as is practical under actual conditions must be undertaken to underwrite the handling envelope and confirm the validity of conditions extrapolated from normalised testing.

6.4 RUNWAY OR HELIPAD SURFACE

6.4.1 Tests shall be made from concrete or grass surface and should be extended to include unprepared grass areas and sloping sites.

6.5 AUTOMATIC FLIGHT CONTROL SYSTEM

6.5.1 Testing must include any stability augmentation system which is part of the rotorcraft's normal build, including runaway protection. Once confidence is gained and the normal handling characteristics of the rotorcraft fully explored and understood testing shall be extended to include the effects of degraded control which may arise from single hydraulic system operation on duplex systems or specific protected failures which may arise in active control systems.

7 TESTS

7.1 SUMMARY

7.1.1 The tests discussed below include take-off, landing, sideways and rearwards flight, spot turns and transition to and from forward flight. The effects and procedures for engine failure or rejected take off will not be covered in depth in this chapter as they form an essential part of Chapters 907 and 908.

7.1.2 Acceptable handling characteristics may be achieved by use of a Stability Augmentation System but failure of such a system must not degrade the rotorcraft's controllability to worse than Pilot Rating 6 on the Cooper-Harper scale, see Leaflet 900/3.

7.1.3 Where the manufacturer elects to depart from direct mechanical controls, for example by the use of Active Control Technology, where total system failure would be catastrophic, then it will be necessary to agree with the Rotorcraft Project Director, a level of system failure where controllability is still acceptable.

7.2 SPEED DEFINITIONS

7.2.1 The tests discussed in this chapter are mainly concerned with operation in the hover and low speed excursions in any defined radial direction. References will either be to hover I.G.E. or O.G.E. Velocities will be defined as a lateral direction of flight relative to the fore and aft axis and speed in knots which will refer to airspeed unless specifically called up as groundspeed.

7.3 TAKE-OFF PHASE

7.3.1 Tests are required to assess the take-off phase and shall cover the following:

(i) Normal take-off i.e., lift off from level ground into the hover, facing into wind.

(ii) Take-off from level ground, out of wind to the hover.

(iii) Take-off from sloping ground, both into wind and out of wind to the hover including assessment with the rotorcraft heading across the slope.

(iv) Running take-off to forward flight from level ground.

7.3.2 The take-off manoeuvre shall be deemed to commence once the collective pitch lever starts to rise. The tests shall demonstrate that the take-off manoeuvre can be completed without the need for excessive control inputs and that adequate control margins are available.

7.4 NORMAL TAKE-OFF

7.4.1 Take-off tests shall commence with smooth steady application of collective pitch to establish a stable hover I.G.E. The tests shall demonstrate that the rotorcraft is fully controllable in all axes over a range of wind strengths up to specification limits including gusts.

7.4.2 The assessment shall include appraisal of the rotorcraft with the landing gear shock absorbers fully extended or skids touching at the point of lift off. The above condition shall be free from resonance with good reserves of directional control.

7.4.3 As the programme continues, testing shall be extended to include rapid take-off and shall cover the range of surfaces from which the rotorcraft may be expected to operate.

7.4.4 Cold climatic trials shall include assessment of take-off from iced surfaces.

7.4.5 All the above tests shall include take-off and hover with the Stability Augmentation system disengaged if applicable.

7.5 TAKE-OFF OUT OF WIND

7.5.1 Tests shall be made to assess take-off with wind on the port and starboard quarters. Opportunity shall be taken on suitable occasions to assess handling with the rotorcraft lifting off with the mean wind direction 030°, 060°, 090° and 120° relative to the fore and aft axis in a programme of tests which works up in wind speed progressively. The aim shall be to extend testing so that the rotorcraft can be cleared for operation out of wind in any direction.

7.5.2 Degradation arising from wind in the rearward quadrants shall be investigated with a view to minimising such effects and demonstrating that the rotorcraft can be safely operated under such conditions albeit that some limitations or safeguards have to be declared.

7.5.3 Areas of interest are:

(i) The point where the landing gear is just in ground contact where it is important to ensure freedom from resonance. At this point adequate control shall be available.

(ii) Once in an established out of wind hover, it must be possible to show adequate lateral and directional control, and to demonstrate that buffeting from the tail rotor (if applicable) is at a low level and non damaging.

(iii) Operation out of wind can result in a considerable change in the flow of cooling air. Cooling cut outs and intakes are designed to be at their optimum with the rotorcraft heading into wind. Transmission bay cooling and jet pipe temperatures shall be monitored during trials involving prolonged out of wind running or flying.

7.6 OPERATION FROM SLOPING SITES AND SURFACES

7.6.1 The limits of sloping surface from which the rotorcraft can takeoff must be determined and clear guidance regarding technique included in the aircrew manual. The programme shall cover take-off heading up and (if feasible) down the slope and laterally to port and starboard. The tests shall establish the limits with rotors kept running and also stopping and starting rotors.

7.6.2 The limiting factors to be assessed are:

(i) The programme will entail a series of landings and take-offs to and from surveyed slopes commencing at a modest angle and progressively working up to an agreed margin beyond specification limits.

The tests will generally be made into wind although some crosswind components are not only acceptable but operationally representative and consequently shall be included in testing.

Each test condition (heading on slope/angle) shall be completed at least three times and the following results gathered:

(a) Pilot's handling, comments.

(b) Landing gear stresses.

(c) Transmission and rotor stress data.

(d) Control position measurements.

(e) Cine or video records. (If required to demonstrate adequate ground or structural clearance).

(ii) From the above testing, a slope landing/takeoff profile should result, the intention shall be to clear the rotorcraft for full operation on all surfaces and including rotor shut down and start up. It may be that the range of slopes usable for landing and take-off with rotors kept running while on the ground will allow operation from steeper slopes than possible if shut down is included. If this is not qualified in the rotorcraft specification it must be clearly stated as the result of testing.

The final statement of sloping site limits must allow a tolerance of ±1° in the measured slope and contain sufficient tolerance to ensure that stress and landing gear limits cannot be easily exceeded by mishandling.

(iii) The limits of sloped site operation will generally be called up in detail in the specification but in the absence of any definition, testing shall take place on slopes up to 7° (in any heading) although limitations with rotorcraft heading down the slope may be negotiable. Any tendency to approach limitations must be discussed with the Rotorcraft Project Director.

7.7 STABILISED HOVER

7.7.1 Tests are required to assess the handling characteristics in a sustained hover I.G.E. and O.G.E. covering the following:

(i) Control movements necessary to hold hover in calm air (wind < 5 kts) covering cyclic, collective pitch and yaw controls.

(ii) Ability to make small adjustments to hover position or altitude.

(iii) Overall work load to sustain hover.

(iv) Effect of ASE (if applicable).

(v) Effect of small deliberate control movements.

(vi) Assessment of cockpit vision, clear vision panels and usable cues to assist in hover position keeping and landing.

7.7.2 Testing shall be extended to include wind and turbulence levels to specification limits and the intended operational use of the rotorcraft must be considered.

7.8 LOW SPEED MANOEUVRES

7.8.1 Tests are required to assess handling in low speed manoeuvre cases. This represents the regime of flight which is particularly the province of the rotorcraft and is of special relevance to shipboard operation. Tests shall cover the following:

(i) Low speed flight covering the following conditions:

(a) Forward 0 to 30 kts.

(b) Aft 0 to 20 kts.

(c) Left and Right 0 to 30 kts.

(d) Flight in the forward quadrants 0 to 30 kts.

Note: The stated limiting speeds have been offered as guidance and may be modified by the specification requirements or with the agreement of the Rotorcraft Project Director.

(ii) Hover spot turns up to yaw rate limits and recovery.

(iii) Recovery from limiting low speed conditions to hover.

(iv) Changes in direction of low speed flight.

7.8.2 During each of the above conditions the following test results and observations are required.

(i) Control positions and margins.

(ii) Assessment of control response and effect.

(iii) Effect of A.S.E. (if applicable).

(iv) Assessment of cockpit vision, clear vision panels and usable cues to undertake the manoeuvres.

(v) Effect of C of G and weight.

7.9 TRANSITION TO FORWARD FLIGHT

7.9.1 Tests shall be carried out to confirm that the rotorcraft is controllable during the process of transition from hover I.G.E. to forward flight in the climb (Vy). This represents the final stage of the take off operation and must be achievable without degradation of control at any stage with systems or power plants functioning correctly. Testing must be extended to include degraded control arising from single system failure in duplex power controls or manual characteristics where manual reversion is a feature of the rotorcraft control system in any axis.

7.9.2 Where A.C.T. (Active Control Technology) is included as a feature of the design, testing will be required to demonstrate that the take off and landing procedures cannot be unacceptably degraded by failure modes in power supplies or software.

7.10 TRANSITION FROM FORWARD FLIGHT TO THE HOVER

7.10.1 Tests shall be carried out to confirm that the rotorcraft is controllable during the process of transition from forward flight into the hover which will generally represent the final stages prior to landing.

7.10.2 Tests shall include the following cases:

(i) Gentle transition from level flight to hover I.G.E.

(ii) Transition from a normal glide slope descending approach to hover I.G.E. Testing shall be extended to include any procedural approach techniques which may apply to the rotorcraft in its normal operations.

(iii) Quick stop from approximately 60 kts level flight to the hover.

(iv) Down wind approach to the hover and turn into wind for landing.

7.11 COCKPIT CONSIDERATIONS

7.11.1 An essential part of the assessment will be a full appraisal of the cockpit and its all round vision. Sufficient transparency area must be available to provide adequate outside visual cues for the landing and take-off process and to allow safe lateral flight. The assessment shall be extended to include the availability of clear vision panels. The ability to accurately sense wheel placing and obstacle clearance during the take-off/landing process has a direct bearing on the tests contained in this chapter and represents a key part of the cockpit environment appraisal.

LEAFLET 902/1

TAKE OFF, HOVER, LOW SPEED MANOEUVRES AND LANDING

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 902, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
5	Pressure Altitude
6	Altitude (Radio Altimeter)
9	Pitch attitude
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
25	Stick position - pitch
26	Stick position - roll
27	Yaw control pedal position
28	Stick force - pitch
29	Stick force - roll
30	Yaw control pedal force
37	Port brake pressure
38	Starboard brake pressure
56	Inlet guide vane position
69	Port and starboard brake temperature
70	Tyre temperature

71	Nosewheel angle
100	Collective pitch control position
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
106	Tail rotor pitch
107	Main rotor head moment
109	Main rotor blade flap bending
110	Main rotor blade lag bending
111	Main rotor blade torsion
114	Main rotor blade rotating control load
115	Tail rotor blade rotating control load
123	Air intake temperature

1.3 The following additional parameters may also be useful:

Item	**Parameter**
20	Landing gear position
24	Fuel contents
51	Fuel flow
52	Fuel temperature at flowmeter
121	Rotor brake temperature
122	Landing gear strut extension/compression

2 GROUND EQUIPMENT

2.1 In addition to the onboard instrumentation, the following ground equipment should be provided:

(i) Means for measuring the speed and trajectory of the rotorcraft. Kinetheodolite equipment provides a very precise means of achieving this but generally involves the arrangement of trials at dedicated Kine ranges requiring the rotorcraft to be detached from its normal operating base. The provision of a calibrated chase vehicle against which the rotorcraft can formate is an acceptable method for assessing low speed manoeuvres. Final assessment can then be undertaken by trials at a Kinetheodolite range.

Adaption of Microwave Distance Measuring equipment used with Radio Altitude can be considered as an alternative means of providing data on speed and trajectory.

(ii) Means for measuring wind speed and direction, e.g. , anemometer and wind direction vane. (Wind speed and direction should be measured at a height representative of the rotorcraft's hover height).

(iii) Means for measuring ambient temperature.

(iv) Means for measuring surface friction of the runway, e.g. Mu-meter.

and, for tests on other than paved runways:

(v) Means for measuring surface hardness, e.g., cone penitrometer, and surface roughness.

(vi) Video cameras provide a good flight safety back up for take off and landing assessment as well as good verification of external observers reports.

(vii) The installation of a low speed measuring device such as L.A.S.S.I.E. should be considered for the low speed manoeuvre phase of the programme.

CHAPTER 903

LONGITUDINAL TRIM, STABILITY AND CONTROL

1 OBJECT

1.1 The object of the tests in this chapter is to demonstrate that the longitudinal handling characteristics of the rotorcraft are satisfactory by investigation of the:

(i) Longitudinal control system characteristics.

(ii) Longitudinal static stability.

(iii) Longitudinal dynamic stability - behaviour in oscillatory modes.

(iv) Longitudinal trim changes, effect of fore and aft C of G changes.

(v) Effect of turbulence.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2, Chapter 601

3 APPLICABILITY

3.1 These tests are applicable to all classes of rotorcraft and to all the variation of rotorcraft control systems discussed in Chapter 900 para 6.2.1.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 The test rotorcraft shall be fitted with a sideslip indicator, an accelerometer, a pitch attitude indicator and control position indicators for longitudinal and lateral cyclic position, yaw pedal position and collective pitch.

4.2 TEST INSTRUMENTATION

4.2.1 The parameters which should be recorded are listed in Leaflet 903/1. The introduction of a control input device capable of applying precise control inputs (amplitude and duration) with good repeatability would be advantageous and should be given serious consideration where the rotorcraft control system allows such a device to be fitted without serious breakdown. Any such system must result in no downgrading of the control system and shall be free of any risk of spurious operation.

5 LOADING

5.1 Tests shall be made covering the following loadings:

W_{max}, $W_{max/aft}$, X, X_1, X_2, Y, Y_1, Y_2, Z, Z_1 and Z_2 as defined in Chapter 900 para 5.1.

5.2 If the rotorcraft is capable of carrying external stores or role equipment, these loadings shall include the most adverse combination of stores in relation to:

(i) Highest mass.

(ii) Fore and aft C of G limitations.

(iii) The most probable combination of store loading(s) for Service use if this is/these are not covered by (i) and (ii) above.

(iv) Consideration shall be given to the possibility of cross coupling arising from lateral C of G. If testing indicates that lateral C of G changes bring about changes in longitudinal control and trim, the test programme shall be extended to include the appropriate cases.

6 GENERAL TEST CONDITIONS

6.1 ALTITUDE
6.1.1 The tests shall be made at low, medium and high altitudes appropriate to the rotorcraft under test and commensurate with flight safety.

6.2 ROTORCRAFT CONFIGURATION AND SPEED RANGES REQUIRED FOR TESTS
6.2.1 The tests shall be made over the speed ranges indicated below. Generally rotorcraft do not employ lift devices which are deployed in various flight regimes but there is some precedent for air brakes and controllable incidence on the tailplane. There is also the possibility that convertiplanes or tilt rotor aircraft may employ variable incidence or activated surfaces to improve control in given regimes. Should such aircraft be presented for test, advice must be sought from the Rotorcraft Project Director and the requirement reviewed.

The table below is applicable to conventional rotorcraft with either tail rotors or twin main rotors.

TABLE 1

Ref.	**Flight Regime**	**Landing Gear**	**Ref Volume 1 Leaflet 600/1 Flight Config Applicable**	**Speed Range**
6.2.1	Take-off	down	C	Hover to Vy
6.2.2	Hover	up and down	C	Hover to 30 kts Vlat, Vfwd and Vaft
6.2.3	Cruise	up	A and B	30 kts to 1.11 VNe
6.2.4	Manoeuvre	up	A and B	30 kts to VNe
6.2.5	Approach	down	C	VNo to hover
6.2.6	Landing	down	C	Hover

7 TESTS

7.1 PREPARATION

7.1.1 Before the tests given in the following paragraphs are commenced, the breakout forces, circuit frictions and backlash of the longitudinal control system shall be measured on those systems where this is appropriate. The control force/displacement characteristics over the whole range shall also be measured. Unless specified, all tests shall be made with autostabilisers operative. Testing shall take into account the effect of autostabilisers on flying control positions.

7.2 LONGITUDINAL CONTROL DURING TAKE-OFF

7.2.1 Tests to assess rotorcraft handling during take-off are discussed in Chapter 902 and it is implicit in the text that the assessment will include longitudinal control. Tests are required to determine the optimum longitudinal trim position to result in minimum disturbance during the progress of lift off while the application of collective pitch is taking place.

7.2.2 Continuous instrumentation records showing control positions, attitude and power application are required at neutral or normal operating loading, also covering aft and forward C of G (Y_1 and Z_1).

7.2.3 Testing shall be extended to determine the effect of mistrim. Take-off records as called up above are required with progressive increments of forward and aft trim applied before commencing take-off until unacceptable handling characteristics are experienced or trim limits reached.

7.3 APPARENT STATIC STABILITY

7.3.1 Measurements are required for cyclic control position V_S forward speed in the hover and through the forward speed range up to 1.11 VNe and also in rearward flight to the max permitted speed. The test programme shall include climb, powered descent and autorotation and to cover full aft and forward C of G. The results must demonstrate that at least 10% of the maximum attainable hover pitching moment is available under all flight conditions to accommodate longitudinal disturbances without reaching control limits.

7.4 TRIM CHANGES WITH POWER

7.4.1 Testing is required to assess the rotorcraft's response to power changes in trimmed level flight at a range of airspeeds. The rotorcraft shall be stabilised in level flight and control forces reduced to zero by use of the trimmer with cyclic free, power shall be increased or decreased and the rotorcraft allowed to respond to the input. Recovery shall be made before limitations are exceeded.

7.4.2 Tests should be made for the following trimmed conditions:

(i) Minimum power level speed.

(ii) VNe - 20 kts.

(iii) Mid speed range between (i) and (ii).

7.5 STATIC LONGITUDINAL STABILITY

7.5.1 Tests are required to determine the control positions and forces necessary to balance the pitching about the rotorcraft's centre of gravity caused by airspeed variation. The tests shall be accomplished by establishing a trim condition (airspeed/power combination) with zero control forces. Without changing collective position, trim setting or rotor speed, the rotorcraft shall be stabilised at incremental airspeeds faster and slower than the trim airspeed in small increments of 2 to 5 knots over a range ±20 knots, height should be maintained ± 1000 ft. of test altitude.

The recommended test points for the above tests are as follows:

TABLE 2

CONDITION	TEST POINT	ALTITUDE
(i) Cruise	(a) Min power level speed (b) VNe - 20 kts (c) Mid range between (a) and (b)	2000 ft & 7000 ft
(ii) Hover	Hover ± 30 kts airspeed	I.G.E.
(iii) Approach	(a) Min power speed, 1000 ft/ min descent (b) Recommended approach speeds/descent rates if significantly different from above	Through 2000 ft.
(iv) Max Power Climb	Best climbing speed	Through 2000 ft.
(v) Autorotation	Best descent speed	Through 2000 ft.

Note: The selection of minimum power speeds, approach speeds and descent rates will be, to some degree dependent upon the outcome of performance measurements and approach techniques.

7.6 MANOEUVRING LONGITUDINAL STABILITY

7.6.1 The manoeuvring longitudinal stability shall be measured on all types of rotorcraft in each of the configurations called up in para 6.2. In the configuration listed as references 6.2.1, 6.2.2 and 6.2.6 tests shall be made at low level only (in ground effect). The approach condition shall be at low level (2000 ft ± 1000 ft). Tests in configuration references 6.2.3 and 6.2.4 shall cover 2000 and 7000 ft ± 2000 ft with further tests at service ceiling if the specification calls for operation above 10,000 ft.

7.6.2 Tests are required to determine longitudinal control forces in steady turns and the effect of increased bank angles.

7.6.3 The dynamic manoeuvre stability must be assessed by an appropriate technique such as symmetrical pull up and push over tests for fixed collective pitch.

7.7 DYNAMIC STABILITY

7.7.1 The purpose of dynamic longitudinal stability flight testing is to investigate the two major characteristic modes of free rotorcraft motion, i.e.:

(i) The long period mode, as phugoid.

(ii) The short period mode.

7.7.2 The dynamic response of the rotorcraft to various pilot control inputs is important in evaluating its handling qualities and one of the prime objects of these tests is to evaluate these handling qualities with respect to the expected missions and to verify that the rotorcraft is suitable for its intended role.

7.7.3 THE PHUGOID MODE

(i) The rotorcraft shall first be stabilised and carefully trimmed with A.S.E. engaged at the required speed and altitude. The phugoid mode shall then be excited by smoothly increasing or decreasing the airspeed by some 10 to 15 kts by means of the appropriate stick displacement (or force) and the stick returned to the trimmed position and released. At least five cycles or more should be recorded if possible.

(ii) The tests shall be repeated with A.S.E. off and pilot intervention will probably be necessary before completion of five cycles. Testing shall also include assessment of the ability of the pilot to suppress any phugoid encountered.

(iii) The rotorcraft configurations, speeds and altitudes which shall be considered are the same as those required for the longitudinal manoeuvring stability tests of para 7.6 but flight tests are only necessary in those areas of the flight envelope where simulator studies and/or flight tests by the contractor indicate that large amplitude or undamped phugoid oscillations may occur.

7.7.4 LONGITUDINAL SHORT PERIOD STABILITY

(i) The rotorcraft shall first be stabilised and trimmed at the required speed and altitude and the short period mode excited by making a sharp stepped control input to the longitudinal control with collective fixed, a control displacement of approximately 25 mm (one inch) should be employed.

(ii) Dynamic displacements can bring about severe stresses in the rotor head, blades and control system. The use of stepped inputs should be approached with caution and with untried rotorcraft, the effect of smaller inputs investigated before embarking on prolonged testing.

(iii) The rotorcraft configurations, speeds and altitudes which shall be tested are the same as those required for the longitudinal manoeuvring stability tests of para 7.6 and may also be combined with the lateral/directional short period stability tests of Chapter 904.

7.8 LONGITUDINAL TRIAL CHANGES

7.8.1 The tests required will largely depend on the rotorcraft under test. The overall effect of C of G changes on handling and trimming will arise from the test programme as defined above and in general there are fewer aerodynamic items employed in rotorcraft than with aeroplanes.

7.8.1 Changes of Trim due to In Flight Configuration Changes. Testing shall include the measurement of the changes in longitudinal control force and longitudinal trim during the following (where appropriate) configuration changes at representative speeds (minimum three) throughout their respective flight envelopes and operating ranges at low (below 5000 ft.) altitude.

(i) Undercarriage, up to down and back to up.

(ii) Airbrake(s)

(a) 'In' through all intermediate positions to fully 'out' and similarly back to 'in'.

(b) 'In' directly to fully 'out' and similarly back to 'in' if feasible.

(iii) Main Cabin Doors and Loading Ramps.

Closed, open singly and multiples where appropriate

(iv) Stores Drops

The effect of stores dropping where the longitudinal trim can be affected shall be investigated within the dropping or firing envelope.

(v) Trailed Role Equipment

(a) M.A.D. The effects of deployment and recovery

(b) Underslung loads or rescue hoists. Testing shall include the effect of load swinging.

(vi) Transfer to Alternate Control Modes

Intentional engagement/disengagement of any part of the longitudinal control system e.g. autostabilisers, autopilot, revision from power to manual control or standby systems etc.

For each configuration change the rotorcraft shall initially be in trim in the datum condition using power for level flight. The required configuration change shall then be initiated keeping the speed, and where applicable the power, constant and any out of trim force measured. The rotorcraft shall then be retrimmed in the new configuration and the change of trim required recorded.

7.9 ROTORCRAFT WITH MANUAL FLIGHT CONTROL SYSTEMS OF MANUAL REVERSION

7.9.1 If the rotorcraft has manual flying controls, the above tests are equally applicable but it will be necessary to confirm that for any trimmed condition the forces can be trimmed out to an acceptable level for sustained operation and that at the limits of forward speed, corrective control inputs can be made without undue force being necessary. Rotorcraft with manual reversion capability shall demonstrate by test that, following reversion, control can be maintained without excessive force being necessary and that when reversion occurs it is not accompanied by an unacceptably sharp cyclic control position change and that residual forces can be reduced to an acceptable level by use of the trimmer.

7.10 EFFECT OF SYSTEM FAILURE ON LONGITUDINAL CONTROL CHARACTERISTICS

7.10.1 The effect of a failure of any system associated with the longitudinal control of the rotorcraft which can be incurred by a single component failure shall be assessed over the flight envelope appropriate to the system under consideration.

(i) Autostabiliser failure.

(ii) Autostabiliser/autopilot runaways.

(iii) Autotrim stabiliser runaway.

7.11 EFFECT OF ACTUATED SURFACES ON LONGITUDINAL TRIM

7.11.1 If the rotorcraft control system includes any actuator driven surfaces, such stabiliser testing shall include assessment of the effect of such surfaces in the flight regimes applicable for their operation. The tests shall include attempts to beat the actuation jacks and the subsequent effect on longitudinal control and trim.

7.12 FLIGHT IN TURBULENCE

7.12.1 The level of turbulence required will normally be stated in the Rotorcraft Specification. To assess the ease of control in turbulence, the rotorcraft shall be flown in level flight in the conditions specified and at aft C of G loadings in rotorcraft configurations 6.2.2, 6.2.3 and 6.2.4 at speeds up to V_{max} for flight in turbulence or the maximum speed the pilot considers to be acceptable from handling or ride considerations.

(Definitions for subjective assessment of turbulence levels are called up in Leaflet 900/4 Table 1).

LEAFLET 903/1

LONGITUDINAL TRIM, STABILITY AND CONTROL

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 903, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time Base
2	Manual Event marker
4	Indicated Airspeed
5	Altitude (Pressure)
6	Radio Altitude
7	Pitch Attitude
10	Roll Angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
18	Normal Acceleration
24	Fuel Contents
25	Stick Position - Pitch
26	Stick Position - Roll
27	Yaw Control Pedal Position
28	Stick Force - Pitch
29	Stick Force - Roll
30	Yaw Control Pedal Force
44	Throttle Position
45	Engine Rotational Speed
46	JPT/TGT
100	Collective Pitch Control Position
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
107	Main Rotor Head Moment
114	Main Rotor Blade Control Load
115	Tail Rotor Blade Control Load
124	Tail Rotor Drive Shaft Torque

CHAPTER 904

LATERAL AND DIRECTIONAL TRIM, STABILITY AND CONTROL

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the lateral and directional handling characteristics of the rotorcraft are satisfactory by investigation of the:

(i) Lateral and directional control system characteristics.

(ii) Lateral and directional static stability.

(iii) Lateral and directional dynamic stability - behaviour in the oscillatory modes.

(iv) Lateral and directional trim changes and the ability to trim, including flight under asymmetric power and, where applicable, asymmetric loadings.

(v) Effect of turbulence.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapter 602 and Chapter 603.

3 APPLICABILITY

3.1 These tests are applicable to all classes of rotorcraft and to all types of control system as defined in Chapter 900 para 6.2.1 except where otherwise stated.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS
4.1.1 The test rotorcraft shall be fitted with a yaw vane or yaw indicator and control position indicators for longitudinal and lateral cyclic position, yaw pedal position and collective pitch.

4.2 TEST INSTRUMENTATION
4.2.1 The parameters which should be recorded during the tests are listed in Leaflet 904/1.

4.2.2 The introduction of a control input device capable of applying precise control inputs (amplitude and duration) with good repeatability would be advantageous and should be given serious consideration where the rotorcraft control system allows such a device to be fitted without serious breakdown. Any such system must result in no downgrading of the control system and shall be free of any risk of spurious operation.

5 LOADING

5.1 The tests shall be made at loadings such that W_{max} is covered with a C of G representative of the service use. Testing shall be extended to include the range of practical C of G loadings, forward, aft and lateral.

5.2 If the rotorcraft is capable of carrying external stores, these loadings shall include the most adverse combination of these stores in relation to:

(i) Highest mass.

(ii) Lowest lateral and/or directional stability.

(iii) The most probable combination of stores loading(s) for Service use if this is/these are not covered by (i) and (ii) above.

6 GENERAL TEST CONDITIONS

6.1 ALTITUDE

6.1.1 The tests shall be made at low (commensurate with flight safety) medium and high altitudes appropriate to the flight phase of the rotorcraft under test.

6.2 ROTORCRAFT CONFIGURATION AND SPEED RANGES REQUIRED FOR TESTS

6.2.1 The tests shall be made over the speed ranges indicated below. In general the tests called up are concerned with accepted forms of rotorcraft lateral and directional control which in the main are rotor derived. Consideration must be given to convertiplanes, compound rotorcraft and tilt rotors which may include aerodynamic controls and/or differential engine thrust. Should such aircraft be presented for test, advice must be sought from the Rotorcraft Project Director and the requirement reviewed.

TABLE 1

Ref.	Flight Regime	Landing Gear	Ref Volume 1 Leaflet 600/1 Flight Config Applicable	Speed Range
6.2.1	Take-off	Down	C	Hover to Vy
6.2.2	Hover	Up and down	C	Hover to 30 kts Vlat, Vfwd and Vaft to specified maximum
6.2.3	Cruise	Up	A and B	30 kts to 1.11 VNe
6.2.4	Manoeuvre	Up	A and B	30 kts to VNe
6.2.5	Approach	Down	C	VNo to hover
6.2.6	Landing	Down	C	Hover

7 TESTS

7.1 Before the tests given in the following paragraphs are commenced, the breakout forces, circuit frictions and backlash of the yaw and roll control systems shall be measured on those systems where this is appropriate. The control force/displacement characteristics over the whole range shall also be measured. Unless otherwise stated tests shall be made with auto stabilisers engaged.

7.2 LATERAL AND DIRECTIONAL CONTROL DURING TAKE OFF

7.2.1 Tests to assess rotorcraft handling during take off are discussed in Chapter 902 and it is implicit in the text that the assessment shall include lateral and directional control. Tests are required to determine the optimum lateral trim position and rudder pedal position to result in the minimum disturbance during the process of lift off while the application of collective pitch is taking place.

7.2.2 Continuous instrumentation records showing control position, attitude and power application are required at neutral or normal operating loading, also the fore and aft and lateral limits shall be assessed.

7.2.3 Testing shall be extended to determine the effect of mistrim. Take off records which the loading and C of G conditions called up above are required with progressive increments of lateral trim applied before commencing take off until unacceptable handling characteristics are experienced or trim limits are reached. The above tests will be expected to demonstrate that in the event of mishandling leading to an out of trim take off, the situation can be recovered without risk to the rotorcraft.

7.3 STATIC LATERAL DIRECTIONAL STABILITY

7.3.1 Trimmed Flight Control Positions. Tests are required to determine the variation of lateral cyclic stick position, yaw pedals position, A.S.E. actuators, fuselage roll attitude and sideslip with increasing airspeed. Tests shall cover the following flight conditions:

TABLE 2

Condition	Speed Range	Altitude
(i) Cruise	30 kts I.A.S. to 1.11 Vne	2000 ft, 5000 ft Max Specified.
(ii) Hover	0 - 30 kts fwd and laterally 0 - max. specified aft	I.G.E.
(iii) Max.power climb	Best climbing speed	Through 2000 ft
(iv) Powered descent and autorotation	Best descent speeds	Through 2000 ft

The above flight conditions to be recorded at 10 knot intervals (where applicable).

7.3.2 Steady Heading Sideslips. A series of tests is required to determine the rotorcraft's lateral static and directional static stability by measuring lateral cyclic and yaw pedal positions and forces with increasing angles of sideslip at a series of equivalent airspeeds. Roll angle shall also be recorded to provide indication of the sideforce characteristics of the rotorcraft.

The tests shall be made for the following flight conditions:

TABLE 3

Condition	Test Point	Altitude
(i) Cruise	(a) Min power level speed (b) VNe - 20 kts (c) Mid range between (a) and (b)	2000 ft and 7000 ft
(ii) Approach	(a) Min power speed, 1000 ft/min descent (b) Recommended approach speed/descent rates if significantly different	Through 2000 ft.
(iii) Max Power Climb	Best climbing speed	Through 2000 ft.
(iv) Autorotation	Best descent speed	Through 2000 ft.
(v) Quartering flight		

Tests are required measuring control positions in the Hover O.G.E. and in forward and lateral flight up to 30 knots. Rearward flight shall be assessed up to max. specified rearwards speed. Testing shall be extended to cover other relative flight directions where problem areas are identified from the handling assessment programme.

NOTE: The selection of minimum power speed, approach speeds and descent rates will be, to some extent dependent upon the outcome of performance measurements and approach techniques.

7.3.3 Turns on One Control. A series of tests is required to assess the lateral control characteristics. The tests will require turns on one control using lateral cyclic and yaw pedals alone. The test conditions are those called up for the Single Heading Sideslip tests (para 7.3.2) using the following test procedures.

(i) Turns on Lateral Cyclic Alone. The rotorcraft shall be stabilised at the required trimmed flight conditions and records taken of control positions. A turn shall then be initiated using lateral cyclic alone while maintaining constant collective, rotor R.P.M. and yaw pedal position commencing with a bank angle of 20° holding the turning condition for approximately 20 seconds using longitudinal cyclic to maintain constant airspeed. Roll out to stable level flight upon completion. Turns shall be carried out in both directions and the test continued to achieve four or five test points in each direction up to bank angles within the rotorcraft's capability and intended role.

(ii) Turns on Yaw Pedals Alone. The rotorcraft shall be established at the required trimmed condition and records taken of control positions. A turn shall then be initiated using rudder pedals alone maintaining constant collective, rotor R.P.M. and lateral cyclic position. The yaw pedal input shall be applied in the form of a steady ramp and the roll response carefully noted starting with small inputs, taking care that sideslip limits are not exceeded. Longitudinal cyclic may be needed to maintain airspeed.

7.4 DYNAMIC STABILITY TESTS

7.4.1 The dynamic lateral directional stability tests are normally carried out in forward flight to evaluate the lateral directional oscillations (Dutch Roll), spiral stability and adverse/proverse yaw. The tests have a strong qualitative nature and pilot impressions regarding the ease or difficulty with which it is possible to suppress these responses are required but instrumentation records shall also be acquired. Gust response characteristics are evaluated both in the hover and in forward flight.

7.4.2 Dutch Roll Mode. Tests are required to provide assessment of the rotorcraft's lateral directional stability response characteristics. The rotorcraft shall be established in trimmed level flight at the desired altitude and airspeed and the flight conditions recorded. The rotorcraft shall then be disturbed from the trimmed condition by the application of a lateral control input (25mm (one inch) held for 0. 5 seconds) and controls returned to the trimmed condition. The tests shall be extended to include recovery from yaw displacements also release from steady heading side slips returning all controls to trim in a sharp deliberate movement.

7.4.3 Spiral Mode. Tests are required to confirm that the rotorcraft has satisfactory spiral stability characteristics in keeping with the role of the rotorcraft. (An attack machine or requirement for high roll agility may not he compatible with convergent spiral stability). The overall stability standard to which the rotorcraft is to be tested should arise from the specification.

7.4.4 Adverse/Proverse Yaw. Cyclic only turns can be conducted to determine the adverse or proverse yaw characteristics and the capability of the rotorcraft to perform coordinated pedals fixed and free turns.

(i) Cyclic only turns shall be initiated from a zero sideslip, wings level attitude maintaining constant collective and airspeed.

(ii) Turn reversals shall be undertaken from a roll stabilised zero sideslip turn (nominal 30° roll).

Both the above manoeuvres shall be accomplished with lateral displacements of various rates consistent with the role of the rotorcraft. Constant collective and airspeed shall be maintained throughout turns to the left and right.

7.4.5 Gust Response and Flight in Turbulence. The level of turbulence for which clearance is required will normally be stated in the Rotorcraft Specification. To assess the ease of control and to determine the rotorcraft's attitude and rate changes (in all axes) in response to gusts, flight shall be undertaken progressively to the conditions specified in level flight and over the practical range of C of G loadings discussed in para 5. Gust response evaluation will involve the determination of the rotorcraft's reaction to lateral and directional disturbances. The evaluation shall lead to assessment of the ability to maintain a desired flight path in turbulence and its effect on the structural integrity of the rotorcraft.

The above assessments can be supplemented by pulse input tests (cyclic, collective or yaw pedal). Definitions for subjective assessment of turbulence levels are called up in Leaflet 900/4 Table 1.

7.4.6 Control Response Tests. Tests are required to evaluate the control sensitivity (deg/sec/inch), control attitude response (deg/sec/inch) control attitude response (deg/inch one second after a unit step input) and cross coupling. The tests shall be conducted by the application of control step inputs working up in magnitude ¼, ½ and one inch) recording a time history of the resultant response. A qualitative opinion of the response taking the proposed role of the rotorcraft into consideration is an essential part of the assessment.

7.5 LATERAL TRIM CHANGES

7.5.1 The tests required will largely depend on the rotorcraft under test. The overall effect of C of G changes on handling and trimming will arise from the test programme as defined above and in general there are fewer aerodynamic items employed in rotorcraft than with aeroplanes.

7.5.2 Changes of Trim Due to In Flight Configuration Changes. Testing shall include the measurement of the changes in lateral control force, trim and response resulting from the following (where appropriate) configuration changes at representative speeds (minimum three) throughout their respective flight envelopes and operating ranges at low (below 5000 ft.) altitude.

(i) Undercarriage, up to down and back to up.

(ii) Airbrake(s).

(a) 'In' through all intermediate positions to fully 'out' and similarly back to 'in'.

(b) 'In' directly to fully 'out' and similarly back to 'in' if feasible.

(iii) Main cabin doors and loading ramps.

Closed, open singly and multiples where appropriate.

(iv) Stores drops.

The effect of stores dropping where the lateral trim can be affected shall be investigated within the dropping or firing envelope.

(v) Trailed Role Equipment

(a) M.A.D. The effects of deployment and recovery.

(vi) Transfer to Alternate Control Modes.

Intentional engagement/disengagement of any part of the lateral control system e.g. autostabilisers, autopilot, revision from power to manual control or standby systems etc.

For each configuration change, the rotorcraft shall initially be in trim in the datum condition using power for level flight. The required configuration change shall then be initiated keeping the speed, and where applicable, the power, constant and any out of trim force measured. The rotorcraft shall then be retrimmed in the new configuration and the change of trim required recorded.

7.6 ROTORCRAFT WITH MANUAL FLYING CONTROL SYSTEMS OR CAPABLE OF MANUAL REVERSION

7.6.1 If the rotorcraft has manual flying controls, the above tests are equally applicable but it will be necessary to confirm that for any trimmed condition the forces can be trimmed out to an acceptable level for sustained operation and that at the limits of forward speed, corrective control inputs can be made without undue force being necessary. Rotorcraft with manual reversion capability shall demonstrate by test that, following reversion, control can be maintained without excessive force being necessary and that when reversion occurs it is not accompanied by an unacceptably sharp cyclic control position change and that residual forces can be reduced to an acceptable level by use of the trimmer.

7.7 EFFECT OF SYSTEM FAILURE ON LATERAL CONTROL CHARACTERISTICS

7.7.1 The effect of a failure of any system associated with the lateral control of the rotorcraft which can be incurred by a single component failure shall be assessed over the flight envelope appropriate to the system under consideration.

(i) Autostabiliser failure.

(ii) Autostabiliser/autopilot runaways.

(iii) Autotrim stabiliser runaway.

7.8 EFFECT OF AERODYNAMIC LATERAL CONTROL SURFACES

7.8.1 Where the rotorcraft incorporates aerodynamic control surfaces such as ailerons to achieve lateral control as may be the case in a compound rotorcraft or where the rotor is offloaded by the introduction of stub wings, testing shall include assessment of these controls to confirm that they are effective in the flight regimes for which they were designed and that rotor derived lateral control is not in any way downgraded by the presence of such surfaces.

7.8.2 It can be generally accepted that assessment of aerodynamic controls will arise as part of the general programme detailed in this chapter but the testing must be expanded to include investigation into low speed flight where the effective roll control power is reduced and the wing sections or sponsons in which the controls are mounted are in a stalled or near stalled condition. Tests shall be included to assess high angles of attack and descent cases, autorotation, sideslip and aileron flutter.

LEAFLET 904/1

LATERAL AND DIRECTIONAL TRIM, STABILITY AND CONTROL

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 904, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time Base
2	Manual Event marker
4	Indicated Airspeed
5	Altitude (Pressure)
6	Altitude (Radio altimeter)
9	Pitch Attitude
10	Roll Angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
18	Normal Acceleration
24	Fuel Contents
25	Stick Position - Pitch
26	Stick Position - Roll
27	Yaw Control Pedal Position
28	Stick Force - Pitch
29	Stick Force - Roll
30	Yaw Control Pedal Force
31	Pitch Trim Position
32	Roll Trim Position
33	Yaw Trim Position
44	Throttle Position
45	Engine Rotational Speed
46	JPT/TGT
100	Collective Pitch Control Position
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
107	Main Rotor Head Moment
114	Main Rotor Blade Control Load
115	Tail Rotor Blade Control Load
124	Tail Rotor Drive Shaft Torque

CHAPTER 905

DEMONSTRATION OF LIMITS OF FLIGHT AND MANOEUVRE ENVELOPES

1 OBJECT

1.1 This chapter describes the tests to be conducted to demonstrate the handling characteristics of the rotorcraft at the limits of the flight and manoeuvre envelopes to which the rotorcraft is released.

1.2 The object of the tests is to show that control about all three axes is acceptable whilst operating the rotorcraft to its declared limits of height, speed and manoeuvrability.

1.3 The manoeuvrability of the rotorcraft shall be demonstrated in such a way as to cover the following factors:

(i) Maximum sustained and transient turn rate.

(ii) Minimum turning radius.

(iii) Rolling capability (maximum roll rate and roll reversal capability).

(iv) Maximum positive and negative load factors.

(v) Agility (ability to change flight path easily and quickly and accelerate and decelerate rapidly).

(vi) Acrobatic and role manoeuvres.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2, Chapters 600, 601, 602, 603, 604 and 605.

NOTE: These requirements are in the process of being written and currently only exist in draft form, if at all. Consequently some revision of this chapter may be necessary when the final standard is approved.

3 APPLICABILITY

3.1 These tests are applicable to all classes of rotorcraft.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 The test rotorcraft shall be fitted with a pitch and roll attitude indicator, control position indicators for longitudinal and lateral cyclic, yaw pedal and collective pitch, and normal and lateral 'g' indicators. In addition any instruments which are required in order to comply with envelope limitations, e.g., cruise guide indication or main rotor rotating control load indicator, should be available.

4.2 TEST INSTRUMENTATION

4.2.1 The parameters which should be recorded are listed in Leaflet 905/1.

5 LOADINGS

5.1 Tests shall be made covering the following loadings:

Wmax/fwd, Wmax/aft, Wmax/neut, Wmax/left, Wmax/right

X, X1, X2
Y, Y1, Y2
Z, Z1, Z2

as defined in Chapter 900, para 5.1.

5.2 Operational loadings, where the carriage of external stores or equipment may produce changes in the handling characteristics, shall also be demonstrated. This demonstration will probably take place as part of the clearance of the particular role fit and will be subject to any restrictions on the flight envelope imposed by that role.

6 LIMITING FACTORS

6.1 The main factors from which the limitations imposed upon the manoeuvre capability of the rotorcraft are derived are as follows:

6.1.1 STRESS LIMITATIONS

(i) To prevent overstressing parts of the rotorcraft and to ensure reasonable fatigue lives of components. This can also include vibration limitations on equipment.

6.1.2 PERFORMANCE LIMITATIONS

(i) The rotorcraft's manoeuvre envelope may be limited by the maximum power available. This means that no further 'g' loading can be achieved unless the airspeed or height are reduced during the manoeuvre.

6.1.3 HANDLING QUALITIES LIMITATIONS

(i) Due to a number of possible factors, for example: insufficient control margins, poor control response, high control forces, vibration or even poor airspeed indications. The deficiencies should be evaluated in relation to the role of the rotorcraft.

6.1.4 EFFECT OF VIBRATION ON CREW PERFORMANCE LIMITATIONS

(i) See Chapter 501, para 2.3.

6.2 Whilst performing the tests described in this Chapter the rotorcraft will be operated to one or more of the above limiting factors. Sufficient information shall be presented to the test pilot to enable him to fly close to the appropriate limit or limits without undue probability of exceeding them. In most cases this will mean providing a cockpit mounted instrument(s) to display a parameter(s) representing the component(s) of the rotorcraft which may become stress limited or which enables such components to be protected. Standard cockpit instruments would normally be sufficient for performance monitoring and control position indicators and a normal 'g' meter for handling qualities assessment. An alternative, mainly for monitoring of stress limitations is to use a telemetry link to a ground station, where technical specialists can monitor the progress of the trial and advise the pilot, via a two way radio link, appropriately.

7 MANOEUVRE ENVELOPE PRESENTATION

7.1 The presentation of the manoeuvre envelope to the pilot in such a way that it covers all the permutations and combinations of variables and at the same time is easily understood and memorised is extremely important. In order to achieve the latter some simplification is generally necessary and this invariably results in the rotorcraft being over restricted in some areas. Therefore, if the full potential of the rotorcraft is to be realised, but the manoeuvre envelope presentation is not to be over complex then a delicate balance needs to be achieved.

7.2 It shall be demonstrated during the conduct of the tests in this chapter that the manoeuvre envelope presentation:

(i) Is easily understood and interpreted by the pilot.

(ii) Provides the necessary information in a form which allows the rotorcraft to be operated as closely as possible to the appropriate limits.

(iii) Allows the pilot to operate within the manoeuvre envelope using the standard cockpit instrumentation with the minimum of subjective assessment i.e., limits which rely on the subjective assessment of vibration, handling characteristics, etc., should be avoided.

8 GENERAL TEST CONDITIONS

8.1 ALTITUDE

8.1.1 Initial tests shall normally be done at a minimum safe altitude, say 2,000 ft. The tests shall then be extended to cover three datum altitudes ranging between minimum and maximum permissible. For particular roles additional emphasis may be necessary at certain heights e.g., for the low level attack role a demonstration at high density altitude may be required where world wide operations are specified.

8.2 LOADING

8.2.1 Initially tests will be carried out at a low or moderate weight and neutral C of G. The tests shall then be extended to cover the full range of operational and permissible loadings described in para 5.

8.3 WEATHER

8.3.1 Early tests should be carried out in smooth conditions but tests shall also be made to demonstrate the effects of turbulence.

8.4 ROTOR SPEED

8.4.1 If it is permissible to operate the rotorcraft at rotor speeds other than the normal optimum speed, or if there is no rotor governing system, or there is a manual throttle reversion capability, then the tests shall cover the range of rotor speeds over which the rotorcraft could be flown.

8.5 SYSTEMS

8.5.1 Where more than one mode of operation of a rotorcraft system is available by selection or occurs due to a single failure and where this may affect the handling of the rotorcraft then the tests shall demonstrate that the handling qualities are not significantly reduced. Where a reduction in the manoeuvre envelope is declared following a system failure, the tests shall show that, if the failure occurs whilst operating at the limits of the full envelope, then the rotorcraft can be recovered to within the limits of the reduced envelope, without undue difficulty or risk. Subsequent operation within the reduced envelope shall show acceptable handling qualities commensurate with the type and severity of the system failure. Examples of the type of systems modes and failures which shall be demonstrated are:

(i) Powered controls - deselection or failure resulting in partial power or manual control.

(ii) Stick feel - deselection or failure.

(iii) Control trim - deselection or failure.

(iv) Cockpit instruments - failure of which may leave the pilot without the required reference whilst operating close to a manoeuvre envelope limit e.g., 'g' meter, artificial horizon, etc.

(v) Actuated aerodynamic control surfaces - failure of which affects the handling qualities of the rotorcraft.

8.6 PILOTING POSITION

8.6.1 If the rotorcraft is fitted with more than one piloting position a representative number of tests shall be made from each position to demonstrate acceptable ergonomic characteristics. In particular, adequate control margins, view and cockpit layout shall be available in all piloting positions.

8.7 AUTOSTABILISATION

8.7.1 If the rotorcraft is fitted with autostabilisation or autopilot equipment intended to be in general use within the role then the majority of testing should be carried out with it in operation. The extent to which testing will be conducted without full autostabilisation will depend upon the nature of the system.

8.7.2 A rotorcraft with a simplex system shall require full investigation throughout the envelope without stabilisation and a thorough demonstration of the effects of system failures. However, if the system has multiple lanes and a high degree of lane redundancy and system integrity, then comprehensive tests will be required with one lane inoperative, but only brief checks with the system completely inoperative.

8.7.3 Because of the wide variety of possible system configurations, the degree and extent to which testing is to be carried out in other than the normal mode of operation shall be agreed with the Rotorcraft Project Director.

9 TESTS

9.1 The following tests shall be demonstrated for all of the conditions listed in para 8, where appropriate. The tests shall be conducted throughout the permissible flight envelope with the majority of test points around the extremes of the envelope but with sufficient points elsewhere to ensure an adequate coverage of the whole envelope.

9.2 MAXIMUM SUSTAINED AND TRANSIENT TURN RATE

9.2.1 The rotorcraft shall first be established in steady level flight at the required speed and altitude. Basic characteristics and the proximity of limiting factors are to be determined and can then be used as a datum (control margins, power available, vibration levels, etc.)

9.2.2 Level turns shall then be flown, to left and right increasing the bank angle incrementally until a limiting factor is reached.

9.2.3 If, as may occur with certain types of rotorcraft, no limiting factor is reached even at high bank angles and load factors, then the inability to hold height is insufficient justification alone for imposing manoeuvre limits. However, the disorientation factors and the degree of pilot capability required to carry out the manoeuvre should be taken into account.

9.2.4 When the rotorcraft's capabilities, characteristics and limitations have been determined during sustained level turns then more complex turning manoeuvre tests shall be conducted. These tests shall cover the whole spectrum of turns in climbs and descents with various power settings and including autorotation conditions. Limiting rotor speeds or very high stress and load factors may be generated particularly during descending, high speed, high bank angle manoeuvres and these conditions should be approached with caution.

9.2.5 Where the rotorcraft's manoeuvre envelope allows the use of transient limits which are higher than those for sustained use then tests shall be carried out to demonstrate these.

9.3 MINIMUM TURNING RADIUS

9.3.1 During the turning manoeuvre tests of para 9.2 turning performance measurements shall be made. This will enable the calculation of minimum turning radius and maximum turn rate for sustained and transient limit manoeuvres. This information shall be presented as a supplement to the manoeuvre envelope and as such should conform to the presentation requirements of para 7.

9.4 ROLLING CAPABILITY

9.4.1 The maximum roll rate and roll reversal capability of the rotorcraft shall be demonstrated during the following tests.

(i) From stabilised straight and level flight at the required speed and density altitude roll the rotorcraft smoothly to reach the sustained limits defined in para 9.2. Repeat the test a number of times progressively increasing the rate of roll until a limiting factor is reached. Rolls in each direction shall be demonstrated.

(ii) From stabilised straight and level flight at the required speed and density altitude carry out a series of roll reversals. Commence the roll in either direction and then reverse it at a pre-determined bank angle within the sustained limits defined in para 9.2. When the same bank angle in the opposite direction is reached reverse the roll again. Complete the manoeuvre by checking the roll when straight and level. Repeat the test a number of times progressively increasing the rate of roll and the rate of reversal until a limiting factor is reached. The manoeuvre shall also be repeated starting in the opposite direction. The bank angle at which the reversals are made shall also be increased until the sustained limits defined in para 9.2 are reached.

9.5 MAXIMUM POSITIVE AND NEGATIVE LOAD FACTORS

9.5.1 Where positive load factor limits are not achieved during the turning manoeuvre tests of para 9.2 then these shall be demonstrated, if possible, by other manoeuvres e.g., if turning manoeuvres were generally limited by roll handling difficulties it may be possible to reach positive load factor limits during straight diving pull ups.

9.5.2 Negative load factor limits shall be demonstrated by means of the manoeuvre most suitable to the type of rotorcraft. The most common manoeuvre used will undoubtedly be the straight 'push over' although the precise details i.e., initial climb angle, rate and magnitude of forward cyclic 'push over', collective movements, etc., will be dictated by the ability of the rotorcraft. If the negative load factor limit for a particular rotorcraft is determined by systems limitations e.g., loss of hydraulic pressure or fuel pressure at low or negative load factors, then sufficient instrumentation shall be provided to enable operation to these limits.

9.6 AGILITY

9.6.1 The agility of the rotorcraft i.e., its ability to change flight path easily and quickly and accelerate and decelerate rapidly, shall be demonstrated in accordance with its intended role and the design specification requirements. Because of the wide variation in abilities and requirements precise details of the proposed demonstration shall be agreed in advance with the Rotorcraft Project Director.

9.6.2 Examples of the types of manoeuvre to be demonstrated are: accelerations from the hover, quick stops to the hover, translations to and from the hover with respect to sideways and rearwards flight, spot turns, acceleration from vertical climbs into forward flight, speed and attitude changes in forward flight, etc.

9.7 AEROBATIC AND ROLE MANOEUVRES

9.7.1 Where the rotorcraft specification demands, the aerobatic and specific role manoeuvre capability of the rotorcraft shall be demonstrated. Due to the type and role specific nature of this testing precise details shall be agreed in advance with the Rotorcraft Project Director.

10 DATA REQUIRED

10.1 The following data shall be obtained during the manoeuvre tests of paras 8 and 9.

10.2 CONTROLLABILITY

10.2.1 Data concerning the ease with which the rotorcraft can be flown:

(i) Control position
- or all steady state and transient conditions.
- any abnormal control movements required particularly during transient manoeuvres.
- frequency and magnitude of control movements necessary to hold steady flight conditions.
- control margins during all conditions.

(ii) Response to control movements
- effectiveness of controls.
- unusual lags, displacements or cross-coupling effects.
- qualitative assessment of rotorcraft reaction to each of the various controls.

(iii) Control forces
- magnitude of control forces.
- magnitude of control forces following a system failure e.g., hydraulics.
- effect of control forces on controllability.
- ease of trimming out forces.
- harmonisation of controls.
- effectiveness of friction devices.

(iv) Manoeuvre limit indications
- ease with which declared manoeuvre limitations can be adhered to.
- ease with which appropriate instruments may be interpreted.
- suitability of instruments and their markings.

(v) Turning performance
- measurements should be taken during manoeuvre tests.
- data should be analysed to quantify minimum turning radius and turn rate for sustained and transient manoeuvres.

10.3 VIBRATION CHARACTERISTICS

10.3.1 Quantitative data on vibration levels and frequencies shall be recorded during all conditions. In addition a qualitative assessment of vibration shall be carried out from all pilot and crew stations. The results of the assessment shall take the form of the scale detailed in Leaflet 1016/1.

10.4 FUSELAGE ATTITUDE AND VIEW

10.4.1 Information about changes in fuselage attitude and field of view from all piloting positions during manoeuvres shall be noted.

10.5 CREW COMFORT

10.5.1 Comments on noise levels, cockpit/cabin conditioning, seat comfort, pilot/crew member security, restriction of arm, leg and head movements and other ergonomic factors.

10.6 ENGINE AND ROTOR BEHAVIOUR

10.6.1 Data concerning engine/rotor performance and handling:

- Effectiveness and behaviour of the rotor speed governor.
- Power required and proximity of limits.
- Rate of engine acceleration/deceleration to meet demands.
- Tendency for the engine to surge or flame out.
- Behaviour of engine/transmission oil system temperatures and pressures.

10.7 ADDITIONAL DATA

10.7.1 Any other data relevant to the assessment, e.g.,:

- Psychological factors such as disorientation, blade flicker, etc.
- Obvious pressure errors in air data systems.
- Comments on cockpit layout.

LEAFLET 905/1

DEMONSTRATION OF LIMITS OF FLIGHT AND MANOEUVRE ENVELOPES

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 905, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
3	Crew speech
4	Indicated airspeed
5	Altitude (pressure)
6	Altitude (radio altimeter)
7	O.A.T.
8	Angle of attack
9	Pitch attitude
10	Roll angle
11	Sideslip angle
12	Heading
13	Pitch rate
14	Roll rate
15	Yaw rate
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
24	Fuel contents
25	Stick position (pitch)
26	Stick position (roll)
27	Yaw control pedal position
28	Stick force (pitch)
29	Stick force (roll)
30	Yaw control pedal force
31	Pitch trim position
32	Roll trim position
33	Yaw trim position
44	Throttle position(s)
45	Rotational speed(s)
46	Engine turbine gas temperature
51	Fuel flow
100	Collective pitch control position
101	Collective pitch trim position
102	Collective pitch stick force
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
106	Tail rotor pitch

Item	Parameter
107	Main rotor head moment
108	Tail rotor head moment
109	Main rotor blade flap bending
110	Main rotor blade lag bending
111	Main rotor blade torsion
112	Tail rotor blade flap bending
113	Tail rotor blade lag bending
114	Main rotor blade pitching moments/rotating control loads
115	Tail rotor blade pitching moments/rotating control loads
116	Main rotor blade pitch angle
119	Tail rotor blade pitch angle

2 TELEMETRY

2.1 The use of telemetry during the tests called up in Chapter 905 is highly recommended. This will allow real time monitoring of all critical parameters, during periods of testing when the aircrew are fully occupied with the immediate task of achieving the required manoeuvre, whilst operating very close, or at, the limits of the rotorcraft. Expert advice can then be made available to the aircrew, either during or immediately after, the manoeuvre. If telemetry is not used then critical parameters should be available for dedicated on-board monitoring by a member of the aircrew.

2.2 The following parameters are recommended for telemetry transmission to a ground station. Details of the parameter ranges etc., are given in Leaflet 900/2, Table 2.

Item	Parameter
1	Time base
2	Manual event marker
3	Crew speech
4	Indicated airspeed
5	Pressure altitude
7	Pitch attitude
8	Roll angle
15	Lateral acceleration
16	Normal acceleration
17	Pitch control position
18	Yaw control position
19	Roll control position
100	Collective pitch control position
102	Input torque
103	Main rotor head moment
104	Tail rotor head moment
105	Main rotor blade torsion
106	Tail rotor blade torsion
107	Main rotor blade flap bending
108	Main rotor blade lag bending
109	Main rotor rotating control load
110	Tail rotor rotating control load
111	Tail rotor drive shaft torque

3 GROUND EQUIPMENT

3.1 An anemometer and wind vane should be used to provide local windspeed and direction data for airfield manoeuvres.

CHAPTER 906

ENGINE HANDLING AND ROTOR GOVERNING

1 OBJECT

1.1 The object of the tests in this chapter is to assess engine handling and rotor governing under all ground and flight conditions for which the rotorcraft has been designed and/or which are likely to be required in Service.

1.2 The tests shall demonstrate that:

(i) All cockpit mounted, engine and rotor related, controls and instruments provide a satisfactory ergonomic interface between the pilot/crew and the engine(s)/rotor during all normal and emergency procedures.

(ii) Each engine installation has satisfactory characteristics with regard to stability, response and handling to power and/or speed demands originating from pilot or rotorcraft requirements during all normal and emergency procedures.

(iii) Engine response to increase or decrease of power demands meets the specification requirements for time needed to achieve a given power change whilst maintaining rotor speed within given limits.

(iv) Rotor speed is maintained within specification limits for all power on conditions occurring during all normal and emergency procedures.

(v) For multi-engined installations the power matching is satisfactory for all normal operating procedures and meets the engine or rotorcraft specification requirements.

1.3 The objectives of this chapter are complimentary to those of Chapter 1001, Engines (Installations). In many cases the tests described in this chapter will be similar to those called up in Chapter 1001* and an integrated test programme would be feasible. There are also many cases where the tests of this chapter can be integrated with rotorcraft handling tests called up elsewhere in Part 9.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2

Chapter 107, Pilot's Cockpit, Controls and Instruments
Chapter 600 and Leaflet 600/1, General Flying Qualities
Chapter 606 and Leaflet 606/1, handling During Autorotation and Failure of One Engine
Chapter 605, Rotor Speed Control (in preparation)

*Chapter 1001 exists in skeleton form only for rotorcraft (JAC Paper No. 1091) but when fully prepared it will be rationalised with this Chapter.

3 APPLICABILITY

3.1 The tests detailed in this chapter are applicable to all new engine installations in rotorcraft, and all installations where modifications have been made likely to affect the results of the tests unless otherwise stated.

3.2 When a rotorcraft is powered by a number of engines of the same type with similar features, the tests detailed in this chapter should be made on one typical engine with repeat tests, as considered necessary, on the remainder. With rotorcraft, engines function as an input to a transmission system which subsequently leads to main rotors, tail rotor (where applicable) and numerous accessory drives. As a consequence multi engines in rotorcraft do not generally have the same independence of operation which can be assumed in aeroplane installations and any testing on individual engines has to take multi engine operation into consideration. This is particularly important where a characteristic is acceptable for single engine operation but could adversely affect multi engine torque matching, or multi engine response. With common intake, shared power off-take, or any other installational feature such that the operation of one engine affects the operation of another, it shall be established that operation of one engine does not produce undesirable effects on the operation of the other (change in response, surge, etc.)

3.3 When tests to this chapter have been made on prototype or development engines, it will usually be necessary to repeat the programme in whole or in part on production engines, to ensure that production methods and modification standards have not degraded the handling characteristics of the engines to an unacceptable degree. The amount of re-testing required will depend on the nature of any modifications incorporated and the variations in the handling characteristics encountered.

3.4 The need to flight test modifications introduced after C.A. Release shall be considered on the merits of each case. However, degradation of engine handling characteristics may result from the cumulative effects of modifications which individually do not justify flight tests. This shall be borne in mind when considering the possible effects of any modification and, where any doubt exists, an abbreviated programme of tests shall be undertaken.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 Cockpit instrumentation shall be capable of monitoring accurately I.A.S., altitude, attitude and sideslip angle, engine R.P.M.(s), J.P.T. or appropriate turbine station temperature, engine torque, main rotor R.P.M., intake temperature and O.A.T. Special test instrumentation may be needed if it is considered that the standard cockpit instruments are inadequate.

4.2 TEST INSTRUMENTATION

4.2.1 Test instrumentation shall be capable of continuously recording the test conditions. Recommended parameters are listed in Leaflet 906/1.

4.3 SPECIALISED TEST EQUIPMENT

4.3.1 If the engine is fitted with automatically controlled bleed from the compressor, or automatically controlled inlet guide vanes, which may affect engine handling, consideration shall be given to substituting a manual control for specific tests. Manual selection of reversionary control modes shall be considered if the engine is controlled by an automatic system and tests are required while the engine is under reversionary control.

5 LOADING

5.1 The tests shall be made at any convenient loading, providing this does not result in the achievable flight envelope being more restrictive than the maximum envelope envisaged for the Service rotorcraft (i.e., maximum normal acceleration, attitude, speed, etc.

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions shall be representative of the Service role of the rotorcraft and the range of climatic conditions for which clearance is required. However, for any test where loss of power will lead to a forced landing, the test must be done in weather conditions such that an engine out landing can be safely made.

6.2 Tests shall be made within the operational limits for the engine stated in the engine Technical Certificate (production engines). The authority of the Engine Project Director shall be obtained before doing any test that would involve exceeding these limits.

6.3 When a rotorcraft is to be cleared for the use of more than one type of fuel, tests shall include the use of each type of fuel for which clearance is required unless otherwise agreed. Tests on the extremes of the specific gravity and/or temperature of the primary fuel shall be made and may be extended to include the secondary fuels.

6.4 Initial tests of a new engine installation shall be carried out with the test rotorcraft on a tie-down facility. This will enable a fairly complete range of tests to be conducted, including the use of high power, responses to power demands, stability, etc., before the engine/rotorcraft combination is flown.

6.5 The majority of the testing shall be carried out in reasonably smooth weather conditions to enable a true assessment of engine/rotor response to control movements to be made without any external influences. However, sufficient testing in turbulent conditions shall be conducted to appraise system behaviour.

6.6 The airframe shall be fully representative of the in-Service standard. If this is not so, the Rotorcraft Project Director shall be informed of the deficiencies and the probable effect these may have on any C.A. Release. Testing shall include role equipment items such as sand filters, F.O.D. guards, anti-icing intakes and I.R. suppression, where the use of such items has an effect on engine handling or rotor governing.

6.7 ENGINE AND ROTOR SETTINGS AND ADJUSTMENTS

6.7.1 Prior to the tests the engine(s) and rotors shall be set to within the current limits applicable to the engine(s) and rotors under test. Where tolerance on settings is permitted, e.g., on rotor minimum pitch, bleed valves and variable inlet guide vanes, etc., which affect engine handling and rotor governing, tests shall be made at the upper and lower limits of these tolerances.

6.8 EMERGENCY POWER SUPPLIES

6.8.1 The rotorcraft shall have adequate alternative power supplies in the event of engine flame-out or shutdown to permit controlled flight while relight attempts are made, or an engine out landing is carried out. In particular there must be adequate electrical supplies to operate the test instrumentation once an engine has been shut down.

7 TESTS

7.1 The following tests shall be carried out over a matrix of speeds and density altitudes to cover the permissible flight envelope and the full range of altitudes, temperatures, wind velocities to cover the engine starting and operating ground envelope.

7.2 ENGINE STARTING ON THE GROUND

7.2.1 Engine starting shall be carried out in accordance with the standard procedures as laid down in the engine/rotorcraft operating manual.

7.2.2 Starts shall be made using all starting power source alternatives both internal and external in origin and, if applicable, with degraded power sources. If an internal storage power source (e.g., battery) is incorporated then a simulated double missed start followed by a successful start shall be demonstrated.

7.2.3 If starting is permissible following the failure of any system or systems which would normally be required to be operational during starting then starts shall be made under simulated failure conditions to demonstrate this capability.

7.2.4 Due to the variable nature of engine starting and the range of operating conditions possible, a considerable number of starts of each type/condition shall be made so that the spread of starting characteristics can be determined. In order that these characteristics can be fully analysed precise details of the type of start, engine control selections and any external influences such as wind speed/direction, precipitation, icing conditions, etc., shall be noted on each start.

7.3 ACCESSORY OR ANCILLARY DRIVE OPERATION

7.3.1 Where the rotorcraft transmission system incorporates means of driving accessories or ancillary systems without driving the rotor system then engine operation in this mode shall be investigated.

7.3.2 Stability, response and handling of the engine shall be assessed throughout, the range of speeds and powers which are permissible. In particular acceleration of the engine between the ground idle speed (G.I.) and normal operating speed for this mode of operation shall be assessed for varying rates up to the maximum allowable or achievable.

7.3.3 The effect of switching accessory loads on engine operation shall also be investigated. This is particularly important in the case of cyclic load switching, such as might occur with anti-icing heating loads, if it is envisaged that these will be operated in accessory drive.

7.3.4 Assessment shall include identification of the effect of failure of any interlinks intended to protect against rotor engagement with the transmission loaded. The sequence for engaging main drive shall also be identified including any possibility of misuse which could lead to transmission damage. Recommendations shall be made to avoid such possibilities.

7.4 ROTOR ENGAGEMENT AND OPERATION ON THE GROUND

7.4.1 Engagement of the engine/rotor and acceleration to normal operating speed shall be assessed using the recommended procedure. Various rates of engagement and acceleration shall be applied up to the maximum permissible or attainable. Similarly deceleration of the engine/rotor from normal operating speed to ground idle speed shall be assessed. Acceleration/deceleration reversals at varying rates and with a variety of starting and ending speeds shall also be conducted. For multi-engined installations acceleration and deceleration of each engine in turn between ground idle and normal operating speed shall be assessed whilst the rotor is running at normal operating speed, powered by the other engine or engines.

7.4.2 If such controls are provided, the ease of making speed trim adjustments over the range allowable and the behaviour of the engine(s)/rotor following rapid trim changes shall be assessed with all engines driving at M.P.O.G.

7.4.3 For multi-engined installations the facilities and procedures for matching engine powers at M.P.O.G. shall be assessed for ease of use and range of mismatch which can be accommodated.

7.4.4 Engine/rotor behaviour during other normal ground operations shall also be investigated. In particular the effect of taxying shall be observed up to limiting speeds and turn rates and in wind velocities within the ground envelope.

7.5 TAKE OFF, LANDING, HOVER AND LOW SPEED MANOEUVRES

7.5.1 Engine/rotor behaviour during take off shall be assessed. Vertical take off with varying rates of collective application up to maximum permissible or obtainable and covering the full range of allowable wind velocities shall be included. Also running take-offs over a range of groundspeeds and crosswinds to cover the ground/flight envelopes.

7.5.2 Engine/rotor behaviour in the hover shall be assessed in a range of wind velocities to cover the flight envelope. The effect of height above ground shall also be assessed covering a range of heights from just above the ground to a full O.G.E. hover.

7.5.3 Low speed manoeuvres including; translational flight over a range of speeds and relative directions, spot turns at various rates, and accelerations from and decelerations to the hover at various rates, to cover the low speed flight envelope shall be investigated to assess engine/rotor behaviour.

7.5.4 Engine/rotor behaviour during landings shall be assessed. A range of probable landing conditions shall be investigated to cover the low speed flight envelope. These shall include; vertical landings at various rates and covering the full range of allowable wind velocities and run on landings at various groundspeeds and crosswinds. Also in the assessment will be a series of baulked landings where collective pitch will be re-applied to 95% of Maximum Continuous power at various rates and at various stages during a vertical landing.

7.6 IN FLIGHT ASSESSMENT OF ENGINE CONTROLS

7.6.1 In steady flight an ergonomic and functional assessment of engine controls shall be conducted.

7.6.2 The position, layout and labelling of all controls and instruments, including warning devices shall be assessed for ergonomic efficiency bearing in mind the role of the rotorcraft and the relative importance of each item during normal and emergency procedures. Each control shall be operated (where practicable) over its full range of movement and assessed for the ease with which it enables its designated function to be carried out. Each instrument and warning device shall be assessed for logical presentation of the information it is designed to convey and for ease of viewing and assimilation of that information by the pilot(s).

7.6.3 The effective range of engine speed/rotor speed trim controls and engine power matching controls if available shall be determined in steady level flight.

7.7 ENGINE/ROTOR GOVERNING IN FORWARD FLIGHT

7.7.1 Engine/rotor governing static droop laws shall be identified for a range of forward flight speeds from zero to Vne. Recordings of rotor speed, torque, collective position, altitude, O.A.T., I.A.S. and engine parameters are to be made over a range of engine powers from zero (engine/rotor speeds just joined) to Maximum Contingency power (single engine installations) or Maximum Continuous/Take-Off power (multi engined installations) in increments of 5%. The tests shall be carried out starting at one end of the range and incrementing towards the other end only. They shall then be repeated incrementing in the opposite direction. This will identify any hysteresis in the governing or the operation of bleed control valves, I.G.V's, etc. No adjustments are to be made to engine/rotor speed trim or engine power matching controls during the tests.

7.7.2 For multi-engined installations individual engine droop laws are to be obtained, where practicable, as well as for all combinations of more than one engine, from zero to Maximum Contingency Power.

7.7.3 Engine/rotor governing characteristics shall be investigated through the forward flight envelope including banked turns to limiting bank angles, climbs and descents to limiting rates and sideslips to limiting angles.

7.8 GOVERNOR RESPONSE AND SYSTEM STABILITY

7.8.1 The engine/rotor governor response characteristics throughout the power range shall be determined by using a series of ramp inputs of collective pitch (in both directions) using a wide range of rates.

7.8.2 The tests shall be conducted at datum powers from zero (engine/rotor speeds just joined) to Maximum Contingency power (single engine installation) or Maximum Continuous power (multi engined installation).

7.8.3 Airspeed for the tests shall be minimum power speed with selected tests repeated at Vmin and Vne.

7.8.4 The amplitude of collective ramp inputs shall be 10% of the Maximum Continuous power.

7.8.5 For multi engined installations the tests shall be repeated for individual engines (if practicable) as well as for all combinations of more than one engine but with at least one engine inoperative. The range of powers tested shall be between zero and Maximum Contingency power.

7.9 ENGINE ACCELERATION/DECELERATION IN FORWARD FLIGHT

7.9.1 The following tests are to be conducted at minimum power speed and repeated, where practicable, at Vmin and Vne.

7.9.2 Deceleration of each engine from the governed flight condition to ground idle shall be carried out using the engine condition control. Acceleration back to the governed flight condition including engagement of freewheel/clutch shall also be assessed. Rates of acceleration/deceleration shall cover the range from slow and cautious to maximum permissible or achievable. In the case of single engined rotorcraft, or multi engined rotorcraft that cannot maintain height with one engine inoperative, these tests should be approached with caution and sufficient height should be allowed to attempt a re-light if the engine flames out. Preparations for a forced landing should be made before commencing the tests.

7.9.3 Acceleration and deceleration of the engine(s) in the governed flight condition using the collective control shall be carried out. For all engines operating inputs are to be applied to achieve a power change of 95% of Maximum Continuous power at increasing rates until one of the following limits is reached.

(i) The rate of power change required by the Rotorcraft Specification is achieved.

(ii) The permitted transient rotor r.p.m. is reached.

(iii) Engine surge or other limitations are encountered.

(iv) Rotorcraft handling difficulties are encountered.

Acceleration tests are to commence from zero power (engine speed/rotor speed just joined) and deceleration tests from Maximum Continuous power. The state of the engine(s) and rotor at the time the acceleration or deceleration is started can be critical, consequently it is important that the starting conditions are noted carefully so the results may be correlated. Similarly instrumentation recordings should be started at least 10 seconds before each test and sufficient time should be allowed after the collective movement has ceased so that the response and damping of the engine(s)/rotor may be assessed.

7.9.4 In the case of single engine installations the tests of para 7.8.3 shall be carried out over a power range of 95% of Maximum Contingency power. This shall also apply to additional tests on multi engined installations for single engine operation (if practicable) and all combinations of more than one engine but with at least one engine inoperative.

7.9.5 In the event that the engine/rotorcraft specification does not detail acceleration and deceleration rates for the tests of para 7.9.3 and 7.9.4 the following shall be used as a guideline for tests at I.S.A. sea level.

Rotorcraft Category	Time for 95% Max. Continuous Power	Time for 95% Max. Contingency Power
Light (up to 5000 lb AUW)	1 second	2 seconds
Medium (up to 10,000 lb AUW)	2 seconds	3 seconds
Large (above 10,000 lb AUW)	3 seconds	4 seconds

These rates will also be dependent on the intended role and consideration shall be given to this. Where doubt exists the rates to which testing is to be carried out shall be agreed with the Rotorcraft Project Director.

7.10 ENTRY TO AND RECOVERY FROM AUTOROTATION

7.10.1 On rotorcraft where split off of engine(s) speed and rotor speed on entry into autorotation may be achieved by use of collective control (as opposed to retardation of engine(s) speed by engine condition control) the handling characteristics of engine(s) and rotor in the transition region shall be investigated.

7.10.2 The tests of this section shall be carried out at the recommended autorotation speed or minimum power speed and repeated at Vmin and Vne.

7.10.3 Entries into autorotation shall be carried out from initially level flight, and eventually in the climb at 95% of Maximum Continuous power. Collective shall be reduced to minimum, or the required setting for autorotation, at increasingly higher rates until one of the following limits is reached:

(i) The rate required by the Rotorcraft Specification is achieved.

(ii) Engine surge or other limitations are encountered.

(iii) Rotorcraft handling difficulties are encountered.

7.10.4 If rates for entry into autorotation are not declared in the Rotorcraft Specification the values quoted in para 7.9.5 plus one second should be used as a guideline. This rate covers entry into autorotation from 95% Maximum Continuous power to engine speed/rotor speed split off. The same conditions as applied in para 7.9.5 also apply i.e.:

(i) Rates are for I.S.A. sea level.

(ii) Rates are dependent on rotorcraft role.

(iii) If any doubt exists rates shall be agreed with the Rotorcraft Project Director.

7.10.5 Recoveries from autorotation shall be carried out from an engine speed/rotor speed just joined condition and to the same rates and conditions as apply in para 7.9.5. In essence, therefore, the tests will be the same as the engine acceleration tests of para 7.9.3 and 7.9. 4. However, an assessment of engine speed and rotor speed handling in autorotation and during the engine/rotor speed matching phase just prior to the acceleration shall be included.

7.10.6 The tests in this sub para are complimentary to the tests of Chapter 907; Autorotation, Partially Powered Flight and Engine Off Landing.

7.11 SIMULATED ENGINE FAILURE

7.11.1 The tests of this sub para are to determine engine and rotor behaviour following an engine failure in a multi engine installation. Engine failure in a single engine installation or complete power failure on a multi engine installation with appropriate collective lever delays are covered by Chapter 907, para 7.5.

7.11.2 The following tests shall be conducted at minimum power speed and a safe height with repeats at Vmin and Vne. They should be considered as a preliminary to the tests of Chapter 907, para 7.7, in order to assess engine/rotor behaviour in less critical conditions.

7.11.3 Initial tests should be conducted at light weight but the final tests shall be at maximum A.U.W.

7.11.4 From a stabilised condition a failure of one engine is simulated. This is normally achieved by retarding the engine condition control to ground idle as rapidly as possible on the 'failed' engine. However, if this is not practicable, or a realistic simulation of engine failure, e.g., run down rate too low, then an alternative method shall be employed. This could take the form of an existing fuel cut off valve sited close to the engine fuel inlet or a similar unit specifically installed for the test. The exact method used would depend on the installation but a realistic simulation of engine failure must result.

7.11.5 Following initiation of the 'failure', engine(s) and rotor behaviour shall be observed without, if possible, moving the engine condition control or collective. However, if it is necessary to move these controls to avoid exceeding transient engine or rotor limits, the delay time between initiation of the 'failure' and pilot intervention shall be noted. Unless otherwise stated in the Rotorcraft Specification a minimum delay of two seconds shall be demonstrated for the most adverse combination of conditions.

7.11.6 The above test shall be carried out at a range of three or four representative powers from low to maximum continuous power at the start i.e., with all engines operating.

7.11.7 The tests shall be repeated with each engine in turn simulating the failure.

7.11.8 During the above tests a full assessment of any failure warning devices shall be carried out.

7.12 IN FLIGHT ENGINE STARTING

7.12.1 Engine(s)/rotor handling and behaviour during in flight engine re-starting shall be assessed over a fully representative range of engine, systems and flight conditions.

7.12.2 The starts shall be carried out over a range of airspeeds from Vmin to Vne, or the permissible range for in flight starting, as appropriate.

7.12.3 In the case of single engine installations the lowest altitude which shall be assessed is that consistent with safety. Necessary precautions for a possible engine off landing should be made.

7.12.4 All possible combinations of power sources for starting the engine(s) shall be demonstrated.

7.12.5 In multi engined installations the tests shall be repeated for each engine position.

7.12.6 If starting is permissible following the failure of any system or systems which would normally be required to be operational during starting then starts shall be made under simulated failure conditions to demonstrate this capability.

7.12.7 Sufficient number of starts of each type and condition shall be made to enable a representative assessment of the spread of starting characteristics to be made.

7.13 POWER MATCHING

7.13.1 In multi engined installations the facility for achieving and maintaining power matching of the engines shall be assessed. In particular the assessment shall include the ease of use, effectiveness and operating range of any controls or available adjustments associated with the facility.

7.13.2 The assessment shall be made based upon observations taken during the full range of tests carried out in accordance with this chapter and where more than one engine was, or should have been, operating in a matched condition.

7.13.3 The power matching of the engines during all normal operations shall be within the limits declared in the Rotorcraft Specification. In the absence of such a requirement the following limits shall apply:

Steady state mismatch	-	less than 2%
Transient mismatch	-	less than 5%

Percentages are to be related to Maximum Continuous power.

7.14 ENGINE AND SYSTEM MALFUNCTIONS

7.14.1 Safety Aspects. There are any number of malfunctions that may occur in an engine and governing system. This sub-sub-para does not try to cover the malfunctions comprehensively for the precise test programme will depend on the installation and should therefore, be agreed with the Rotorcraft Project Director. However, the obvious points of safety that should be covered for any of these tests are:

(i) That initial flight tests are done under the most favourable conditions of weight, airspeed and weather and over suitable terrain for engine off landings.

(ii) That sufficient height is allowed for the tests bearing in mind recovery from difficult situations.

(iii) That, wherever possible, it is prudent for initial tests to be done on the ground or tie down facility as applicable.

7.14.2 Emergency Engine Control. If an emergency system for control of the engine is fitted the following aspects shall be assessed:

(i) The ease and smoothness of changeover to emergency control and back to normal control over a range of flight conditions and engine powers.

(ii) That in emergency control:

(a) Full control of the engine is retained, i.e., maximum and minimum power can be selected and no lost throttle movement is experienced.

(b) The sensitivity and ease of operation of the control is satisfactory.

(c) The handling of the engine is satisfactory bearing in mind that some engine safety devices may be inoperative. The case of accelerating the engine from a low power condition shall be investigated carefully and thoroughly.

7.14.3 Engine Starting Under Emergency Control. If applicable, engine starts, both on the ground and in flight, shall be made under emergency control. The starts shall be assessed from the points of view of:

(i) The ease and safety of doing such a start.

(ii) The effectiveness of the recommended technique, and any safety precautions which have to be observed.

Sufficient range of conditions shall be covered to demonstrate that the declared ground and flight starting envelope under emergency control is achievable.

7.14.4 Engine Control System/Governor Runaway or Malfunction. The engine control system/governor may be subject to numerous failures or malfunctions of its component parts or sensors which result in a loss of its control function. The effects of such a failure would depend upon the precise details of the installation and, therefore, should be the subject of a Failure Modes and Effects Analysis. Those failures which require ground or flight testing on the rotorcraft should be agreed with the Rotorcraft Project Director and a detailed test plan approved. However, the major effects produced by failures are for the engine to either partially or fully runaway to maximum or minimum throttle, or to freeze in a fixed condition. Testing of such failures shall encompass the following points:

(i) Control freeze - although this case is serious on a single engine installation, there are no tests required other than to prove subsequent ease of establishing emergency control of the system.

(ii) A downward runaway may be tantamount to an engine failure depending upon the system that is used. An upward runaway, particularly from a low power condition, can be serious and might jeopardise the integrity of the engine or cause loss of control in a turbine through blade stall or compressibility effects. Tests may be done initially on tie-down to check the acceleration of an upward runaway by opening the throttle a certain amount at varying rates. These checks on the tie-down and tests in the air may be done with a variable rate runaway box. The tie-down tests will give some indication of the rate of acceleration of the rotor that is likely to occur and the likelihood of surging the engine. Subsequent tests in the air might take the form of starting from two or three selected power settings (one of them being minimum power obtainable in the air) and accelerating the engine by varying amounts at various rates to check the R.R.P.M. acceleration, the steady rotor speed if tolerable, and assess with what speed and to what extent the pilot has to react. Note, the worst case will be acceleration from Flight Idle, and these conditions cannot be fully simulated by tie down tests.

(iii) Similar runaway down checks shall be done in the air at two or three power settings (one being maximum power), the throttle is closed at various rates to check the rotor deceleration and how quickly and to what extent the pilot has to react. For this test the Flight Idle stop, if fitted, will be selected to 'IN'. This test will be similar to and complimentary to the engine failure test.

(iv) On a multi engine installation the ease with which a runaway engine can be correctly identified shall be assessed. (The 'good' engine will always try to compensate for the 'bad' engine and appear to runaway in the opposite sense). Note whether or not the identification can be made without moving the collective lever. If a discriminatory failure warning device is fitted its effectiveness shall also be assessed.

7.14.5 Overspeed Trip. Tests to assess the suitability of the engine overspeed trip are complimentary and similar to the tests outlined in para 7.14.4. The probable sequence of events would be as follows:

(i) Check the engine speed in autorotation and during transient overswing tests to determine whether the overspeed trip R.P.M. would be reached or exceeded during normal operation of the rotorcraft. The most critical case for these tests would be maximum A. U. W. and high density altitude.

(ii) In the air at a safe height, A.U.W., and position, slowly overspeed the engine/rotor (by manual throttle if necessary) and check the correct functioning of the overspeed trip; this can be done from a low collective setting; and engine off landing will almost certainly have to be made so the necessary precautions must be taken. If required, further tests can be carried out at higher power settings.

(iii) Tests will be done in the same fashion for multi engined installations, tie down checks first and then flight tests to check the operation of the overspeed trip.

7.14.6 Other Malfunctions. There are, inevitably, a multitude of other malfunctions that may need to be investigated, many of which are peculiar to the particular installation. These shall be identified to the Rotorcraft Project Director and a detailed test programme agreed.

7.14.7 Engine Systems Failure/Cautionary Warnings. A thorough assessment of all engine associated failure/cautionary warnings shall be made to determine that they are effective and unambiguous. During the tests called up in para 7.14 the requirement for warnings shall be reviewed.

LEAFLET 906/1

ENGINE HANDLING AND ROTOR GOVERNING

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 906, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
5	Altitude (pressure)
7	Ambient temperature
9	Pitch attitude
10	Roll angle
11	Sideslip angle
12	Heading
24	Fuel contents
44	Throttle position(s)
45	Rotational speed(s)
46	JPT or TGT
48	Intake position (s)
51	Fuel flow
52	Fuel temperature
56	Inlet guide vane position
57	Variable stator vane position
68	Relight event
100	Collective pitch control position
101	Collective pitch trim position
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
123	Air intake temperature

2 TELEMETRY

2.1 Telemetry is not normally a requirement for engine handling tests as detailed in Chapter 906, but its use for monitoring critical engine parameters should be considered for tests involving a significant flight safety hazard, high pilot workload, or where real time analysis would reduce the required flight time significantly.

2.2 The following parameters are recommended for use with telemetry if it is decided that this shall be used for some or all of the tests of Chapter 906. Details of parameter ranges etc., are given in Leaflet 900/2, Table 2.

Item	**Parameter**
1	Time base
2	Manual event
4	Indicated airspeed
5	Pressure altitude
20	Throttle position
21	Rotational speed
22	Turbine entry temperature
23	Relight event
100	Collective pitch control position
101	Main rotor speed
102	Input torque

2.3 The parameter lists of para 1.2 and 2.2 are general and it would, therefore, be necessary to adapt the lists to suit the individual engine(s)/rotor(s) systems under test.

CHAPTER 907

AUTOROTATION, PARTIALLY POWERED FLIGHT AND ENGINE OFF LANDING

1 OBJECT

1.1 The object of the tests in this chapter is to investigate the handling of the rotorcraft in the engine off and partially powered cases covering the following:

(i) Entry into autorotation, flight idle descent or partial powered descent.

(ii) Flight in autorotation, flight idle or partial powered descent.

(iii) Recovery to power on, level or climbing flight.

(iv) Landings from the above cases and determination of Height/Velocity data for engine off flight.

(v) Partial power failure in multi engine rotorcraft.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2, Chapter 100, para 9.2, Chapter 304, Chapter 600 and Chapter 606 including Leaflet 606/1.

3 APPLICABILITY

3.1 The tests are applicable to all types and classes of rotorcraft.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 The pilot shall be provided with visual presentation of control positions, sideslip and normal 'g'.

4.2 TEST INSTRUMENTATION

4.2.1 The parameters which should be recorded during these tests are listed in Leaflet 907/1. The use of video to provide a record of all landings shall be considered and a telemetry back up to on board records is desirable.

5 LOADING

5.1 The tests shall be aimed at a weight and C of G representative of service use. The programme shall commence at a light weight and as confidence is gained work progressively up to the maximum as represented by condition X in Chapter 900, para 5.1.

5.2 If the ground clearance for tail rotor is such that there is doubt that a safe margin of clearance during touch down can be maintained, testing shall be extended to include aft C of G.

5.3 The culmination of all the above will be to attempt engine off landings based on the experience and data gained by the previous tests. The initial attempt will be based on the best understanding regarding airspeed and main rotor speed for optimum descent and ground speed at contact.

5.4 Once a number of successful attempts have been completed under the apparent optimum conditions the effects of airspeed and rotor speed on approach and touchdown can be investigated until it is evident that test limits with an acceptable safety margin are being approached.

6 TRANSITION TO AUTOROTATION WITH COLLECTIVE LEVER DELAYS

6.1 Having demonstrated by the preceding tests that autorotation can be entered with confidence from any flight regime and recovery to powered level flight established without unacceptable handling penalties, the next phase of testing will be concerned with introducing a margin to allow a delay in reaction time following an engine failure.

6.2 The aim shall be to allow two seconds to elapse from the occurrence of an engine failure to the point where collective pitch is sharply reduced to establish autorotation, this being an allowance to provide reaction time for an average pilot to identify under favourable conditions that a failure has occurred and then to respond correctly by reducing collective pitch to the minimum.

6.3 Before commencing the delay tests, tests are necessary to establish a minimum rotor speed from which a safe recovery to autorotation for landing can be consistently achieved.

6.4 The lever delay tests shall start from a 'safe' altitude (nominal 2000 ft A.G.L.) at V_{IMP}. Close throttles and observe a brief delay on all controls allowing the main rotor speed to fall to a pre determined value (NR 1) well above the minimum permitted power off rotor R.P.M., then lower collective lever fully. The minimum transient rotor speed (NR 2) and the time taken to recover normal autorotative rotor speed should be noted and confirmed from instrumentation to determine a value of NR (NR 1 - NR 2).

6.5 Testing on the above lines shall continue to increase NR 1 incrementally until the minimum permitted rotor R.P.M. is approached or a collective delay of two seconds is achieved.

6.6 During all the above testing handling characteristics shall be investigated and the effect of autopilot modes included in the assessment.

6.7 The lever delay tests shall continue to cover the range of forward speeds, weight and C of G positions to ensure that autorotation can be safely achieved with an allowable delay in pilot reaction.

7 AVOID AREA TESTS

7.1 Having established a full understanding of the rotorcrafts' characteristics and behaviour through the whole procedure following engine failure through to landing, a series of tests are required to determine the complete avoid area.

7.2 A typical avoid curve and the key points requiring definition is shown in Fig 1. The whole process involved in the investigation of autorotation, delayed response, engine off landing and avoid curve calls for flying close to the rotorcraft's handling and structural limits.

7.3 Stress limits must be established with adequate margins before testing commences and progressive increases in forward speed and altitude can then proceed until these limits are approached.

7.4 Ideally rotor and structural stresses including landing gear should be monitored on telemetry. Video record of each landing is recommended. The sequence of testing defined below is intended to provide a progressive build up in experience and understanding of the autorotative characteristics.

7.5 HIGH SPEED - LOW HEIGHT REGIME

(i) This regime is represented by idents 4, 5 and 6 in Fig 1 and is primarily concerned with rotor inertia during decay and the trade off of speed for lift during the flare. The testing would commence with simulated failures in the very low hover (nominal 1 ft) increasing height and forward speed in small increments until stress limits approached or the handling qualities during the subsequent run on are such that a maximum allowable touch down speed $(V_{TD})_{max}$ can be identified (reference Item 4 in Fig 1).

(ii) Once $(V_{TD})_{max}$ is established testing shall continue with simulated engine failures in low flight at higher speeds with touch down preceded by a quick stop manoeuvre until sufficient test points are achieved to confidently define the high speed - low height regime from $(V_{TD})_{max}$ to the limiting speed or V_{max}.

7.6 LOW SPEED REGIME

(i) The low speed regime is represented in Fig 1 by idents 1, 2 and 3 and it is this phase which concerns itself with the ease with which autorotation can be established and will include the introduction of a delay in reducing collective pitch as established in the tests called up in para 6.

(ii) The maximum height for low hover is represented by ident 2 in Fig 1 or h_{max}, This will entail a progressive increase in height from a 'wheels clear' hover until landing gear or structural limits for the test are approached.

(iii) Once h_{max} has been established with adequate safety margins, test points can be determined with increasing forward speed until the knee for the low speed regime (Ident 3 in Fig 1) can be identified representing as it does the 'lower boundary'.

(iv) The final phase concerns exploration of the 'upper boundary' identifying the minimum height for high power h_{min} and sufficient intermediate points to fully define the low speed regime (Idents 1, 2 and 3).

7.7 GENERAL

(i) Assessment of the avoid area characteristics with its requirement to approach structural and handling limits can be clearly identified as a high risk trial and must be approached with extreme care. The trials should be undertaken in calm conditions with generous altitude and stress; margins at the outset until confidence in the rotorcraft's characteristics is gained.

(ii) The effect of weight and C of G must be investigated and check points included in planning for climatic trials to confirm the effect of temperature and altitude.

(iii) Tests to define the avoid area are mandatory for single engined rotorcraft. In the case of multi engines, the capability for the rotorcraft to achieve autorotation shall be demonstrated and sufficient test work carried out to allow a procedure for engine off landing to be defined.

8 PARTIAL POWER FAILURE

8.1 In the case of a multi engined rotorcraft total power failure represents at least a double failure occurrence in other than isolated cases, such as fuel contamination.

8.2 Most multi engined rotorcraft can be expected to be capable of maintaining height following a single engine failure in level flight. Tests are required to demonstrate the rotorcraft's capability to flyaway following single engine failure in critical flight regimes, e.g., take-off or low hover.

8.3 POWER DOUBLING

(i) An important feature in multi engined rotorcraft is the ability of the engine control system to maintain power for the required flight condition with the good engine or engines in the event of a failure or rundown of any of the power units.

(ii) Tests of this feature are an essential part of the engine control system testing and as such shall be completed in advance of the programme to determine the safety aspects concerned with partial power failure. The tests shall demonstrate that following single engine failure, the remaining power unit/s provide the necessary power contribution in a smooth fashion without unacceptable rotor droop and at a rate which will not result in a severe or sharp loss of height.

(iii) Tests to determine the height loss against weight, altitude and temperature shall be carried out. It is important to demonstrate that following a power failure that there are adequate cues to the pilot to avoid limitations being exceeded on the remaining engine/s. It is to be expected that most rotorcraft will include some form of engine failure warning which will be an important part of this assessment. If such warnings are not available, the requirement shall be reviewed as part of the assessment and recommendations made.

8.4 DETERMINATION OF 'AVOID' AREA

(i) There are operational considerations which make it impossible to remain within the flyaway area at all times e.g., dunking sonar and covert anti tank role.

(ii) Tests shall be carried out to provide data of hover height against weight, altitude and temperature above which, flyaway can be achieved with minimum ground clearance and below which it becomes necessary to land or ditch and which may hazard the rotorcraft. The tests shall be extended to determine an 'avoid' area to include forward speeds.

(iii) Testing shall commence with a series of simulated failures at a safe hover height to determine the height loss arising from a single engine failure and subsequent power doubling reaction to establish partial powered flight. These tests shall be extended to include simulated failures in forward flight up to VNo. From the above programme an 'avoid' area profile can be predicted and testing shall then continue with the intention of demonstrating the 'avoid' profile.

(iv) The final tests shall commence from a safe margin above the predicted safety height and the height reduced by small increments until the stress and handling limits are approached.

Ident Number	Symbol	Defining Phrase
1	h_{min}	- minimum height for high hover (ft AGL)
2	h_{max}	- maximum height for low hover (ft AGL)
3	$(h_{cr})_s$ - $(V_{cr})_s$	- "knee" low speed regime
4	$(V_{TD})_{max}$	- maximum allowable touchdown speed (GS)
5	$(h_{cr})_H$ $(V_{cr})_H$	- "knee" high speed regime (kt IAS)
6	$(h_{min})_H$	- minimum height at V_{Limit} or V_{max} (ft AGL)

1-3-2 Low Speed Regime

1-3 "Upper Boundary" Low Speed Regime

2-3 "Lower Boundary" Low Speed Regime

4-6 High Speed - Low Height Regime

4-5 "Initial Rise" High Speed - Low Height Regime

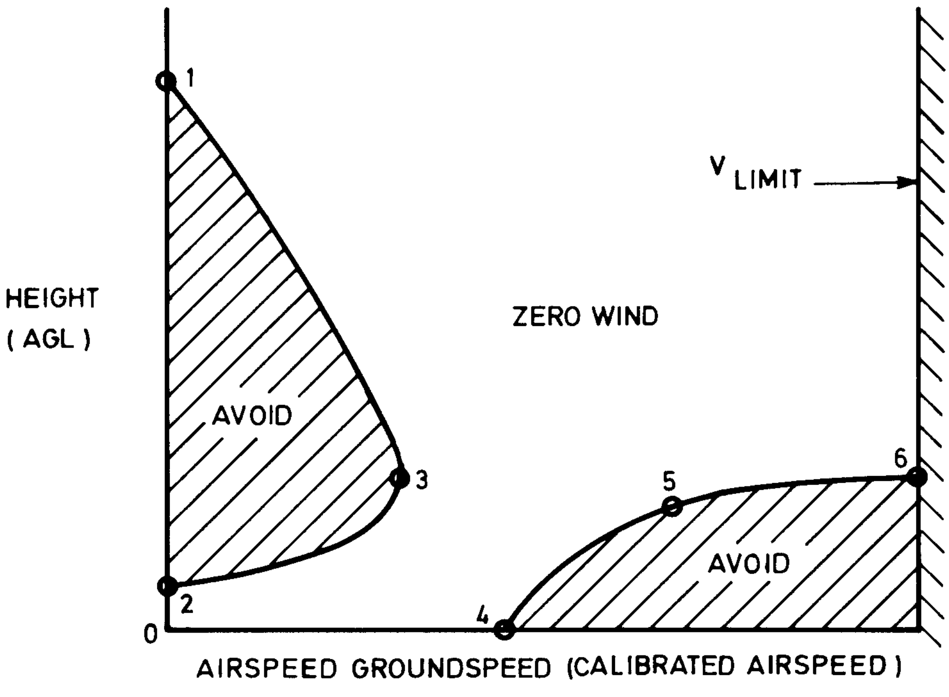


FIG.1 DEFINING PHRASES AND SYMBOLS FOR AVOID CURVE CONSTRUCTION

LEAFLET 907/1

AUTOROTATION, PARTIALLY POWERED FLIGHT AND ENGINE OFF LANDING

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 907, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
5	Altitude (pressure)
6	Altitude (radio altimeter)
9	Pitch attitude
10	Roll angle
11	Sideslip angle
13	Pitch rate
14	Roll rate
15	Yaw rate
16	Longitudinal acceleration
17	Lateral acceleration
25	Stick position (pitch)
26	Stick position (roll)
27	Yaw control pedal position
44	Throttle position(s)
45	Rotational speed(s)
100	Collective pitch control position
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
106	Tail rotor pitch
109	Main rotor blade flap bending
110	Main rotor blade lag bending
122	Landing gear strut extension/compression

1.3 If the use of telemetry is not practical an on board stress monitor showing landing gear stress and possibly main rotor head stress would be advised.

2 TELEMETRY

2.1 The use of telemetry would make it possible to monitor the key parameters as the trial takes place. This has particular relevance for the monitoring of structural stresses during touch down and should accelerate progress.

2.2 If it is decided to fit telemetry for these trials (see para 2-1) then the following parameters should be telemetered to a suitable ground receiving station. Details of the parameter ranges etc., are given in Leaflet 900/2, Table 2.

Item	**Parameter**
1	Time base
2	Manual event
4	Indicated airspeed
5	Pressure altitude
7	Pitch attitude
16	Normal acceleration
17	Pitch control position
18	Yaw control position
19	Roll control position
100	Collective pitch control position

3 GROUND INSTRUMENTATION

3.1 It is recommended that all landings and flyaway runs are recorded with video equipment. Such equipment should include a timebase synchronised with the on board equipment and telemetry capable of providing a time record to within 0.1 seconds.

CHAPTER 908

DECK LANDING

1 OBJECT

1.1 The object of the tests in this chapter is to demonstrate the rotorcraft's capability to operate from ships decks and to assess the handling issues brought about due to:

(i) Deck movement.

(ii) Induced turbulence.

(iii) Enforced out of wind approach and landing.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2, Chapter 304, Leaflet 304/3 and Chapter 1009.

3 APPLICABILITY

3.1 The tests are applicable to all rotorcraft nominated for deck landing in the specification.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 The pilot will require presentation of collective pitch, cyclic pitch and yaw pedal position. A clear calibrated torquemeter must also be available. Tail rotor pitch presentation shall be available in the cockpit, unless it is directly proportional to yaw control position (i.e., no A.F.C.S. or collective interlink).

4.2 TEST INSTRUMENTATION

4.2.1 The parameters which should be recorded during the tests are listed in Leaflet 908/1.

4.3 FLIGHT DECK INSTRUMENTATION

4.3.1 Provision shall be made for the following:

(i) Deck roll and pitch angle.

(ii) Heave (vertical acceleration at the deck).

(iii) Lateral acceleration at deck level.

(iv) Calibrated measurement of the ship's relative wind direction in as near to free stream as possible.

(v) Means of assessing true wind velocity (note - a velocity states speed and direction).

(vi) Means of assessing relative wind direction at deck.

5 LOADING

5.1 Testing shall commence at a medium AUW with a C of G representative of service use and progressing to Wmax and including the practical light weight case. If as the trials progress, it becomes evident that the extremes of C of G can lead to major handling problems, effort shall be made to include some limited assessment of problem areas in the programme.

5.2 The conditions available may mean that the read across to a world wide envelope is too far removed to allow safe extrapolation in which case the Rotorcraft Project Director shall be informed so that a decision regarding the need for further trials can be made.

6 GENERAL TEST CONDITIONS

6.1 Ships trials present a difficult logistic problem and suitable ships are not readily available. However, it is important that a trial period of approximately 2 weeks is put aside with the ship in an area, and at a time of year, that will give a high probability of a range of conditions from calm to sea state 6 or more. During the period when a ship is made available there is a need for a cautious approach commencing with low sea state and calm conditions working up to rough sea and high variable winds. Such ideal arrangements are unlikely to occur and judgement is necessary to take full advantage of the conditions available.

7 TESTS

7.1 GENERAL

7.1.1 Before a full sea trial is undertaken the rotorcraft shall have completed tests covering the following:

(i) Starting, take off, landing and stopping on sloped surfaces.

(ii) Any rolling platform trials required by the programme, if applicable.

(iii) Tests with deck securing devices or equipment.

(iv) Ditching and escape tests, also airborne inflation of flotation gear if applicable.

(v) Hover performance tests.

7.1.2 The tests covered in this chapter are concerned with the handling issues associated with take off, departure, approach and landing from ships decks and platforms. The engineering issues will be the subject of Chapter 1009

7.2 DECK LANDING TECHNIQUE

7.2.1 In current practice, two techniques are used to approach and land on small ships and these will be considered for the purpose of this Chapter and in general are concerned with conventional tail rotor or multi rotor designs. If future development or usage bring about changes in technique or configuration the need for revised testing will have to be considered and agreed with the Rotorcraft Project Director.

7.2.2 The techniques under consideration are as follows:

(i) INTO WIND LANDING (FIG.1a)
The rotorcraft lands facing into the relative wind following an approach from the downwind side of the ship.

(ii) FORWARD FACING LANDING (FIG.1b)
The rotorcraft lands facing forward following an approach to the ship from a windward or leeward direction.

7.2.3 Having decided upon the deck landing technique to be assessed, testing shall commence taking off and landing in relatively steady deck conditions.

7.2.4 Once the basic process of operating from the ships deck has been established the trial shall proceed, accumulating test points over a range of deck motions and relative wind velocities, building up experience in a progressive manner aiming as far as the weather conditions will allow, to approach the more severe conditions towards the end of the trial.

7.3 DECK OPERATION ENVELOPE

7.3.1 The main objective of the trial shall be to produce a Ship Helicopter Operating Limits diagram (SHOL) which will provide guidance over a range of relative wind velocities over the deck with sufficient margin to allow for weight variation. The aim shall be to provide the widest and simplest SHOL possible consistent with aircraft handling penalties and ship/rotorcraft limits.

7.3.2 The SHOL assessment covers the following aspects:

(i) Landings will be undertaken at a value of W/σ as close to the target weight as possible.

(ii) Qualitative assessment of the effect of turbulence with special reference to that generated by the ships superstructure over the vessels speed range and relative wind velocity.

(iii) Assessment of visual cues and deck markings allowing positioning and awareness of the deck and local obstructions.

(iv) Assessment of directional control power margins (e.g. tail rotor power and/or rudder pedal).

(v) Margin of power available to deal with deck motion.

(vi) Night assessment.

7.3.3 Typical SHOL diagrams have been attached for guidance as follows:

(i) Into wind landings (Fig.2).

(ii) Forward facing (Fig.3).

7.4 PILOT RATING

7.4.1 In order to assist in providing in consistent standard of judgement from which recommendations for the Ship Helicopter Operating limits can be determined the following rating scale shall be used:

TABLE 1 A PILOT'S RATING SCALE FOR DECK OPERATIONS

1 2	ACCEPTABLE	SAT	Sufficient power and control in hand Low workload, easy task.
3 4		UNSAT	Safe landings possible, but power and control limits approached or reached. Moderate workload but difficult due to one or more factors.
5	UNACCEPTABLE		Unable to complete sequence without exceeding power or control limits. High workload and difficulty.
5	DANGEROUS		Attempting sequence causes limits to be exceeded. Excessive workload.

7.5 DECK OPERATIONS

7.5.1 In the preparation of the SHOL diagram the following aspects shall be given consideration:

(i) Landing gear should have good energy absorption characteristics and be tolerant of high rates of descent and drift. The rotorcraft's ability to adhere to the deck without restraint shall be assessed and may require assessment of a number of alternative wheel arrangements. If the rotorcraft is provided with the means of applying negative collective pitch to assist in holding the rotorcraft on the deck, testing shall include assessment of this feature over a range of ships movements and wind conditions.

(ii) Agility: Response to controls, particularly collective pitch where small corrections may be necessary to remain clear in the hover over a heaving deck and directional control must be available under all take off and landing conditions. Engine response and power margins must therefore be adequate.

(iii) Gust response: The rotorcraft's response to gusts and turbulence (as in the lee of superstructure) should be convergent and well damped to minimise workload in the hover.

(iv) A.F.C.S.: Landings shall be made with and without stabilisation and the SHOL diagram shall reflect any downgrading which may become evident. In addition, where a single failure results in significantly altered handling characteristics (such as the yaw control reverting to manual) landings at representative conditions shall be made with hydraulics off.

(v) Aircraft Securing System: Where security or recovery system such as harpoon or RAST (Recovery, Assist, Secure and Traverse System) is an essential part of the deck operating procedure, the trial shall not only include a full assessment of normal operation but confirmation that malfunction or misuse of the system does not endanger the rotorcraft or result in unacceptable handling features to recover the situation. The resultant SHOL shall also indicate the rotorcraft's freelanding capability. Any emergency release of such systems should be included in the test programme.

8 THE PREPARATION AND PRESENTATION OF SHOL ENVELOPE

8.1 For a landing to be rated as 'acceptable' the following shall be consistently demonstrable.

(i) The steady state and transient torque limits shall not be exceeded.

(ii) The steady state and transient yaw control margins shall, in general, not be less than 10% and 5% of total control range respectively.

(iii) The above parameters can be quantified by analysis of instrumentation but should be augmented by a pilots subjective assessment of difficulty, so that field of vision, control workload and where applicable, the acquisition of deck securing device can be taken into account.

8.2 Tests shall be conducted as close to a target W/σ as possible and the resultant values of torque and tail rotor power adjusted (based on datum performance measurements) to correct to target weight.

8.3 Once sufficient test points are available judgement will be necessary to allow a small margin for pilot experience and a power margin to deal with ships motion. Complex envelopes should be avoided as they are difficult to use in practice.

8.4 The final envelope will be valid for sea level I.S.A. conditions only. To use it for non standard conditions it will be necessary to apply a correction to the A.U.W. derived from hover performance tests, yaw control margins determined from low speed manoeuvre tests and engine/transmission limits. The resultant diagram will therefore show envelopes for corrected A.U.W. in excess of the permitted A.U.W. for the rotorcraft.

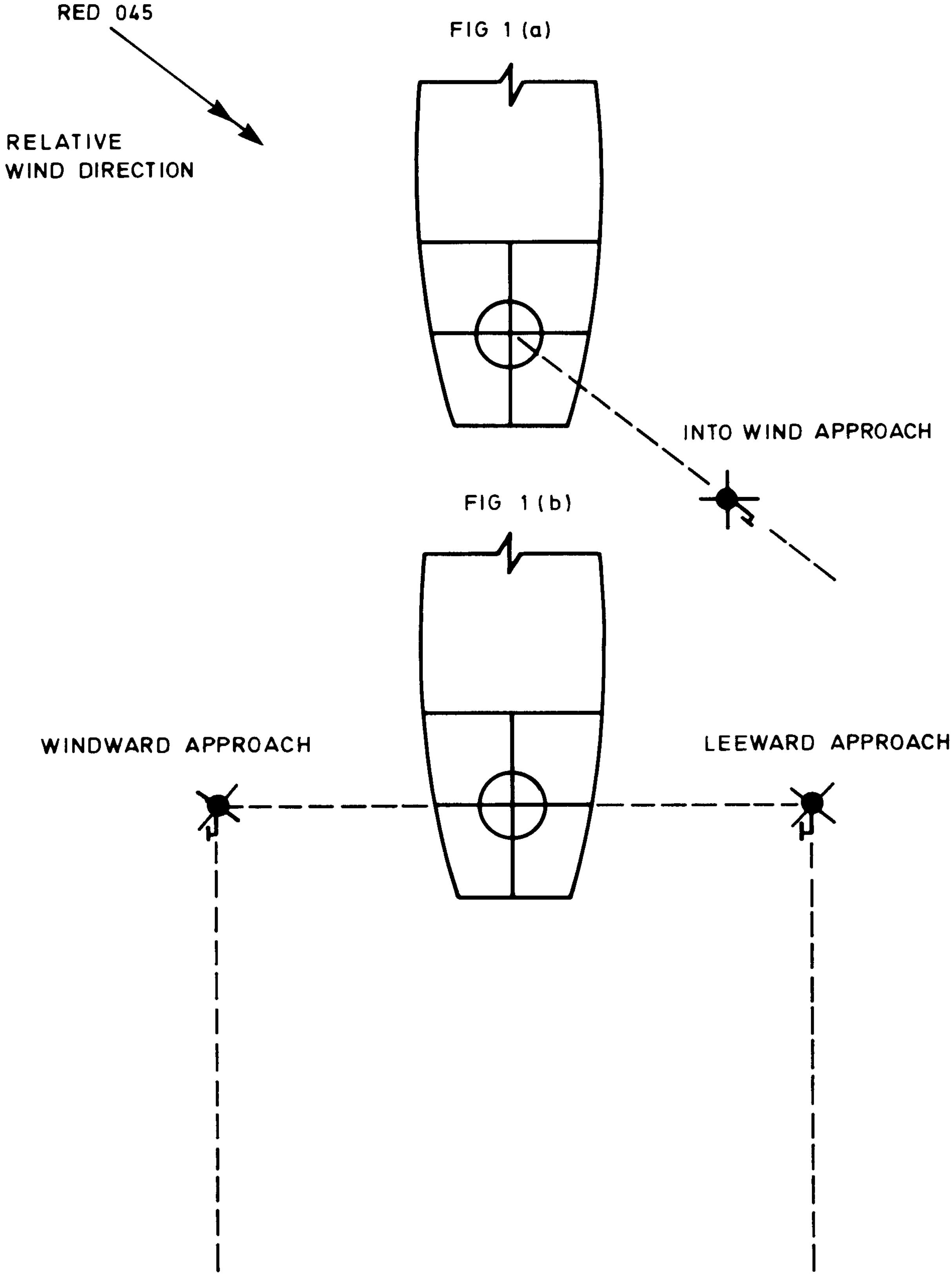
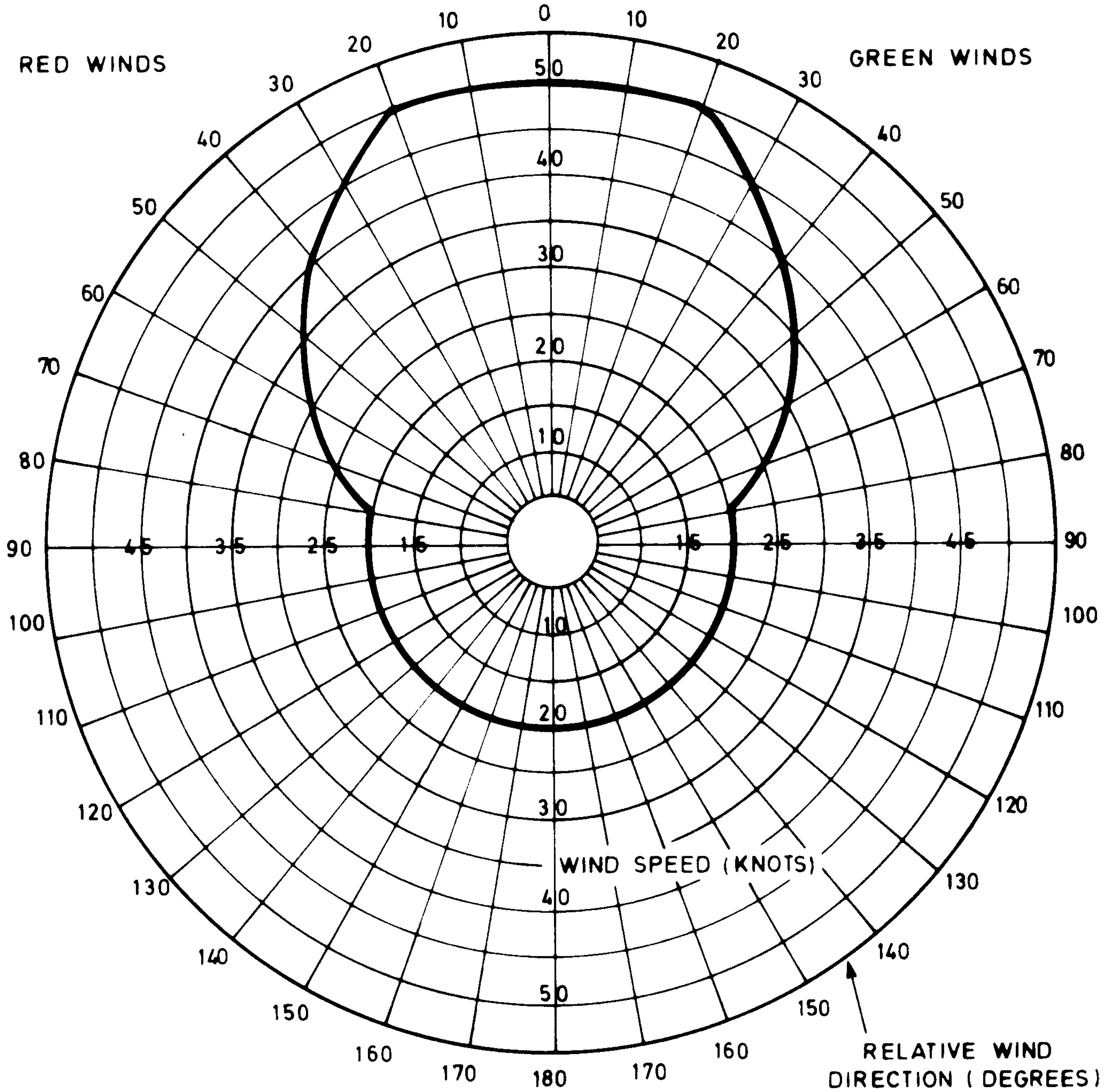


FIG.1 CURRENT APPROACH AND LANDING TECHNIQUES



ISA SEA LEVEL CONDITIONS
ROTORCRAFT LANDS INTO THE RELATIVE WIND
WIND RELATIVE TO SHIPS HEAD
ALL WEIGHTS

FIG.2 TYPICAL SHOL DIAGRAM FOR INTO WIND ENVELOPE

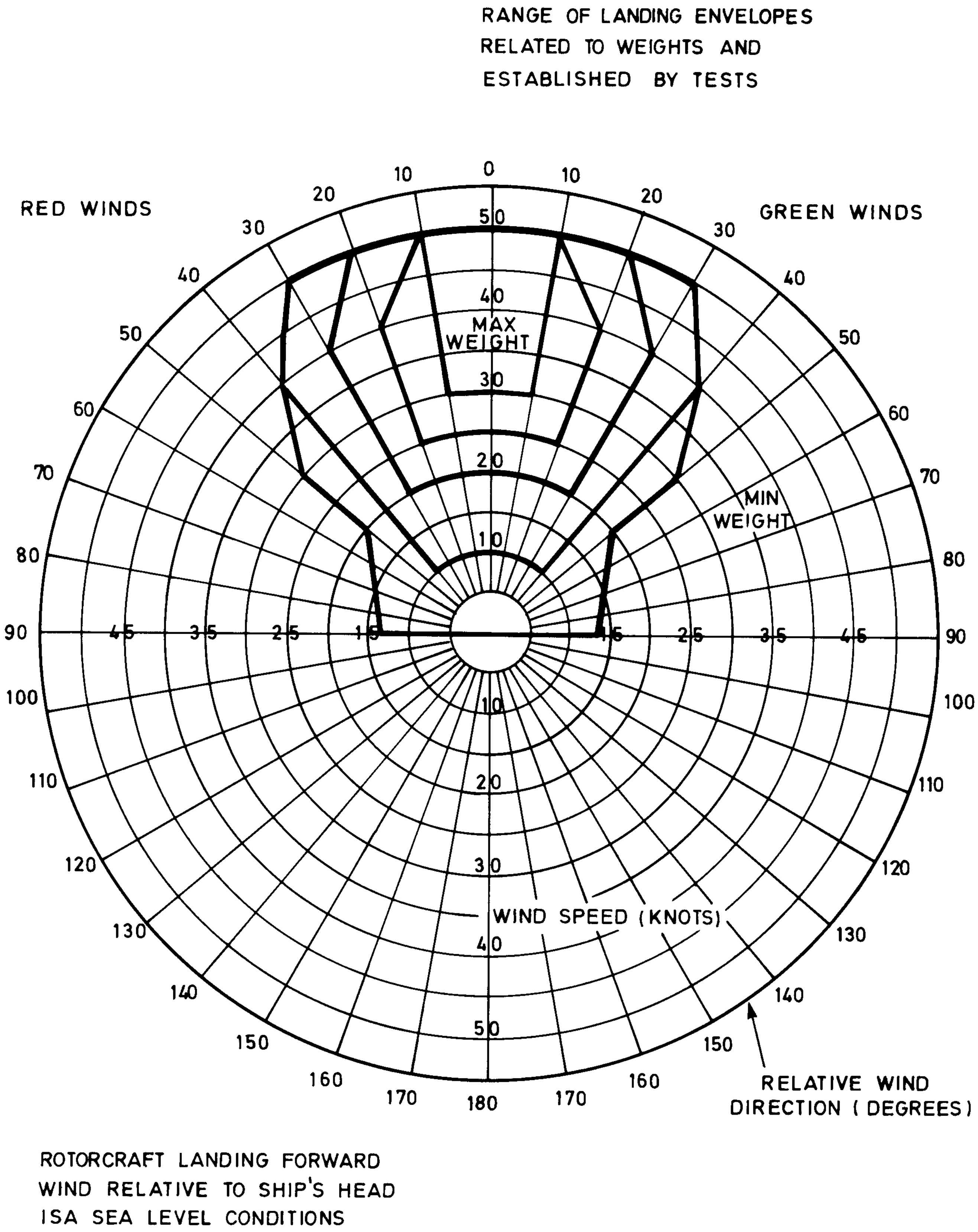


FIG.3 TYPICAL SHOL DIAGRAM FOR FORWARD FACING ENVELOPE

LEAFLET 908/1

DECK LANDING

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 908, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
5	Altitude (pressure)
6	Altitude (radio altimeter) (if available)
9	Pitch attitude
10	Roll angle
12	Heading
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
24	Fuel contents
25	Stick position (pitch)
26	Stick position (roll)
27	Yaw control pedal position
30	Yaw control pedal force
69	Brake temperature (port and starboard)
70	Tyre temperature
71	Nose wheel angle
100	Collective pitch control position
103	Main rotor speed
104	Input drive torque
106	Tail rotor pitch
121	Rotor brake temperature
122	Landing gear strut extension/compression

1.3 Items not currently listed in Leaflet 900/2.

(i) Deck securing device activation.

(ii) Main rotor blade/fuselage structure clearance.

1.4 An onboard stress monitor should be considered to allow safety monitoring of key stresses during the limiting conditions.

2 SPECIAL EQUIPMENT

2.1 Instrumentation to record flight deck conditions is listed in Chapter 908, para 4.3.

2.2 Consideration should be given to including an accurate rate of descent instrument.

CHAPTER 909

AUTOMATIC FLIGHT CONTROL SYSTEMS

1 OBJECT

1.1 The object of the tests in this chapter is to evaluate the performance of the automatic flight control system (AFCS) in providing stability augmentation and automatic control of the rotorcraft.

1.2 In particular each mode of the AFCS will be tested to:

(i) measure performance of the rotorcraft and system in all its modes of operation against the rotorcraft and system specification .

(ii) determine any handling limitations of the rotorcraft with the system engaged and assess the cockpit layout and AFCS controls against the requirements of its roles.

(iii) establish the effects of system malfunction.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970, Volume 2, Chapter 604.

3 APPLICABILITY

3.1 These tests are applicable to all new combinations of rotorcraft and AFCS equipment.

3.2 The scope of the testing required will depend on the modes of operation of the AFCS available, the operational roles for which the rotorcraft is being evaluated and the AFCS safety analysis. The types of AFCS to which this chapter is applicable are only those which employ analogue and/or digital computing.

3.3 Testing of rotorcraft which include Flight Directors is dealt with in para 8.

3.4 In general the term AFCS does not include the rotorcraft power supplies or the primary flying controls. However, for the purpose of safety and tolerance examinations the effects of these systems on the behaviour of the AFCS shall be considered.

3.5 Fly by wire systems or ACT systems may incorporate aspects of stability enhancement and certain autopilot modes, in which case these functions shall be tested as described in this Chapter. The testing of systems which replace conventional controls and or produce some special manoeuvre characteristics will require different and more complex testing than that involved in only AFCS functions. This Chapter is, therefore, limited to showing compliance with Chapter 604.

4 EQUIPMENT

4.1 COCKPIT INSTRUMENTS

4.1.1 In addition to the normal flight instruments the test rotorcraft shall be fitted with an indicating normal accelerometer, a sideslip gauge and indicators of control position.

4.2 TEST INSTRUMENTATION

4.2.1 The parameters listed in Leaflet 909/1 should be recorded. In addition various equipments will be needed to be fitted for the purpose of providing repeatable disturbances to the system under test in order to measure the dynamic behaviour and to simulate those system failures in flight that it has been agreed with the Project Director to test.

4.2.2 Care must be taken when designing such equipments to ensure that the size of the system disturbance is not unintentionally modified by the response to the rotorcraft behaviour.

4.3 SYSTEM CALIBRATIONS

4.3.1 Before tests are started, calibrations of the AFCS in the rotorcraft will be undertaken to confirm that the equipment is within specification and to record the precise values of gearing, time constants etc. It will be necessary to study the effect of production tolerances in flight; limiting the tests to those that simulation show to be critical. Tolerances which shall be considered in addition to those of the AFCS computer are those of actuators, powered flying controls, sensors, indicators etc.

4.3.2 The rig and equipment used in support of the Failure and Effect Analysis required by section 3 para 604 of this document (Safety Assessment Analysis and Test) shall be adequately calibrated before use.

5 LOADINGS

5.1 All modes of AFCS operation shall be tested over a weight and C of G range compatible with the operational roles of the rotorcraft.

5.2 If the carrying of external stores is required in any operational role then practicable combinations of these stores shall be considered when calculating the weights and C of G ranges to be tested.

6 GENERAL TEST CONDITIONS

6.1 ALTITUDE AND SPEED RANGE

6.1.1 The height and speed ranges for the test conditions are those applicable to the rotorcraft roles and their criticality should first be examined by simulation in order to establish the minimum rotorcraft testing necessary to ensure satisfactory performance over the flight envelope. Adequate flight testing in a range of wind speeds and turbulence shall be carried out as well as in operational conditions of visibility, weather and terrain to give confidence that the AFCS is adequate for the roles intended for the rotorcraft.

6.2 ROTORCRAFT CONFIGURATIONS

6.2.1 Testing shall be carried out with any auxiliary lift or moment producing devices, e.g., variable incident tail plane, in the positions most appropriate for the flight conditions or phase under test. Incorrect positions or operation of these devices shall be considered during the failure analysis. The speed range over which the system is tested shall be that applicable to the rotorcraft and AFCS combination.

7 TESTS

7.1 The calibrations carried out under para 4 will give familiarity with the equipment and its modes of operation. This familiarity is important to acquire before starting the following tests, which shall be carried out in all modes of system operation. The particular tests and the conditions under which they will be conducted shall be agreed with the Project Director as shall be the extent and standard of software documentation before flight.

7.2 COCKPIT ASSESSMENT

7.2.1 A preliminary assessment of the layout of the controls and indicators shall be made before flight and also of the ability to engage, disengage and override the AFCS.

7.3 ENGAGEMENT AND DISENGAGEMENT TESTS

7.3.1 It shall be demonstrated that all AFCS modes come into operation smoothly and safely even when engaged in "out of trim" conditions and at different rotorcraft configurations. The details of such demonstrations shall be agreed with the Rotorcraft Project Director. It shall similarly be demonstrated that disengagement can also be made safely and smoothly.

7.4 AFCS CONTROL DURING CHANGES OF CONFIGURATION AND FLIGHT CONDITION

7.4.1 With the AFCS engaged in the appropriate mode of operation the flight conditions shall be varied throughout the ranges of airspeed, altitude, rotorcraft configuration, external stores release etc. The performance of the AFCS shall be determined in each flight case and where appropriate when changing from one trimmed case to another. When an auto-trim system is fitted changes in flight conditions shall be made at varying rates to determine the adequacy of the auto-trim follow-up rate.

7.4.2 The performance and management of the AFCS just prior to and during autorotation shall also be determined.

7.5 SHORT TERM STABILITY

7.5.1 Testing shall be carried out axis by axis at constant speeds and heights throughout the weight and C of G ranges. Repeatable pulses or steps of control displacement should be used to disrupt the rotorcraft so that its short term stability, with and without the AFCS operating can be measured. The AFCS actuators can be a convenient means of providing the repeatable control displacements. The tests shall be repeated in all modes of operation over ranges of conditions appropriate to the mode of use in the operational roles.

7.6 DATUM ACQUISITION AND HOLDING AND LONG TERM STABILITY

7.6.1 The object of these tests is to assess the ability of the AFCS to acquire and maintain the selected datum conditions in each mode, throughout the range of height, speed, rotorcraft loadings and configurations required by the specification and to assess the characteristics of any long term behaviour.

7.6.2 For all modes involving datum acquisition and for hold, the AFCS mode to be assessed shall be selected and the AFCS allowed to seek the datum condition from a range of non steady conditions e.g., barometric or radar height hold shall be selected during climbs and descents at various rates and heading and track acquire and hold from large and small changes of azimuth etc.

7.7 MANOEUVRES

7.7.1 The affect of the AFCS modes of operation on the handling of the rotorcraft shall be assessed. All automatic manoeuvres which the AFCS is required to perform shall also be examined for accuracy of performance and ease of monitoring correct operation over the range of operational conditions of use. Control inputs shall be injected to check for any oscillatory tendencies and to ensure that the actuators remain free from saturation even during turbulent conditions. It will also be necessary to examine the performance of the AFCS in one axis while the pilot manoeuvres the rotorcraft in another e.g., Automatic height holding during accelerated and or turning flight, airspeed hold performance in climbs descents and turns.

7.8 WEAPON AIMING MODES

7.8.1 Where the rotorcraft, together with some mode of the AFCS, is to be used in a weapon aiming role; tests shall be made at the appropriate flight conditions to assess the adequacy of the AFCS and the rotorcraft combination in each of the weapon aiming roles.

7.9 AUTOMATIC APPROACH

7.9.1 Where the AFCS is coupled to a radio approach aid tests shall be conducted to assess the behaviour of the rotorcraft in this mode. The performance shall be assessed on a number of beams of known characteristics and, if possible, a wider range of beam width variations shall be tested.

7.9.2 Measurement of lateral and vertical ILS signal displacements from the beam centrelines and measurements of the flight path of the rotorcraft shall be made down to decision height on all approaches and in all conditions of wind and turbulence envisaged for service use. Where applicable, similar tests shall be made for all other aids e.g., MLS.

7.9.3 Overshoot performance following a decision to overshoot at the decision height shall be assessed with all engines operating and where applicable with one engine inoperative.

7.10 ASW AND OVERWATER SEARCH AND RESCUE

7.10.1 These modes of operation involve accurate coordinated control of the collective and cyclic axes in order to make the transition from an entry gate of 50 to 120 knots at 100 ft to 200 ft to an exit gate of 20 ft to 100 ft at the hover. Because of the difficulty of carrying out this task repeatedly and safely especially at night it is usual to provide modes of operation of the AFCS for this purpose.

7.10.2 Flight testing shall establish that transition performed throughout the required ranges of entry and exit conditions can be carried out easily, consistently and safely with transition profiles being smooth and comfortable and very easily monitored for safety threatening departures from normal. Particular attention shall be paid to the profiles at the end of the transition down to ensure that the height and speed overshoots are kept to very small values and that a comfortable phasing between height changes and speed changes is maintained throughout the manoeuvre.

7.10.3 Short term stability testing and malfunction testing shall be done during both the transition and hover conditions.

7.11 OVERLAND SEARCH AND RESCUE

7.11.1 No special test methods are considered for this role since it is not envisaged that special autopilot modes will be developed for this purpose.

7.12 MALFUNCTION TESTING

7.12.1 The types of malfunction which may occur in any specific AFCS installation shall be determined. In considering the consequence of AFCS malfunctions and in planning the test programme a failure effects analysis shall be carried out to the satisfaction of the Project Director and supported by rig testing, calculation, simulators etc., to determine the effect of the malfunction on rotorcraft behaviour.

7.12.2 The rotorcraft tests to be carried out shall be agreed with the Project Director and can be expected to include representative examples of the following forms. In all cases the clarity of the cues to the pilot and the obviousness and ease of performing the actions he must take shall be assessed.

7.12.3 OSCILLATORY MALFUNCTIONS

(i) It is important to ensure that the system malfunctions tested can practically occur and will produce oscillatory system behaviour at significant frequencies.

7.12.4 STRAIGHT RUNAWAYS

(i) These shall normally be made from in trim conditions i.e., actuators at mid stroke and rotorcraft in trimmed flight, but on systems not fitted with auto-trim (i.e., automatic trim to maintain zero constant series actuator displacement) runaways shall also be made from initially out of trim conditions. The amount of the "out of trim" which should be considered is the maximum amount that it can be maintained taking into account the workload involved in the pilot retrimming the actuator position and the severity of the runaway from the adverse offset position. On those rotorcraft fitted with an auto-trim system the probabilities and consequences of malfunction of such a system shall also be evaluated.

(ii) It should be noted that there may be circumstances when the behaviour of a monitor or protective device will produce more hazardous effects than the malfunction being tested. Similarly slow runaways may lead to larger flight path excursions than fast runaways and be more difficult for the pilot to recognise, particularly in IMC. Recovery action should not be initiated until a readily recognisable cue is perceived by the pilot (see DEF STAN 00-970, Volume 2, Chapter 604, Section 2, para 13.

7.13 AUTOPILOT MONITORING

7.13.1 The ease with which the crew can monitor the correct functioning of the AFCS shall be assessed. For systems which automatically re-configure themselves, the ease with which the configuration at any time can be established shall be determined. Any pre-flight or in-flight testing necessary for flight safety purposes shall also be evaluated for accuracy and ease of operation.

8 FLIGHT DIRECTORS

8.1 INTRODUCTION

8.1.1 Flight Directors can be considered as operating in one or another of two broad conditions.

(i) In support of an automatic AFCS mode, i.e., coupled.

(ii) Independently of the AFCS, i.e., uncoupled.

8.2 GENERAL

8.2.1 All flight directors shall be assessed for the visibility and clarity of their displays under both day and night conditions.

8.3 UNCOUPLED FLIGHT DIRECTOR SYSTEMS

8.3.1 The extent by which the pilots work load is eased by obeying the commands of the flight director system in its various modes of operation shall be assessed. Accuracy of achievement with and without flight director shall also be measured.

8.3.2 Both assessments shall be made over the appropriate flight envelope in VMC and IMC.

8.3.3 The safety implications of failures within the flight director system shall be considered with particular attention being paid to unannounced failures that can direct slow changes of height, or speed.

8.4 COUPLED FLIGHT DIRECTOR SYSTEMS

8.4.1 When operating in conjunction with an AFCS the flight director system performance shall be assessed as a monitor of the performance of the AFCS and as an indicator of AFCS failures in all compatible modes of operation in both IMC and VMC over the appropriate flight envelope.

8.4.2 Failures of the flight director system shall be considered in addition to those of the AFCS when analysing the effects of the systems on the safety of the rotorcraft. When examining AFCS failures in flight, consideration shall be given to the suitability of the flight director in assisting the pilots intervention required as a result of the AFCS failure and in allowing the operation, interrupted by the AFCS failure, to be continued.

LEAFLET 909/1

AUTOMATIC FLIGHT CONTROL SYSTEMS

TEST EQUIPMENT

1 INSTRUMENTATION

1.1 Details of the parameter ranges, accuracies and resolutions are given in Leaflet 900/2, Table 1.

1.2 The following parameters should be recorded for the tests detailed in Chapter 909, or a lesser selection determined as appropriate and agreed by the Rotorcraft Project Director.

Item	Parameter
1	Time base
2	Manual event marker
4	Indicated airspeed
5	Altitude (pressure)
6	Altitude (radio altimeter)
9	Pitch attitude
10	Roll angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
25	Stick position (pitch)
26	Stick position (roll)
27	Yaw control pedal position
28	Stick force (pitch)
29	Stick force (roll)
30	Yaw control pedal force
31	Pitch trim position
32	Roll trim position
33	Yaw trim position
100	Collective pitch control position
101	Collective pitch trim position
102	Collective pitch stick force
103	Main rotor speed
104	Input drive torque
105	Main rotor collective pitch
106	Tail rotor pitch
119	Tail rotor blade pitch angle
125	Fore and aft servo position
126	Lateral servo position
127	Collective servo position
128	Yaw servo position
129	Pitch series actuator position
130	Roll series actuator position
131	Collective series actuator position
132	Yaw series actuator position

2 TELEMETRY

2.1 The use of telemetry for autopilot testing will generally, only be necessary where specific tests such as control runaways are likely to result in stress penalties.

2.2 The parameters to be measured will need to be considered for each case separately.

2.3 Tests on new technology control systems such as ACT and tests where there are flight safety implications may call for telemetry, in which case the parameters for such tests should be agreed with the Rotorcraft Project Director.

3 SPECIAL EQUIPMENT

3.1 Control inputs and simulated runaways with good repeatability are important features of autopilot test programmes.

3.2 The introduction of specialised control input units capable of inserting consistent stepped and pulse inputs into the control system should be considered.

3.3 A system for inserting simulated runaways should also be considered.

3.4 It must be possible to override such equipment at any time following initiation.

PART 9 APPENDIX No. 2
FLIGHT TESTS - HANDLING
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 900: GENERAL HANDLING FLIGHT TEST REQUIREMENTS

900	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-490	SPECIFICATION PRACTICES
	MIL-STD-961	MILITARY SPECIFICATIONS AND ASSOCIATED DOCUMENTS
	MIL-STD-962	MILITARY STANDARDS, HANDBOOKS, AND BULLETINS, PREPARATION OF
	MIL-STD-2124	FLIGHT DATA RECORDER, FUNCTIONAL STANDARDS FOR
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-W-25140	WEIGHT AND BALANCE CONTROL SYSTEM (FOR AIRCRAFT AND ROTORCRAFT)
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION
900 1.

2. RELEVANT DESIGN REQUIREMENTS
900 2.

3. APPLICABILITY
900 3.

4. EQUIPMENT
900 4.

5. LOADING
900 5.

6. GENERAL TEST CONDITIONS
900 6.

7. TESTS
900 7.

CHAPTER 901: GROUND HANDLING

901	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-F-9490	FLIGHT CONTROL SYSTEMS-DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. OBJECT
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2. RELEVANT DESIGN REQUIREMENTS
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3. APPLICABILITY
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4. EQUIPMENT
901 4.

5. LOADING
901 5.

6. GENERAL TEST CONDITIONS
901 6.

7. TESTS
901 7.

CHAPTER 902: TAKE-OFF, HOVER, LOW-SPEED MANOEUVRES AND LANDING

902	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. OBJECT

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2. RELEVANT DESIGN REQUIREMENTS

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3. APPLICABILITY

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4. EQUIPMENT

902 4.

5. LOADINGS

902 5.

6. GENERAL TEST CONDITIONS

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7. TESTS

902 7.

CHAPTER 903: LONGITUDINAL TRIM, STABILITY AND CONTROL

903	MIL-STD-250	STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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1. OBJECT
903 1.

2. RELEVANT DESIGN REQUIREMENTS
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5. LOADING
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6. GENERAL TEST CONDITIONS
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7. TESTS
903 7.

TABLE 1

TABLE 2

CHAPTER 904: LATERAL AND DIRECTIONAL TRIM, STABILITY AND CONTROL

904	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
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1. OBJECT
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2. RELEVANT DESIGN REQUIREMENTS
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904 4.

5. LOADING
904 5.

6. GENERAL TEST CONDITIONS
904 6.

7. TESTS
904 7.

TABLE 1

TABLE 2

TABLE 3

CHAPTER 905: DEMONSTRATION OF LIMITS OF FLIGHT AND MANOEUVRE ENVELOPES

905	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-1472	HUMAN ENGINEERING DESIGN CRITERIA FOR MILITARY SYSTEMS, EQUIPMENT AND FACILITIES
	MIL-H -8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
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1. OBJECT

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3. APPLICABILITY

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5. LOADING

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6. LIMITING FACTORS

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7. MANOEUVRE ENVELOPE PRESENTATION

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8. GENERAL TEST CONDITIONS

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9. TESTS

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10. DATA REQUIRED
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CHAPTER 906: ENGINE HANDLING AND ROTOR GOVERNING

906	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
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1. OBJECT

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2. RELEVANT DESIGN REQUIREMENTS

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3. APPLICABILITY

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4. EQUIPMENT

906 4.

5. LOADING

906 5.

6. GENERAL TEST CONDITIONS

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7. TESTS

906 7.

CHAPTER 907: AUTOROTATION, PARTIALLY POWERED FLIGHT AND ENGINE OFF LANDING

907	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
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3. APPLICABILITY

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4. EQUIPMENT

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5. LOADING

907 5.

6. TRANSITION TO AUTOROTATION WITH COLLECTIVE LEVER DELAYS

907 6.

7. AVOID AREA TESTS

907 7.

8. PARTIAL POWER FAILURE

907 8.

FIGURE 1: DEFINING PHRASES AND SYMBOLS FOR AVOID CURVE CONSTRUCTION

CHAPTER 908: DECK LANDING

908	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
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1. OBJECT
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5. LOADING
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6. GENERAL TEST CONDITIONS
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7. TESTS
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TABLE 1: A PILOT'S RATING SCALE FOR DECK OPERATIONS

8. THE PREPARATION AND PRESENTATION OF SHOL ENVELOPE
908 8.

FIGURE 1: CURRENT APPROACH AND LANDING TECHNIQUES

FIGURE 2: TYPICAL SHOL DIAGRAM FOR INTO WIND ENVELOPE

FIGURE 3: TYPICAL SHOL DIAGRAM FOR FORWARD FACING ENVELOPE

CHAPTER 909: AUTOMATIC FLIGHT CONTROL SYSTEMS

909	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TEST OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
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SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT
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2. RELEVANT DESIGN REQUIREMENTS
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6. GENERAL TEST CONDITIONS
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7. TESTS
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8. FLIGHT DIRECTORS
909 8.

PART 10

FLIGHT TESTS - INSTALLATIONS AND STRUCTURES

CONTENTS

* To be issued
**** In Preparation**

CHAPTER 1000

GENERAL FLIGHT TEST REQUIREMENTS SYSTEMS AND STRUCTURES

1 INTRODUCTION

1.1 This part states those tests which shall be made to demonstrate that the Rotorcraft engines, systems and structures function satisfactorily in accordance with the requirements of Part 7.

1.2 Each chapter of this Part contains tests of the Rotorcraft under specific flight conditions and is arranged in accordance with a standard layout, the paragraphs being grouped under headings in the following sequence:

(i) Object

(ii) Relevant design requirements

(iii) Applicability

(iv) Equipment

(v) Loading

(vi) General test conditions

(vii) Ground tests

(viii) Flight tests

2 RELEVANT DESIGN REQUIREMENTS

2.1 Each chapter of this Part lists for the convenience of flight test, instrumentation and data analysis personnel the design requirements associated with the flight tests of that chapter. Such requirements are listed for convenience only, for design purposes the full requirements must be consulted.

3 APPLICABILITY

3.1 The tests contained in this Part are to apply to all new types of Rotorcraft, and all Rotorcraft where modifications have been made which are likely to affect tests which ensure compliance with the Rotorcraft Specification. However, where a Rotorcraft is designed to foreign military or civil requirements the extent to which the Rotorcraft shall conform to this Part will be defined in the Rotorcraft Specification.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The total requirement for instrumentation shall always be the subject of advance discussion with the Rotorcraft Project Director in accordance with DEF STAN 05-123 Chapter 240. It should be noted that many of the parameters listed in this part are common with those listed for Part 9, consideration must be given to this when planning the total development trials programme for a Rotorcraft, both when the parameter lists are being drawn up and when the nature and form of the data acquisition system is being decided. Economies of effort and cost can be effected if proper attention is given to this phase of the trials planning.

4.1.2 The recommendations for instrumentation for the trials of Rotorcraft systems and structures are discussed in detail in Leaflet 1000/1, and the minimum parameters are contained in Tables 1 and 2.

4.1.3 All normal flying and engine instruments shall be fitted and serviceable for the tests of this Part. In addition, particular tests require special instruments, and these are detailed in the appropriate chapters. Obtaining, installing and calibrating all instruments shall be the responsibility of the contractor in accordance with the requirements of DEF STAN 05-123 Chapter 240.

4.1.4 Unless dispensation is obtained from the Rotorcraft Project Director, test instrumentation capable of continuous recording shall be fitted for the tests of Part 10.

4.1.5 Instrumentation recommended for recording data relevant to a particular test is given in Leaflet 10xx/1 of each subsequent chapter of this part. Paragraph 4 of each chapter specifies these parameters by reference to items listed in Tables 1 and 2 of Leaflet 1000/1 - “Recommended Test Instrumentation” and “Telemetry Parameters” respectively. There will be a need for additional parameters for many of the tests detailed in the individual Chapters. These will be listed in brief in the respective Leaflets. The ranges, accuracies and resolutions shall be determined by discussion with the Trials Establishments and equipment manufacturers. Telemetry parameters, as appropriate, are recommended primarily for flight safety reasons and do not necessarily improve trials efficiency.

4.1.6 For some trials, particularly those which constitute a significant flight safety hazard or those where real time analysis/monitoring would be likely to significantly reduce elapsed and/or flight times (e.g. structural loads and dynamic behaviour during manoeuvres at the fringe of the flight envelope) the use of telemetry shall be considered.

4.1.7 An Accident Data Recorder (A.D.R.) shall be fitted in accordance with Defence Standard 05-123 Chapter 240 and DEF STAN 00-970 Chapter 100.

4. 2 MISCELLANEOUS EQUIPMENT

4.2.1 Tests involving planned flight over water and in particular prolonged hover (e.g. AFCS/Sonar coupled modes) shall require:

(i) Operative flotation gear, if standard to the Rotorcraft. Such tests on Rotorcraft which do not have flotation gear as part of their standard build may only be made following consultation with the Rotorcraft Project Director.

(ii) A sonar locator beacon/pressure activated SARBE shall be installed or where such an installation is not part of the Rotorcraft standard build a portable unit shall be carried.

4.2.2 All external equipment likely to affect the dynamic characteristics of the Rotorcraft shall either be fitted or fully represented by a flightworthy mock-up of representative mass, moment of inertia and stiffness.

4.2.3 All equipment and systems associated with the satisfactory operation of the Rotorcraft in all the flight conditions anticipated for the trial shall be in place and fully operational before the trial commences.

4.2.4 In the interests of safety whenever flight tests of Rotorcraft fitted with dummy bombs or other stores which must not be released in flight, are required, all electrical wires leading to the carrier release slip, or similar device, shall be disconnected and any cables or rods for operating the mechanical release mechanism or mechanisms, if fitted, shall be disconnected. However, when carrying stores which may be released safely in flight a jettison system, should be fitted.

5 LOADING

5.1 The required loadings are stated in each chapter and are given either in full or as one of a combination of the following standard loadings.

(i)	W Max	Maximum permissible all up weight or mass at take-off, C of G to be specified.
	W max/aft	Maximum permissible all up weight or mass at take-off with maximum aft C of G.
	max/fwd	Ditto, maximum forward C of G
	max/right	Ditto, maximum right C of G
	max/left	Ditto, maximum left C of G

(ii) X The full service loading.

X1 The maximum mass with full operational equipment and the furthest forward C of G position at this mass.

X2 The maximum mass with full operational equipment and the furthest aft C of G position at this mass

X3 The maximum mass with full operational equipment and the furthest right C of G position at this mass.

X4 The maximum mass with full operational equipment and the furthest left C of G position at this mass.

(iii) Y The forward loading.

Y1 That practical loading giving the furthest forward C of G position and the greatest mass obtainable at this C of G.

Y2 That practical loading giving the furthest forward C of G position and the minimum mass obtainable at this C of G.

(iv) Z The aft loading.

Z1 That practical loading giving the furthest aft C of G position and the maximum mass obtainable at this C of G.

Z2 That practical loading giving the furthest aft C of G position and the minimum mass obtainable at this C of G.

When more than one operational role is specified then the most adverse combination of the above standard loadings shall be used.

5.2 Since the above loading conditions represent limiting cases, it is important that each test, and each part of each test, be made at the loading condition specified. Due allowance may be made for fuel consumed whilst the Rotorcraft is climbing to the test altitude but flight duration shall be limited to that which reduces all up mass by a maximum of 10%. If consumption of fuel causes a material change in C of G position, provision shall be made to enable the load carried or the fuel to be redistributed to compensate for this.

5.3 The mass of C of G position at which each test was made shall be reported. The centre of gravity being quoted in millimetres, in three dimensions, in Cartesian co-ordinates relative to any convenient set of mutually perpendicular axes intersecting at or near the centre of gravity. In each case the positive directions shall be aft of and upward from the origin.

6 GENERAL TEST CONDITIONS

6.1 The tests are to be made in full on a representative sample of the Rotorcraft. Tests must be conducted on systems and structures which are fully representative of the final Service standard. Where the system under test interfaces with other systems, they shall also be fully representative of the final service standard. When modifications liable to affect the aerodynamic or dynamic characteristics of the Rotorcraft are incorporated, the test shall be carried out at the most adverse conditions established from previous testing.

6.2 The conditions at which the tests shall be made are specified in each chapter. Reference is made to the appropriate para. of Part 9 if the tests require the same test condition. The test report shall contain a full report of the conditions of each test.

6.3 The terms, abbreviations and symbols used in this part are in accordance with Part 6. The additional abbreviations specified in Part 9 have also been used.

Note: the terms, abbreviations and symbols will be taken from Volume 1, Part 6 amended to cater for Rotorcraft.

7 TESTS

7.1 The Contractor is primarily responsible for flight tests which must be conducted to the satisfaction of the Rotorcraft Project Director. Where the tests are likely to make a contribution to Military Aircraft Release (MA Release) the flight test programme and instrumentation fit must be agreed by the Contractor and the Test Establishment in accordance with the procedures laid down in DEF STAN 05-123 Chapter 240.

7.2 Before the Rotorcraft is submitted for flight test at a Test Establishment the contractor shall certify that the Rotorcraft is safe to be flown by authorised Service pilots and specify the limitations to be observed. (see Leaflet 900/1).

7.3 In many cases it will be necessary in the interests of safety, to make preliminary tests at less severe conditions than those prescribed in the following chapters. This is a matter for the discretion of the contractor and consequently no reference to preliminary tests is made in this Part.

LEAFLET 1000/1

GENERAL FLIGHT TEST REQUIREMENTS - SYSTEMS AND STRUCTURES

TEST INSTRUMENTATION AND TELEMETRY

1 INTRODUCTION

1.1 The object of this leaflet is to provide guidance with respect to the instrumentation required or recommended for flight test trials concerned with the clearance of installations and structures of Rotorcraft. Because these trials involve a range of phenomena varying from quasi-static to very high frequency dynamic or very short duration transients, the data acquisition system has to be capable of handling very large volumes of data. At the same time there is a need to record substantially the same data as is required for handling and performance trials in order to determine which regime of flight, manoeuvre or even individual parameter has to be limited in order to avoid excessively damaging or dangerous situations (Ref. Part 9 Leaflet 900/1 for definition of limitations and Leaflet 900/2 for handling instrumentation parameters). There is also a very real need for good visual reference instruments for the pilot and flight test engineer(s) to ensure that test conditions and manoeuvres can be properly and consistently executed. It must be appreciated that the science of data gathering, retrieval and analysis is constantly advancing and contractors will be encouraged to take full advantage of the state of the art although not to the extent that the data acquisition system is in the throes of development during a major new Rotorcraft trials programme.

2 ON BOARD RECORDING

2.1 Accurate records of all the critical parameters are called for from all the tests specified in the various chapters of Part 10. These should sensibly utilise digital magnetic recording data acquisition techniques which lend themselves to automated forms of data retrieval, processing and analysis. The on-board recording system should normally be supplemented by an RF telemetry system when the “line of sight” requirement for UHF transmission can be met. The extent or complexity of the instrumentation fit will depend on the level or innovation within the Rotorcraft systems and structures, the knowledge of characteristics derive from simulation and on the state of development of the Rotorcraft.

3 TELEMETRY

3.1 The use of telemetry for any flight tests of an exploratory nature should be encouraged for the confidence it gives the crew during excursions into untested areas of the flight envelope and for the way in which the test programme may be advanced rapidly with the minimum number of sorties without compromising flight safety. When coupled with real time data analysis, with appropriate on line displays, telemetry offers the advantage of allowing thorough probing in problem areas and a unique involvement for specialists whilst tests are in progress. When used purely as a monitor system the overall accuracy of the telemetry should be better than ±5.0%, if the data so obtained is to be used as a prime source for fatigue substantiation, etc. then the resolution should be better than ±0.5% and the overall accuracy should be better than ±2.0%.

3.2 It should be noted that one of the difficulties encountered with telemetry monitored flight operations is that the primary radio communications link may be taken away from Air Traffic Control (ATC).

Telemetry facilities should contain provision for:

(i) ATC to monitor all radio traffic on the telemetry facility.

(ii) A reliable and instant alert link with ATC for emergency.

(iii) A predetermined procedure whereby ATC can perform their normal function in an emergency situation. This should include duplex radio facilities so that specialist advice from the telemetry personnel can continue to be available or on call.

4 VIDEO AND PHOTOGRAPHIC RECORDING

4.1 Video cameras and recorders (with or without visual monitors) can be employed in a number of cases, for example:

(i) Ice accretion on airframe, engine intakes, transmission cooling intakes and external stores carriers (Ref. Chapter 1006).

(ii) Behaviour of underslung loads, etc (Ref. Chapter 1017).

(iii) Detection of incipiently destructive shimmy of alighting gear (Ref. Chapter 1008).

4.2 Special purpose cameras can be employed as follows:

(i) High speed (frame rates up to 2400 pps) for weapons trials ie. Missile, Weapon launch and jettison, for which one or more cameras may be required to study motion in several degrees of freedom relative to the parent Rotorcraft.

(ii) Fast shutter speed, low frame rate large format cameras for study of ice accretion/shedding on main and tail rotor blades. This requirement may require natural light to be supplemented by high intensity synchronised electronic flash.

5 COCKPIT VISUAL DISPLAYS

5.1 Much of the information required by the pilot to set up the required test conditions is available from standard cockpit panel displays, however these should be calibrated to a standard appropriate to the requirements of each Chapter. Some standard displays may not be capable of presenting data to the required accuracy or resolution, in this event they will need to be supplemented with higher performance displays, calibrated appropriately. For specific trials it will be necessary to supplement the normal cockpit displays with specially installed equipment to display such parameters as control positions, blade pitch angles, hover indicators, etc.

6 AUTO OBSERVER PANELS

6.1 A.O.P's are not generally appropriate to major development trials on systems, due to the time consuming methods of data analysis and correction of data. The instruments therein are capable of distorting data unless response and damping characteristics are carefully chosen. They may however be appropriate for small scale work such as clearing a minor system cooling modification.

7 EQUIPMENT

7.1 Facilities should be provided for the application of calibration loads, angles, etc. to structures, components and systems so that total system calibrations and/or pre-and post-flight checks may be carried out. Wherever possible components or assemblies with load, stress or other sensors installed should be calibrated in laboratory test rigs prior to installation on trials Rotorcraft.

7.2 Provision should be made, where the instrumentation system is complex and the parameter requirements involve substantial numbers, for a ground check-out system based on a data logger/computer which can:

(i) Monitor input loads, etc., process data and display output in graphical and/or tabular form for checks on quality and validity of calibrations.

(ii) Once a calibration has been established as being to a satisfactory standard, output the data to magnetic disc or tape in a form suitable for direct input to the data base of the data analysis system in use.

7.3 General procedures for determining the instrumentation requirements for both contractors and official flight tests are laid down in DEF STAN 05-123 Chapter 240.

7.4 Many of the parameters required to monitor the conditions under which data from systems and structures will be obtained are common with the handling performance requirements and these parameters together with appropriate ranges, accuracies, are detailed in Leaflet 900/2. Where a chapter in Part 10 requires the recording of any of these parameters for specific tests each will be referred to only by its Item No. and parameter description. Only if the required resolution and accuracy are markedly different from those laid down in Leaflet 900/2 will any other recommendations be made.

TABLE 1

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
	GENERAL						
1	Time Base	Sec	Duration of flight + ½ hr	-	-		
2	Manual Event Marker	-	-	-	-	16	With external transmission facility
3	Crew Speech	-	Duration of flight + ½ hr				
	FLIGHT CONDITIONS						
4	Indicated Airspeed	kt	0 to Vne +20%	Greater of 1 kt or 0.5%	0.5 kt	8	See also Item 133 of this table
4A	Ground Speed	kt	0 to max required	Greater of 1 kt or 0.5%	0.5 kt	1	
5	Altitude (pressure)	ft	-1000 to ceiling + 3000ft	Greater of 20ft or 0.5 mb	0.1mb	8	Ceiling is highest altitude anticipated during test
6	Altitude (radio altimeter)	ft	0 to 5000 ft	1	0.5	8	
7	Total Temperature (or Ambient Temp/OAT)	°C	-60 to +60	1	0.5	8	
8	Angle of Attack	Deg	-30 to +30	0.25	0.1	16	Relative to fuselage datum
9	Pitch Attitude	Deg	±60/±10	1/0.1	0.5/0.05	16	
10	Roll Angle	Deg	±90/±10	1/0.1	0.5/0.05	16	
11	Sideslip Angle	Deg	±50	0.25	0.1	16	
12	Heading	Deg	0 to 360 & ±10	1/0.1	0.5/0.05	16	
13	Pitch Rate	Deg/ Sec	±100/±10	1/0.1	0.5/0.05	16	
14	Roll Rate	Deg/ Sec	±100/±10	1/0.1	1.5/0.05	16	
15	Yaw Rate	Deg/ Sec	±100/±10	1/0.1	0.5/0.05	16	
16	Longitudinal Acceleration	g	±1	0.01	0.001	16)	All accelerometers mounted as close as possible to the C of G (See Note 4)
17	Lateral Acceleration	g	±1	0.01	0.001	16)	
18	Normal Acceleration	g	-4 to +5	0.05	0.005	16)	

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
	AIRFRAME CONFIGURATION AND STATE						
19	Flap Setting*						
20	Landing Gear Position	Event	Up/Down	-	-	4	
21	Airbrake Position	Deg	Full	1	0.1	8	
22	Failure State	Event	-	-	-	-	See Note 1
23	Brake Parachute						
24	Fuel Contents	Kg/lb	Full	20/40	10/20	1	See Note 2
	CONTROL INPUTS						
25	Stick Position (Pitch)	Deg	Full	To be determined		16)))	Including Side Arm Controller where applicable
26	Stick Position (Roll)	Deg	Full	To be determined		16))	
27	Yaw Control Pedal Position	Deg	Full	To be determined		16	
28	Stick Force (Pitch)	lbf	±50	To be determined		16	
29	Stick Force (Roll)	lbf	±50	To be determined		16	
30	Yaw Control Pedal Force	lbf	±200	To be determined		16	
31	Pitch Trim Position	Deg	Full	To be determined		16	
32	Roll Trim Position	Deg	Full	To be determined		8	
33	Yaw Trim Position	Deg	Full	To be determined		8	
34	Elevator Position						
35	Aileron Positions						
36	Yaw Control Positions						
37	Port Brake Pressure	N/m^2	Full	To be determined	16		
38	Starboard Brake Pressure	N/m^2	Full	To be determined	16		

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
	SIGNAL SENSORS						
39	FCS State/Mode	Event					
40	ILS deviation (elevation and azimuth)						
41	Flight director demand (elevation and azimuth)						
42A	Stall/spin recovery device activation						
42B	Spin recovery device attachment load						
43A	Reserved						
43B	Stall Warning						
	ENGINE (EACH)						
44	Throttle Position(s)	Deg	Full	To be determined		16	As applicable to engine under consideration
45	Rotational Speed(s)	% or rpm	Full	To be determined		32	Power turbine and/or Gas Generator speed
46	JPT or TGT	°C	Full	To be determined		32	
47	Reserved						
48	Intake Position(s)	Deg	Full	To be determined		32	
49	Reserved						
50	Thrust Reverser Position						
51	Fuel Flow	Kg/hr	Full	To be determined		16	
52	Fuel Temp at Flowmeter	°C	Full	To be determined		1	If required for fuel mass flow
53 to 55	Reserved						
56	Inlet Guide Vane Position	Deg	Full	To be determined		32	
57	Variable Stator Vane Position	Deg	Full	To be determined		32	

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
58 to 60	Reserved						
61	Nozzle Angle						
62	Auxiliary Intake Door Position*						
63 to 67	Reserved						
68	Relight event	Event	-	-	-	16	
	MISCELLANEOUS SYSTEMS						
69	Brake Temperature (Port and Starboard)	°C	0 - 1000	10	1	1	
70	Tyre Temperature	°C	0 - 150	10	1	1	Suitable on-board instrumentation may be difficult to provide and it may be necessary to resort to measuring tyre temperature after the manoeuvre using a portable temperature probe inserted into pre-drilled tyres
71	Nosewheel Angle	Deg	Full	To be decided		16	

ADDITIONAL ROTORCRAFT RELATED INSTRUMENTATIONS

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
100	Collective Pitch Control Position	Deg	Full	15´arc	5´arc	16	Including Side Arm Controller where applicable
101	Collective Pitch Trim Position	Deg	Full	To be determined		8	
102	Collective Pitch Stick Force	lbf	±100	2	0.5	16	
103	Main Rotor Speed	% or rpm	0 -130%	0.3%	±0.1%	8	
104	Input Drive Torque	% or lbf ft	0 - max	1.0	±0.2%	16	Max = Full single engine emergency range +10%
105	Main Rotor Collective Pitch	Deg	Full	15´arc	5´arc	16	
106	Tail Rotor Pitch	Deg	Full	15´arc	5´arc	16	
107	Main Rotor Head Moment	Nm or lbf ft	Full ±20%	1.5%	0.5%	See Note 7	Two channels required (pitch and roll)
108	Tail Rotor Head Moment	Nm or lbf ft	Full ±20%	1.5%	0.5%	"	
109	Main Rotor Blade Flap Bending	Nm or lbf ft	Full ±20%	1.5%	0.5%	"	
110	Main Rotor Blade Lag Bending	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
111	Main Rotor Blade Torsion	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
112	Tail Rotor Blade Flap Bending	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
113	Tail Rotor Blade Lag Bending	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
114	Main Rotor Blade Pitching Moments/Rotating Control Load	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
115	Tail Rotor Blade Pitching Moments/Rotating Control Load	Nm or lbf in	Full ±20%	1.5%	0.5%	"	
116	Main Rotor Blade Pitch Angle	Deg	full	5´arc	5´arc	"	

ITEM No	PARAMETER	UNITS	RANGE	ACCURACY ± VALUES	RESOLUTION	SAMPLING RATE PER SEC	REMARKS
117	Main Rotor Blade Flap Angle	Deg	Full	5´arc	5´arc	See Note 7	
118	Main Rotor Blade Lag Angle	Deg	Full	5´arc	5´arc	"	
119	Tail Rotor Blade Pitch Angle	Deg	Full	5´arc	5´arc	"	
120	Tail Rotor Blade Flap Angle	Deg	Full	5´arc	5´arc	"	
121	Rotor Brake Temp	°C	0 - 6000	10	1fC	See Note 8	
122	Landing Gear Strut Extension/Compression	in/mm	Full	2%	½%	See Note 7	
123	Air Intake Temp	°C	-60/+60	1°C	0.5°C	See Note 8	
124	Tail Rotor Drive Shaft Torque	Nm/lbf ft	Full	1.5%	0.5%	See Note 7	
125	Fore and Aft Servo Position	mm	Full	1% of full stroke	0.2mm	50	
126	Lateral Servo Position	mm	Full	1% of full stroke	0.2mm	50	
127	Collective Servo Position	mm	Full	1% of full stroke	0.2mm	50	
128	Yaw Servo Position	mm	Full	1% of full stroke	0.2mm	50	
129	Pitch Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
130	Roll Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
131	Collective Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
132	Yaw Series Actuator Position	±mm zero at mid stroke	Full	1% of full stroke	0.2mm	50	
133	Vector Airspeed	Azimuth Degrees/Knots	Full	1% of range	0.5%	8	
134	Horizontal Stabiliser	± Degrees	Full	1°	0.5%	1	

TABLE 2

RECOMMENDED TELEMETRY PARAMETERS

The selection listed below is offered for consideration. The final selection should be determined by the requirement for the specific trials programme if capacity is limited.

ITEM	PARAMETER	UNITS	RANGE	REMARKS
	GENERAL			
1	Timebase	-	-	
2	Manual Event	-	-	Available to both Ground and Flight Observers
3	Crew Speech	-	-	“Duplex” R/T with ground observers
	FLIGHT CONDITIONS			
4	Indicated Airspeeds	kt	0 to Vne+ 20%	
5	Pressure Altitude	ft	-1000 to ceiling + 3000	
6	Incidence*			
7	Pitch Attitude	Deg	±60/±10°	
8	Roll Angle	Deg	±90°	
9	Sideslip Angle	Deg	±50°	
10	Heading	Deg	0 to 360°	
11	Pitch Rate	Deg/Sec	±100/±10	
12	Roll Rate	Deg/Sec	±100/±10	
13	Yaw Rate	Deg/Sec	±100/±10	
14	Longitudinal Acceleration	g	±1	) See note 4
15	Lateral Acceleration	g	±1	) See note 4
16	Normal Acceleration	g	-4 to +5	) See note 4

ITEM	PARAMETER	UNITS	RANGE	REMARKS
	CONTROL INPUTS			
17	Pitch Control Position	Deg	Full	)) Including Side Arm
18	Yaw Control Position	Deg	Full	) Controller where) applicable
19	Roll Control Position	Deg	Full	)
	ENGINE (EACH)			
20	Throttle Position	Deg	Full	
21	Power Turbine Rotational Speed	%	Full	
22	Turbine Entry Temp	°C	Full	Or equivalent parameter
23	Relight Event	-	-	

RECOMMENDED ADDITIONAL TELEMETRY PARAMETERS FOR ROTORCRAFT

ITEM	PARAMETER	UNITS	RANGE	REMARKS
100	Collective Pitch Control Position	Deg	Full	Including Side Arm Controller where applicable
101	Main Rotor Speed	%	0-130%	
102	Input Torque	% or lbf ft	0-Max	Max = Single Engine Emergency Torque +10%
	STRESS MONITOR			
103	Main Rotor Head Moment	Nm/lbf ft	Full ±20	Two channels required (pitch and roll)
104	Tail Rotor Head Moment	Nm/lbf ft	Full +20%	
105	Main Rotor Blade Torsion	Nm/lbf ft	Full +20%	)))
106	Tail Rotor Blade Torsion	Nm/lbf ft	Full +20%	))
107	Main Rotor Blade Flap Bending	Nm/lbf ft	Full +20%	) Should be at multiple stations) or at known critical stations if) capacity is limited
108	Main Rotor Blade Lag Bending	Nm/lbf ft	Full + 20%	))
109	Main Rotor Rotating Control Load	N/lbf	Full +20%	
110	Tail Rotor Rotating Control Load	N/lbf	Full +20%	Or equivalent
111	Tail Rotor Drive Shaft Torque	Nm/lbf ft	Full +20%	

NOTES

1 Indication of each discrete failure state of primary control and/or stability augmentation systems incorporating reversionary modes is desirable. For auto-pilot/auto-stabiliser runaway trials the magnitude of the signal from the runaway box should also be recorded to ensure that this is appropriate to the failure under consideration.

2 Data on the disposition of fuel contents is also required: it is acceptable for such data to be derived from fuel usage combined with measurements of total fuel contents.

3 Power input for rotorcraft may involve engine mounted or gearbox mounted torquemeters.

4 Accelerometers should be chosen such that they have an adequate low threshold and high enough sensitivity and yet are not over-ranged by the level of vibration at the mounting point. Low pass filters should be provided to remove frequencies higher than those of interest.

5 If there is a device which indicates to the pilot that control load or some other critical parameter is approaching or exceeding pre-determined limits and that he should withdraw from the situation, then that data should be recorded.

6 Sufficient data should be recorded to determine the operating loads/modes which occur in the dynamic components of the rotorcraft.

These should include the following components:

Blades - Main or tail
Hubs - Main or tail
Shafts - Main or tail
Transmission - Main or tail
Flying Controls - Main or tail - stationary and rotating
Angular freedom of motion of blades on main and tail rotors

7 To ensure fidelity of recorded data, the digital sampling rate must be at least twice that of the highest frequency component present in the original analogue signal. If, for any reason, this cannot be achieved, then low pass anti-aliaising filters will be required.

The characteristics of the anti-aliaising filters should be as follows:

Amplitude response:	Flat ±2% up to the maximum frequency of interest
Phase response:	Linear ±3% up to a maximum frequency of interest
Roll off:	At least 24db/octave (18 db/octave may be used but would result in a need for higher sampling rates)

Where anti-aliasing filters are used the minimum sampling rates should be either; seven times the maximum frequency of interest, or twice the value of the frequency at the -14db point of the filter, whichever is the greater.

8 The ranges specified for items 9,10,12,13,14 and 15 in this form xxx°/xx° indicate that a precision gyro is installed with either multi-range outputs provided directly or a single output processed by suitable signal conditioning to provide dual-ranges to the data acquisition system.

9 Some items in Table 1 are aeroplane related parameters which are listed in DEF STAN 00-970 Volume 1 Chapter 900 Table 1. They have been retained in these Tables by number and title, and marked thus *, since they may be relevant in some design of Rotorcraft, i.e. Tilt Rotor, Tilt Wing or compound helicopter.

CHAPTER 1001

ENGINES
GENERAL FLIGHT TEST REQUIREMENTS

1 GENERAL

1.1 The object of the testing set out in this chapter is to demonstrate that the engine operation is satisfactory under the ground and flight conditions likely to be encountered in service, including engine failure.
This includes the specific issues pertaining to multi-engined rotorcraft where testing is required on the effectiveness of control and the ease of restarting shutdown engines.
This chapter also contains tests appropriate to the establishment of acceptable operating procedures for release to Service.

1.2 The tests of this chapter are designed to cover all factors that can affect the operating characteristics, functioning and handling of the engine during ground operation and in flight. Many are common with those of Chapter 906.

1.3 The flight testing of engine installations in rotorcraft is to validate any analysis and bench testing carried out by the engine and aircraft manufacturers which may have produced an understanding of the engine controls, operational margins and engine/aircraft operating procedures. Flight testing will ensure that validated mathematical models are used in the prediction of engine behaviour particularly around the critical flight regimes to complement the analytical basis for certification. Flight testing should also ensure satisfactory integration of the engine with the airframe and other systems (eg. fuel system, pneumatics, transmission etc.).

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2. Chapters 100, 101, 107, 700, 702, 706, 708, 710, 711, 712 and 801.

3 APPLICABILITY

3.1 The requirements of this chapter are applicable to new rotorcraft, or existing rotorcraft that have been subject to such modifications that may affect engine performance, engine controls, or engine/airframe interactions.

3.2 The need to flight test modifications has to be considered on the merits of each case. However, changes to or degradation of engine handling characteristics requiring revision of Service operating procedures may result from the cumulative effects of modifications which individually would not justify retesting. This shall be borne in mind when considering the possible effects of any modification and, where any doubt exists, an abbreviated programme of tests exploring the areas of concern shall be undertaken.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 It shall be noted that many of the parameters required for the tests of this chapter are common both with those of Part 9 and the remaining chapters of Part 10. (Refer to Chapters 900 and 1000, also to Leaflets 900/2 and 1000/1).

4.1.2 The recommendations for instrumentation for the trials of rotorcraft engines and engine control systems are contained in Leaflet 1000/1. (Refer also Chapter 900 and 1000 together with appropriate leaflets).

4.1.3 The instruments and their standard of calibration shall be such that the crew can monitor accurately IAS, altitudes, side slip angle, engine RPM(s), PTIT(s), AIT(s), OAT, engine torque(s), main rotor RPM, fuel flow(s), etc. Special test instrumentation shall be provided where the standard cockpit instruments are considered inadequate for this purpose, e.g. digital RPM indication or OAT or where the interfaces between the engine and rotorcraft are crucial.

4.1.4 Telemetry is not normally a requirement for engine tests but for some trials, particularly those which constitute a significant flight safety hazard or those where real time monitoring/analysis would be likely to materially reduce elapsed and/or flight times (e.g. Weapon firing where ingestion of rocket exhaust gases might cause surge or other engine malfunction) then telemetry provision shall be considered.

4.2 TEST EQUIPMENT

4.2.1 Standard of airframe and engines:- The aerodynamic standard of the airframe in so much as it affects engine behaviour, i.e. Intake, FOD/Ice/Snow guards, Auxiliary intake doors, etc., the standard of power off-takes and airbleeds used for rotorcraft services, the standard of the engine(s) and the associated engine control equipment, shall be fully representative of the In-Service standard. Testing shall include such combinations of role equipment as required by the Rotorcraft Specification.

4.2.2 Engine setting and adjustments:- Prior to commencement of the tests the engine(s) and engine control system(s) shall be set to within the current limits applicable to the engine(s), etc. under test. Where tolerances are permitted, (e.g. on bleed valves, compressor blow off valves, variable inlet guide vanes, stall margin), which may affect engine handling, tests are to be made at the upper and lower limits of those tolerances.

4.2.3 Specialised Test Equipment. If an engine is fitted with automatically controlled bleed from the compressor, automatically controlled inlet guides valves, or like equipment, which may affect engine handling, then consideration shall be given to substituting a manual control for specific tests. If there is doubt to the magnitude of the engine(s) compressor stall margins then consideration shall be given to the provision of the necessary test equipment and instrumentation to establish the adequacy of the margins and the nature of any remedial actions found necessary. Manual selection of reversionary control modes shall be considered if the engine(s) is (are) controlled by an automatic system and tests are required whilst the engine(s) is (are) under reversionary control.

4.2.4 Emergency Power Supplies. The Rotorcraft shall have adequate alternative power supplies in the event of engine flame-out or shut down to permit controlled flight whilst relight attempts are made, or an engine out landing is carried out. In particular there shall be adequate electrical supplies to operate the test instrumentation once an engine has been shut down.

5 LOADING

5.1 The tests may generally be made at any convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification or, taking into account limitations imposed by the carriage of external stores or other loads (i.e. Maximum normal acceleration, attitude, speed etc.). For testing to explore the full collective pitch map (collective reset) characteristic throughout the flight envelope, both maximum and minimum all up masses will be required to be flown.

5.2 On rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial e.g., the effect of rocket/missile launch or gun firing on engine behaviour.

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions are to be representative of the Service role of the rotorcraft in the range of climatic conditions defined in the Rotorcraft Specification. However, for any test where loss of power could lead to a forced landing, the tests shall be done in conditions such that an engine out landing may be safely made. Analysis may be used to reduce and rationalise the flight and ground test programme particularly to extrapolate to extremes of climate and manoeuvre, provided that predictive methods are appropriately validated.

6.2 When a rotorcraft is powered by a number of engines of the same type with similar features, the tests detailed in this chapter shall be made on one typical engine, with supplementary tests as considered necessary on the remainder. Where intake positions and configurations are such as to be subjected to dissimilar flow conditions or recirculation of exhaust gasses then tests shall be made to determine the effects on engine performance and handling, including any tendency to surge. With rotorcraft, engines function as an input to a transmission system which subsequently leads to main rotor(s), tail rotor (where applicable) and numerous accessory drives. As a consequence multiple engines in rotorcraft do not necessarily have the same independence of operation which can be assumed in aeroplane installations and any testing on individual engines has to take into account the operation of the other engine or engines. It is particularly important to establish that operating characteristics are satisfactory for all configurations of engine operation; i.e. Single, twin, triple, etc. in any practical combination (including engines that have two (or more) different control systems e.g. primary and revisionary), since engine control system characteristics may result in adverse effects on torque sharing or transient response. With common or different intakes, shared power off-take, or any other installation feature such that the operation of any one engine affects operation of the other(s), it shall be established that the operation of any engine does not produce undesirable effects on the operation of other(s). (Change in response, surge, etc.).

6.3 When tests to this chapter have been made on prototype or development engines and engine control systems, it will usually be necessary to repeat all or part of the programme on production engines, to ensure that production methods and modification standards have not caused degradation of handling characteristics to an unacceptable degree. The amount of testing required will depend on the nature of any modifications incorporated and the degree of variation in handling and control characteristics which is predicted or encountered.

6.4 The tests shall be made within the operational limits for the engine stated in the engine Technical Certificate (production engines).

6.5 Appropriate tests shall be made to ensure the rotorcraft can operate within the specification requirements for all fuels shown in the Rotorcraft Specification. Tests on the extremes of specific gravity and/or temperatures of the primary fuel shall be required and be extended to include secondary fuels, where these fuels have characteristics which differ markedly from the primary fuel (Specific Gravity, viscosity, boiling temperature, etc.).

6.6 The aerodynamic standard of the airframe in so far as it affects engine behaviour, i.e. Intake, FOD/Ice/Snow guards, engine intake and exhausts shall be as defined in the Rotorcraft Specification. Testing shall include such role equipment as may be applicable: e.g. Sand Filters, OD guards, anti-icing intakes and IR suppression, etc.

6.7 Engine settings and adjustments. Prior to commencement of tests the Engine and its control system(s) shall be set to within the current limits applicable to the unit(s) under test. Where tolerances are permitted, (e.g. on bleed valves, compressor blow-off valves, speed controls, etc.) which may affect engine functioning, tests are to be made at the upper and lower limits of these tolerances.

6.8 Tests to establish the effects on power plant operation on icing conditions are covered by the requirements of chapter 1006.

7 GROUND TESTS

7.1 The tests to be carried out shall take into account the following in respect of engine starting and control up to the minimum self sustaining speed and shall include an assessment of the effects of wind speed and direction.

a) Smooth functioning, sensitivity and suitability of controls.

b) Functioning, sensitivity and suitability of the provisions made for monitoring engine functions during start up cycle.

c) Engine starting using both onboard and external power sources. This shall include the investigation of failed starts and the establishment of procedures for dealing with them.

d) Hot restarting shall be investigated and suitable procedures established.

e) Cold starting, including the effects of any cold start fuel enrichment devices.

7.2 Test to establish suitable procedure for:

a) Engagement of accessory drive (where applicable) including an assessment of running and control stability at all selectable operating conditions (e.g. N range, trim and engine matching) up to the maximum permitted engine power/RPM.

b) Rotor engagement and acceleration up to normal operating RPM under both manual and automatic control with single or multi-engine(s) as appropriate. Running and control stability shall be assessed.

7.3 Tests shall be carried out to demonstrate the ability to comply with any operational rapid reaction requirements laid down in the Rotorcraft Specification, particularly at the low temperature end of the climatic range.

7.4 Tests shall be undertaken to establish the following during prolonged ground running, including in accessory drive:-

a) The effect of exhaust gas efflux(es) on local structures including rotor blades.

b) The effects of recirculation and induction of exhaust gas efflux(es) into engine, APU, transmission, and equipment bay cooling air flows.

c) That all engine bay, component and accessory temperatures remain within specification limits, especially during post-shut-down soak following stabilisation of temperatures at the maximum-on-ground power condition. The soak shall, where applicable, take place in still air conditions (less than 5 knots).

d) The effects of a variety of wind conditions.

7.5 Tests shall be made to access the efficacy of fire suppression system(s) in the engine bay(s) including suppressant/extinguishant distribution and concentration and the dilution effects of the flow of bay cooling air.

7.6 Tests shall be made to establish the ability of the engine bay drains to cope with the ingress of fluids (water, fuel, hydraulic fluid, etc.) and that the drains dispose of such fluids safely without undue contamination of airframe, rotor, stores, etc.

7.7 Tests to establish Electro-Magnetic Compatibility shall be made (refer to Chapter 1011).

7.8 Tests shall be made to check the safety and integrity of air bleed systems for anti-icing and cabin/cockpit heating at all levels of bleed throughout the achievable engine power range on the ground.

8 FLIGHT TESTS

8.1 Tests are required to investigate the following in order to determine operating procedures/limitations:

a) Engine behaviour during take-off, hover, sideways and rearwards flight and low speed manoeuvres in and out of ground effect near the ground. These tests shall also include an investigation of the effects of engine and APU exhaust gas recirculation, together with the effects of strength and direction of surface wind.

b) Engine/rotor speed governing throughout the flight envelope to determine control and stability margins, together with the behaviour of the engine(s)/control system(s) in response to rapid power demands, e.g. collective, cyclic and yaw control inputs and to simulated engine failure in multi-engined Rotorcraft.

c) The effect on engine handling upon operation of any auto surge protection system(s) (where applicable).

d) The behaviour of engine(s)/control system(s) during autorotation out of governed speed range.

e) Multi-engine matching throughout the flight regime taking into account any requirement to operate with one engine shut-down.

f) Operation on any of the specific fuels and lubricants including hot fuel (refer to para 6.3).

g) Engine lubrication over a range of rotorcraft attitudes and accelerations (g) taking into account the possible effect on engine bearings, accessories etc. of long periods of engine in-flight shut-down operations, where the Rotorcraft Specification requires this for operational purposes.

h) The effects of compressor air bleed, from zero to maximum, on steady-state installed engine performance throughout the flight envelope.

j) Flight conditions and operational envelope where not specifically defined in the Rotorcraft Specification.

k) Engine restarting in flight, throughout the Rotorcraft Specification flight envelope, with particular attention being paid to the following:-

i) Restart after prolonged period of engine shut-down where this condition is required by the Rotorcraft Specification (i.e. for most economical cruise).

ii) Restart after an all engine flame out. This latter may be simulated by restarting an engine in auto-rotative flight with the other engine(s) held at flight idle. The restart should be accomplished utilising emergency power supplies/auxiliary power unit as appropriate.

m) Measurements shall be made of engine bay accessory component temperatures throughout the flight envelope, including shut-down soak following prolonged still air hover at maximum all up mass.

8.2 Tests shall be carried out to confirm analytical/bench test activities ensuring freedom from any tendency to stall or surge or exhibit undesirable control characteristics during weapon firing.

8.3 Tests shall be carried out to establish the effects on intake performance and engine bay cooling of any baffling system(s) which may be added to reduce or minimise radar reflections from engine compressor front ends.

8.4 Combat or tactical support rotorcraft will in all probability be required to be capable of carrying some form of infra-red suppression which will generally involve a form of efflux cooling/dispersion. Such systems are likely to have a major influence on an engine installation and as such have been included in this chapter. Testing shall cover the following aspects:-

a) Assessment of the effect on installation losses.

b) Engine/engine bay temperatures.

c) IR signature measurements.

d) Vibration/loads transmitted to engine structure.

8.5 Ground and flight vibration measurements shall be made throughout the governed range of engine/rotor RPM, flight and ground idle and shut-down conditions as applicable in the following areas:-

a) At engine/airframe interfaces.

b) On critical engine components and engine module interfaces (to be done in association with the engine manufacturer).

c) On engine accessories, whether engine or airframe mounted.

8.6 There is a need to demonstrate that any rotor or transmission generated torsional oscillation fed back to the engine output shaft either:-

a) Does not occur at any engine power turbine disc/blade/shaft resonant frequency in any mode of vibration

or

b) That the resultant effects are non-damaging due either to the low level of forcing or adequate levels of damping in the engine components.

NOTE: These tests will require supporting engine/transmission test rig measurements to be made as well as mathematical modelling studies. It may under some circumstances be possible to clear a particular combination by analogy with similar installation in another rotorcraft or with a variant of the other under consideration. However, clearance by actual measurement is to be preferred and due consideration to this requirement shall be given when selecting any system(s) which is to be used to monitor engine output/transmission input torque(s) (a system frequency response from DC to a minimum of 1kHz would be essential).

The alternative might be to provision special torque sensors for testing with an attendant design and installation complexity, together with some risk of altering the resonance characteristics of the rotorcraft transmission.

8.7 Where Full Authority Digital Engine Control Systems are installed tests shall be carried out to ensure that matching of engine/engine controller characteristics over the full range of operating conditions in the Rotorcraft Specification and within the range of variations of engine performance and dynamic characteristics achievable in production, shall always provide for consistent engine handling characteristics and adequate safety margins in the event of either hardware malfunction or software deficiency. In addition, tests shall be conducted to assess the effect of system failure modes on Rotorcraft handling and mission capabilities as appropriate throughout the flight envelope.

8.8 Test shall be made to assess the efficiency of fire suppression system(s) in the engine bay(s), including suppressant/extinguishant distribution and concentration and the dilution effects of the flow of bay cooling air.

NOTE: Because of the distinctive way in which the engine installation is integrated with the rotor system in rotorcraft a large part of the testing called for in this chapter will be associated with handling and will as a consequence be covered by tests required to satisfy Part 9.

LEAFLET 1001/1

ENGINES
GENERAL FLIGHT TEST REQUIREMENTS
TEST INSTRUMENTATION PARAMETERS

1. INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc, are given in Leaflet 1000/1 Table 1.

1.2 The following parameters are proposed for the tests detailed in Chapter 1001. Other parameters may be necessary to establish the ability of the engines to function and perform in the Rotorcraft environment. Note that "Item No." below relates to Leaflet 1000/1.

ITEM NO.	PARAMETER
1.	Time Base
2.	Manual Event Marker
3.	Crew Speech
4.	Indicated Airspeed
5.	Altitude (pressure)
7.	Total Temperature (or ambient Temp/OAT)
8.	Angle of Attack
9.	Pitch Angle
10.	Roll Angle
11.	Sideslip Angle
12.	Heading
13.	Pitch Rate
14.	Roll Rate
15.	Yaw Rate
16	Longitudinal Acceleration
17	Lateral Acceleration
18.	Normal Acceleration
24.	Fuel Contents
27	Yaw Control Pedal Position
44.	Throttle Position(s) on Engine
45.	Rotational Speed(s)
51.	Engine Fuel Flow(s)
56.	Inlet Guide Vanes Position(s)
57.	Variable Stator Vane Position(s)
	Main Rotor Shaft Torque(s)
	Fuel Pressure at Engine Inlet(s)
	Engine Bay Air Pressures
	Engine Component, Bay and Accessories (including EECU)
	Engine Accessory 3-axis Vibration (including EECU)
	Rotorcraft AC and DC Bus Bar Voltages
	Engine FADEC Control Data-stream (where applicable)

ITEM NO.	PARAMETER
100.	Collective Pitch Control Position
103.	Main Rotor Speed
104.	Input Drive Torque
105.	Main Rotor Collective Pitch
106.	Tail Rotor Pitch
123.	Air Intake Temperature
124.	Tail Rotor Drive Shaft Torque
	Compressor Delivery Air Temperature
	Power Turbine Inlet Temperature
	Turbine Exhaust Temperature
	HP Fuel Pump Fuel Inlet Temperature
	Air Intake Air Temperature Distribution
	Fuel Tank Temperature
	Compressor Delivery Air Pressure
	Air Intake Air Pressure Distribution Dynamic and Static Head
	Air Starter(s) Air Inlet Pressure
	Compressor Bleed Valve Position(s)
	Compressor Pressure Relief Valve Position(s)
	Air Start Control Valve Position(s)
	Engine(s) Mounts, 3-axis Vibration
	IR Suppressor Mounting(s), 3-axis Vibration
	MGB Input Housing(s), 3-axis Vibration
	Store Release/Jettison Even(s)
	De-icing Cycle Initiation Event(s)

2 TELEMETRY

2.1 Some of the tests of Chapter 1001 could result in hazard to the rotorcraft so that telemetry would be considered appropriate and recommendations are made accordingly. Where telemetry is not, for any reason, feasible, then real time monitoring of the critical parameters should be provisioned for the flight test engineer. The following parameters should be telemetered. Details of parameter ranges, etc, are given in Leaflet 1000/1, Table 2. Note that "Item No." below relates to Leaflet 1000/1.

ITEM NO.	PARAMETER
1.	Time Base
2.	Manual Event Marker
3.	Crew Speech
	<u>FLIGHT CONDITIONS</u>
4.	Indicated Airspeed
5.	Pressure Altitude
6.	Incidence
7.	Pitch Attitude
8.	Roll Angle
9.	Sideslip Angle

ITEM NO.	PARAMETER
10.	Heading
11.	Pitch Rate
12.	Roll Rate
13.	Yaw Rate
14.	Longitudinal Acceleration
15.	Lateral Acceleration
16.	Normal Acceleration
	CONTROL INPUTS
17.	Pitch Control Position
18.	Yaw Control Position
19.	Roll Control Position
	ENGINE (EACH)
20.	Throttle Position on Engine
21.	Power Turbine Rotational Speed
22.	Turbine Entry Temperature
23.	Relight Event
100.	Collective Pitch Control Position
101.	Main Rotor Speed
102.	Input Torque
111.	Tail Rotor Drive Shaft Torque

CHAPTER 1002

AUXILIARY POWER SYSTEMS

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that Auxiliary Power System operation is satisfactory under all ground and flight conditions likely to be encountered and flight tests of this chapter are designed to cover all factors which can affect the functioning and operating characteristics of the Auxiliary Power System(s) during ground operation and flight. These include the influence of gearbox and accessory drive(s), engine starting loads and rotor acceleration loads on the operation of the Auxiliary Power System(s).

1.2 The tests shall include an assessment of the ease with which the Auxiliary Power System can be started/restarted and controlled both on the ground and in flight.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 100, 101, 107, 702, 703, 704, 705, 706 and 712.

3 APPLICABILITY

3.1 When a rotorcraft has one or more Auxiliary Power System(s) with similar design characteristics, then the tests of this chapter shall be carried out on one typical system with consideration being given to supplementary tests on the other(s). Where intake configurations and positions are such as to be subject to dissimilar flow conditions, exhaust recirculation, etc., then tests shall be made to determine the effects on auxiliary power system performance and controllability, including, with gas turbine powered units, any tendency to surge under transient conditions.

3.2 When tests to this chapter have been done on prototype or development installations, it will usually be necessary to repeat the tests in whole or in part on production installations, to ensure that production methods and modification standards have not degraded operating characteristics of the installation to an unacceptable degree. The amount of retesting required will depend on the nature of any changes incorporated and the degree of variation encountered in control and response characteristics.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 It shall be noted that the basic parameters required for the tests of this Chapter are common with those of Part 9 and the other chapters of Part 10 (refer to Chapter 1000 and Leaflets 900/2 and 1000/1).

4.1.2 The recommendations for instrumentation for the trials of rotorcraft Auxiliary Power Systems are contained in Leaflet 1002/1.

4.1.3 The instruments and their standard of calibration shall be such that the crew can monitor accurately IAS, Altitude, Attitude, side-slip angle, air intake temperature, OAT, main rotor RPM, APU RPM, turbine entry temp, etc. Special test instrumentation shall be provided where the standard cockpit indications are considered inadequate for this purpose, e.g., digital RPM indication or OAT.

4.1.4 In addition the standard of the instrumentation shall be such that it is possible to obtain a continuous record of those parameters which are necessary to monitor the behaviour of the auxiliary power system(s) and directly affected rotorcraft systems e.g., APU, engine(s), fuel system, pneumatic, hydraulic, electrical and transmission systems.

4.2 TEST EQUIPMENT

4.2.1 If the Auxiliary Power Systems and/or any of it's functions are controlled by an automatic system, then consideration shall be given to the substitution of a manual control for specific tests.

5 LOADING

5.1 The tests may be made at any convenient loading and centre of gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification or, if not so defined, as envisaged by the Service user, taking into account limitations imposed by the carriage of external stores or other loads (i.e. maximum nominal acceleration, attitude, speed etc.).

5.2 On rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial (e.g., the effect of rocket/missile launch or gun firing on engine behaviour).

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions are to be representative of the Service role of the rotorcraft, in the range of climatic conditions defined in the Rotorcraft Specification. However, for any test where loss of power could lead to a forced landing, the tests shall be done in weather conditions such that an engine out landing may be safely made.

6.2 Tests shall be made within the operational limits for the APU stated in the Technical Certificate. The authority of the Rotorcraft Project Director shall be obtained before doing any test that could involve exceeding those limits. In the event that these limits are inadvertently exceeded to any degree then the Rotorcraft Project Director shall be consulted before tests are continued.

6.3 When a rotorcraft is to be cleared for the use of more than one type of fuel, tests shall include the use of each type of fuel for which clearance is required unless otherwise agreed by the Rotorcraft Project Director. Tests on the extremes of specific gravity and/or temperatures of the primary fuel shall be required and shall be extended to include secondary fuels, where these fuels have characteristics which differ markedly from the primary fuel (Specific Gravity, viscosity, boiling temperature, etc.).

6.4 The aerodynamic standard of the airframe in so far as it affects APU behaviour, i.e. Intake, FOD/Ice/Snow guards, engine intake and exhausts shall be fully representative of the in-service standard. Should this not be so the Rotorcraft Project Director must be informed of any deficiencies and the probable effect these may have on any CA Release. Testing shall include such role equipment as may be applicable: e.g., Sand filters, FOD guards, anti-icing intakes and IR suppression, etc.

6.5 Auxiliary Power Unit settings and adjustments. Prior to commencement of tests the unit and its control system(s) shall be set to within the current limits applicable to the unit(s) under test. Where tolerances are permitted, e.g., on bleed valves, compressor blow-off valves, speed controls, etc., which may affect APU functioning, tests are to be made at the upper and lower limits of these tolerances.

6.6 When an Auxiliary Power System is mechanically connected to a rotorcraft accessory gearbox, then tests shall be made to ensure that there is no adverse interaction between units, particularly where such an accessory gearbox can also be driven by a main engine and/or main transmission unit.

7 GROUND TESTS

7.1 The tests to be carried out shall take into account the following in respect of starting and control up to the minimum self sustaining speed.

(i) Smooth functioning, sensitivity and suitability of controls.

(ii) Functioning, sensitivity and suitability of the provisions made for monitoring APU functions during the start up cycle.

(iii) APU starting using both onboard and external power sources. This shall include the investigation of failed starts and the establishment of procedures for dealing with them.

(iv) Hot restarting shall be investigated and suitable procedures established.

7.2 Tests to establish suitable procedures for:

(i) Engagement of accessory drive (where applicable) including an assessment of running and control stability under both manual and governed conditions up to the maximum permitted power/RPM and with various accessory loads as appropriate.

(ii) Acceleration up to and operation at normal (100%) RPM under both manual and automatic control as appropriate. Running and control stability shall be assessed.

7.3 Tests shall be carried out to establish the operational effectiveness of the APU as a ground power source and engine start facility under all the climatic conditions required by the Rotorcraft Specification.

7.4 Tests shall be carried out to demonstrate the ability to comply with any operational rapid reaction requirements laid down in the Rotorcraft Specification, particularly at the low temperature end of the climatic range.

7.5 Tests shall be undertaken to establish the following during prolonged ground running, including in accessory drive:-

(i) The effect of exhaust gas efflux on local structures including rotor blades.

(ii) The effects of recirculation of exhaust gas effluxes on APU performance.

(iii) The effects of indication of exhaust gas effluxes into engine, APU, transmission, and equipment bay cooling air flows.

(iv) That all APU bay, component and accessory temperatures remain within specification limits, especially during post shutdown soak. The relative direction and strength of surface wind shall be taken into account.

7.6 Tests shall be made to assess the efficacy of fire suppression system(s) in the APU bay(s) including suppressant/extinguishant distribution and concentration and the dilution effects of the flow of bay cooling air.

7.7 Tests shall be made to establish the ability of APU bay drains to cope with the ingress of fluids (water, fuel, hydraulic fluid, etc.,) and that the drains dispose of such fluids safely without undue contamination of airframe, rotors, stores etc.

7.8 Tests to establish Electro-Magnetic Compatibility shall be made (refer to Chapter 1011).

8 FLIGHT TESTS

8.1 Tests are required to investigate the following:-

(i) APU behaviour during ground running, take-off, hover, sideways and rearwards flight and low speed manoeuvres in and out of ground effect, near the ground. These tests shall also include an investigation of the effects of engine and APU, exhaust gas recirculation, together with the effects of strength and direction of surface wind.

(ii) Effects of autorotation with engine(s) out of governed range.

(iii) Operation on any of the specific fuels and lubricants including hot fuel (refer to para 6.4).

(iv) APU lubrication over a range of rotorcraft attitudes and accelerations (g) taking into account the possible effect on APU bearings, accessories, etc., long periods of APU in-flight shut-down operations, where the Rotorcraft Specification requires this for operational purposes.

(v) The effects of using compressor bleed air for cabin heating, intake de-icing, etc., on APU performance and control.

(vi) APU starting in flight, with particular attention being paid to restart after prolonged period of APU shutdown where this condition is required by the Rotorcraft Specification.

(vii) Flight conditions and envelope where not specifically defined in the Rotorcraft Specification.

8.2 Tests shall be carried out to ensure freedom from any tendency to stall or surge or exhibit undesirable control characteristics during weapon firing.

8.3 Tests shall be carried out to establish:-

(i) The effects on intake performance and bay cooling of any baffling system(s) which may be added to reduce or minimise radar reflection from engine compressor front ends.

8.4 Combat or tactical support rotorcraft will in all probability be required to be capable of carrying some form of infra-red suppression which will generally involve a form of efflux cooling/dispersion. Such systems are likely to have an influence on an installation and as such have been included in this chapter. Testing shall cover the following aspects:-

(i) Assessment of the effect on installation losses.

(ii) Bay temperatures.

(iii) IR Signature measurements.

(iv) Vibration/loads transmitted to structure.

8.5 Ground and flight vibration measurements shall be made throughout the governed range of engine/rotor RPM, flight and ground idle and shut-down conditions as applicable in the following areas:-

(i) At airframe interfaces.

(ii) On critical components and module interfaces (to be done in association with the manufacturer).

(iii) On accessories, whether airframe mounted.

8.6 Where continuous operation of the APU is required by the Rotorcraft Specification then consideration shall be given to the need to demonstrate that any torsional oscillation feed back to the APU through a mechanical power take-off to transmission accessory drive(s) either:-

(i) Does not occur at an APU power turbine disc/blade/shaft resonant frequency in any mode of vibration.

or

(ii) That the resultant effects are non-damaging due either to the low level of forcing or to adequate damping in the APU components.

These tests will require supporting APU/transmission/accessory drive unit test rig measurements to be made as well as mathematical modelling studies. It may under some circumstances be possible to clear a particular combination by analogy with a similar installation in another rotorcraft or with a variant of the one under consideration, however measurement is to be preferred.

LEAFLET 1002/1

AUXILIARY POWER SYSTEMS

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are considered to be the minimum for the tests detailed in Chapter 1002. Such other parameters as are necessary to establish the ability of the Auxiliary Power System(s) to function and perform to specification in the rotorcraft environment should be determined by consultation with the equipment manufacturer(s)/supplier(s) and the Official Trials Establishments and agreed with the Rotorcraft Project Director.

ITEM No	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
5	Altitude (pressure)
7	Total Temperature (or Ambient Temp/OAT)
8	Angle of Attack
9	Pitch Attitude
10	Roll Angle
11	Sideslip Angle
12	Heading
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
24	Fuel Contents
27	Yaw Control Pedal Position
100	Collective Pitch Control Position
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
	Air Intake Air Temperature
	Turbine Entry Gas Temperature
	Compressor Air Bleed Temperature
	Air Starter Air Inlet Temperature
	Fuel Pump Fuel Inlet Temperature
	Fuel Tank Temperature
	APU Air Intake Surface Temperature
	Fuel Pressure at APU/Airframe i/v
	Air Bleed Pressure
	Air Starter Air Inlet Pressure
	Booster Pump Outlet Pressure
	Air Start Control Valve Position
	Accessory Drive Select Lever Position
	APU mounts, 3-axis Vibration
	APU - IR Suppressor, 3-axis Vibration
	Acc GB Input Housing, 3-axis Vibration
	De-icing Cycle Initiation Event(s)

CHAPTER 1003

ELECTRICAL SYSTEMS

1 OBJECT

1.1 The object of the tests of this Chapter is to demonstrate that the electrical system installed in the Rotorcraft is suitable for Service use, in particular that:

(i) The system can be used satisfactorily on the ground with the use of approved sources of electrical power

(ii) Electrical power can be satisfactorily generated on-board the Rotorcraft and utilised to operate actuators, motors and other electrical power dependent equipment, both on the ground and in flight throughout the specified flight envelope, and to satisfy all operational and environmental requirements.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 1 Chapter 706

3 APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new electrical systems in Rotorcraft, and shall be conducted on systems which are fully representative of the final service standard and where the system(s) interfaces with others, they shall also be representative of the service standard.

3.2 The tests are also applicable to electrical systems which are modified to the extent that results of tests are likely to differ from results achieved prior to the modification.

3.3 The need for further testing must also be considered where modifications are made to other aircraft systems which affect the operation or performance of the electrical system.

4 TEST PHILOSOPHY

4.1 The test philosophy assumes that the electrical system clearance activity is divided in specific areas of activity, namely, supplier tests, system rig tests, aircraft pre-flight tests and finally ground/flight trials.

This type of approach is particularly beneficial for new electrical systems where it is essential that comprehensive rig testing of the fully configured electrical system is conducted prior to assessment on the aircraft.

In principle, an electrical rig installation which is fully representative of the aircraft installation will enable a large proportion of the system clearance testing to be conducted without the need for extensive ground testing on the aircraft. The individual components or subsystems which make up the electrical system will have been fully tested by the equipment supplier and compliance with the Requirement Specification will have been adequately demonstrated and reported in the applicable type approval documentation.

The Rotorcraft constructor shall satisfy himself that sufficient evidence of satisfactory operation and performance has been provided by the supplier. Rig testing will then be undertaken by the Rotorcraft constructor to evaluate the operation and performance of the collective equipments when installed in the electrical system, in order to confirm that the individual components of the system are correctly related with one another and perform the required duty for which they are employed.

Following an extensive rig test programme, sufficient testing will normally have been completed to enable a 'system operating' release to be given to operate the system under aircraft ground running conditions.

Therefore, the aircraft ground tests will normally be limited to confirming the system control and indications, general busbar power supply integrity, demonstration of, and recovery from simulated power supply failures and a more thorough assessment of any of the system interfaces which have not been fully investigated during rig testing.

Similarly, any tests which cannot be representatively carried out during the rig testing phase shall be conducted during aircraft pre-flight testing, ground or flight trials as applicable.

Finally, the system will be assessed during a period of aircraft ground and flight trials at which time the system assessment will be conducted under ground and flight conditions representative of the Service role of the Rotorcraft. and shall cover the whole of the operational flight envelope and environmental conditions defined in the Rotorcraft Specification.

5 GENERAL TEST CONDITIONS AND REQUIREMENTS

5.1 Ground and flight conditions shall be representative of the Service role of the Rotorcraft and shall cover the whole of the operational flight envelope and the environmental conditions defined in the Rotorcraft Specification.

5.2 The electrical systems of the aeroplane under test are to be fully representative of the production standard and functioning. The Rotorcraft systems and equipment which interface with the electrical systems in such a way as to be likely to interact are also to be fully representative of the production standard. Where for any reason this is difficult to achieve then agreement shall be obtained from the Rotorcraft Project Director before proceeding with the tests.

5.3 For new electrical systems it is essential that comprehensive rig testing of a fully configured system has been completed and satisfactory operating characteristics established, prior to the commencement of testing. In particular the following shall be taken into account:

(i) The evidence from preliminary ground rig testing, including operating extremes of transmission/rotor rpm, from minimum up to maximum transient conditions. Where multiple generation systems have common interfaces then the interaction of systems one with the other shall be considered.

(ii) The standard of the electrical system(s) and components and their suitability for the tests proposed.

5.4 For "mid term up-dates" and modifications generally to electrical systems where the original ground test rig installation is no longer available and provision of a new rig is deemed to be uneconomic, Rotorcraft ground tests assume additional importance and particular care must be given to the planning and execution of them to establish the necessary level of confidence in the system(s) required for initial flight tests.

5.5 The Electro-Magnetic Compatibility (EMC) characteristics shall be considered before flight trials are undertaken.

Continued assessment will be an activity which cannot be separated from Electrical System Tests and is covered in Chapter 1011.

5.6 The standard of instrumentation employed shall be such that parameters of interest can be measured simultaneously with sufficient accuracy to enable the electrical system characteristics to be established in accordance with the applicable electrical specification'

The standard of data processing equipment shall be selected to meet the same objective and evidence of calibration traceable to national standards provided where necessary.

5.7 Aircraft tests may be conducted at any convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification or, if not so defined, as envisaged by the Service user, taking into account limitations imposed by the carriage of external stores or other loads (i.e. maximum normal acceleration, attitude, speed, etc.).

5.8 On Rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial.

6 ELECTRICAL RIG TESTS

In general terms Electrical rig tests enable the satisfactory collective performance of equipment, which comprise the system, to be established under simulated ground and flight operating conditions.

This will normally be achieved early in an aircraft development programme, substantiating the Design philosophy thereby enabling a high level of confidence to be established prior to flight trials taking place.

6.1 Rig Facilities

The provision of suitable electrical rig test facilities will enable a large proportion of the overall test programme to be conducted prior to aircraft ground and flight trials. Wherever possible, the electrical system installation on the rig should be fully representative of the aircraft installation, ideally built to aircraft standard drawings, employing aircraft standard components and connected using correct aircraft cable type, gauge, length and routing.

Where physical constraints render this approach impractical, every effort should be made to reproduce the equivalent aircraft cable impedance between components.

Generator drive rig facilities should enable the full range of generator functions, load and speed characteristics to be represented.

Aircraft system interfaces which cannot be fully represented (for example, aircraft computer based control and display systems) shall be simulated to enable the electrical system warning and status information to be suitably represented.

Aircraft electrical loads may be simulated by the connection of resistive, inductive and capacitive load banks to the electrical system to enable assessments to be undertaken over the full load conditions (including overload) of the aircraft.

Critical aircraft loads which may impact on the performance of the electrical system shall be evaluated on the rig.

6.2 Rig Tests

6.2.1 Following commissioning, initial rig tests should demonstrate that the design philosophy has been achieved and that the means of selection, control and indication have been implemented correctly.

6.2.2 An in-depth study of the operation of the system may then be conducted to assess the following aspects.

(a) Switching control logic and indication of AC generator, inverter and external power supplies as applicable.

(b) Switching control logic and indication of DC generator, TRU battery and external power supplies as applicable.

(c) Regulation characteristics of AC/DC generators over available speed/load range including limited overload conditions.

(d) System load/volt drop characteristics.

(e) Power supply load sharing capabilities (where applicable).

(f) Battery charging characteristics.

(g) GCU operation including fault testing.

(h) Distribution system and direct fault testing.

(i) Simulated system failures and recovery action.

(j) General power supply quality checks.

(k) Equipment thermal assessment.

(l) Generator cooling medium characteristics.

(m) Power supply interrupts and voltage disturbances arising from system control and load switching.

7 AIRCRAFT PRE-FLIGHT TESTING

7.1 This phase of testing will normally be conducted following hangar clearance of the system by means of the relevant Acceptance Test Procedure, which enables clearance of the installation to be undertaken prior to aircraft operating conditions being established.

7.2 Any of the activities detailed under ‘rig testing’ which cannot be completed successfully shall be demonstrated on the aircraft during the ground or flight test phase as necessary.

7.3 The evidence of an extensive rig test programme will reduce the pre-flight testing phase to one of a confirmatory nature during which the following shall be demonstrated:

(i) The transition from external to internal supplies and vice versa is carried out in a satisfactory manner;

(ii) confirmation of system control and switching logic;

(iii) confirmation of acceptable busbar voltage levels under max/min extremes of electrical load;

(iv) primary power system failures and establishment of secondary or emergency power sources as applicable;

(v) assessment of aircraft services due to loss of busbar supplies;

(vi) assessment of electrical system advisory, cautionary and warning indications;

7.4 During the test period measurements of the steady state.

AC and DC busbar voltage shall be made at selected load conditions.

In addition, suitable recording shall be made of AC and DC busbar voltage disturbances caused by system control operations, load switching, generator speed transients and failure simulations.

8 AIRCRAFT GROUND AND FLIGHT TRIALS

8.1 Satisfactory systems(s) operation shall be demonstrated over the full flight envelope and manoeuvre range of the rotorcraft. Generally the objective is to confirm the results of rig and ground testing and to explore operating regimes and duty cycles that are fully representative of the service role. It is particularly important that the extremes of aerodynamic and acceleration loads, including zero and negative “g” (where applicable) are experienced. In particular checks and measurements shall be made as follows:

(i) Temperatures of generators, TRUs, batteries, battery charging equipment and other temperature sensitive items which may be affected by recirculation of exhaust gases and/or cooling air during prolonged hover, sideways and rearwards flight (including the need to hover other than head into wind).

(ii) Effect of vibration and shock (including that which might arise from gun firing or the firing of air launched missile booster rockets, etc.) on the performance and functioning of contractors, constant speed drives and other mechanical systems and devices.

(iii) Extending the range of system operation and loading beyond that achievable during ground testing. This may include short-term overload conditions (above maximum continuous rating where required) and use of emergency generator(s) which can only be operated in flight such as those driven by ram air turbines (if applicable).

(iv) Failure simulations shall be explored to the extent that is considered acceptable and necessary in flight tests.

(v) Checks shall be made to ensure that adequate and correct indication of system(s) or equipment failure or malfunction takes place and that reversionary or emergency functions are available as required.

(vi) Confirmation of the power supply quality and integrity shall be established over the full flight envelope and Service role conditions to ensure that the requirements of the applicable standards and specifications have been met.

(vii) Checks shall be made on charging characteristics of the rotorcraft to establish that the capacity of charged batteries is adequate to support essential services for required duration following total primary electrical power failure.

(viii) Following a battery cold soak to minimum environment temperature, engine or APU starting shall be demonstrated in accordance with the Rotorcraft Specification.

(ix) The system(s) operate(s) satisfactorily after cold and hot soaks of the rotorcraft, in the inoperative condition, at the extremes of the environmental range as required by the Rotorcraft Specification.

(x) Verification of emergency power supply operation and adequacy of procedures under in flight conditions.

CHAPTER 1004

HYDRAULIC SYSTEMS

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the hydraulic system(s) installed in the rotorcraft is suitable for Service use, in particular that:-

(i) The various actuators, motors and other components of the system can be satisfactorily operated on the ground using approved sources of hydraulic power; and

(ii) Hydraulic power can be satisfactorily generated on-board the rotorcraft and utilised to operate the actuators, motors and other units of the system both on the ground and in flight throughout the specified flight envelope and to satisfy all operational and environmental requirements.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapter 100, 101 and 704.

3 APPLICABILITY

3.1 The need for further testing shall also be considered where modifications are made to systems which interface with the hydraulic system(s). Of particular concern are those affecting loading on actuators and motors, changes to flow demands and directions and heat transfer to the hydraulic fluid. Changes to systems which involve altering fluid volumes or column lengths shall be investigated to ensure freedom from hydraulic resonance whether excited by pump or load characteristics.

3.2 Tests on Alighting Gear and Powered Flying Controls are not included except in so far as they affect the hydraulic system as a whole, as tests on these systems are covered by Chapters 1008 and 1010 respectively.

4 INSTRUMENTATION

4.1 The standard of instrumentation shall be such that a continuous recording can be obtained of flight conditions (e.g., IAS, Altitude, OAT, Rotorcraft attitudes) and hydraulic system conditions (e.g., pressures, including transients and pulsations; displacements, temperatures, fluid flow rates and vibration) during both ground and flight tests.

4.2 The suggested parameters to be recorded to meet the requirements of this chapter are detailed in Leaflet 1004/1.

5 LOADING

5.1 The tests may be made at any convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification or, if not so defined, as envisaged by the Service user, taking into account, limitations imposed by the carriage of external stores or other loads, i.e. maximum normal acceleration, attitude, speed etc.

5.2 On rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial (e.g., the effect of rocket/missile launch or gun firing on engine behaviour).

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions shall be representative of the Service roles for the rotorcraft and shall cover the whole of the operational flight envelope and the environmental conditions defined in the Rotorcraft Specification.

6.2 For new hydraulic systems it is essential that comprehensive rig testing of a fully configured and representative system has been completed and satisfactory characteristics established prior to commencement of the tests which are subject of this chapter. In particular the following shall be taken into account:-

(i) The evidence from preliminary ground rig testing, including any tendency towards hydraulic resonance up to at least the maximum possible pump pulsation frequencies (i.e. including transient overspeed conditions) and the possibility of large transient sub-systems. Where multiple hydraulic systems have common interfaces then the effects of operation of systems one upon the other shall be considered.

(ii) The standard of the hydraulic system(s) and components and their suitability for the tests proposed.

(iii) Natural frequency (rapping tests) on major components and pipe runs.

7 GROUND TESTS

7.1 Tests shall be conducted on the systems utilising:-

(i) External hydraulic supply(ies) as appropriate to the rotorcraft.

(ii) All on-board hydraulic power sources, i.e. main supplies (primary, secondary and utility if so designated), auxiliary and emergency supplies (where applicable) throughout the range of system operating configurations provided, for which testing on the ground is practicable.

7.2 Generally it shall be established that system(s) can be operated in accordance with the requirements laid down in the Rotorcraft Specification and as previously demonstrated on the ground rig. In particular it shall be demonstrated that:-

(i) Means for selection, control and monitoring of systems (including coupling and decoupling in the case of external supplies) is satisfactory.

(ii) The accessibility of all components is satisfactory with respect to bleeding and replenishing of reservoir fluid, replacement of filter elements and checks necessary to establish the state of the system(s).

(iii) Operation of the combination of actuators, motors, etc., which is most demanding in terms of flow requirement (within the range of normal or emergency utilisation) does not result in the pressures at any point in the system(s) falling below the minimum acceptable (for the particular system configuration) and that the time/rate of operation of each actuator, motor, etc., is in accordance with specification requirements.

(iv) Each actuator, motor, etc., operates satisfactorily when selected and only when selected and there is no detrimental interaction between actuators. The actuators shall operate smoothly and without hesitation or instability over full operating ranges.

7.3 Tests as follows shall be carried out to ensure that the systems environmental and duty cycle requirements are capable of being met:-

(i) At zero loading of the system, i.e., with no actuators, motors, etc., operating, the steady state pressures, fluid and component temperatures do not exceed the maximum and minima permitted and the system(s) operate(s) with no evidence of instability or extreme pressure fluctuations.

(ii) Ascertain that prolonged application of wheel brakes during taxying does not cause local fluid temperatures to exceed limitations nor pressures to go outside the range necessary for satisfactory operation.

(iii) Ascertain that the normal or emergency operation of rotor brake does not cause fluid temperature to exceed limitations nor the pressure to go outside the range necessary for satisfactory operation.

(iv) The system(s) operate(s) satisfactorily after cold and hot soaks of the rotorcraft in the inoperative condition, at the extremes of the environmental range as detailed in the Rotorcraft Specification.

(v) System(s), drains and vents shall be observed to ensure that excessive loss of fluid does not take place at environmental or duty cycle extremes.

7.4 Tests shall be carried out to demonstrate that:-

(i) Such failures as can be readily represented; i.e., partial, single, multi-engined or APU failures or shutdowns (where applicable) result in correct warning indication, automatic changeover to standby or emergency power (where applicable) and continuing satisfactory operation of system(s) (or specified part(s) of system(s)) in accordance with requirements.

(ii) When gas blow down systems are used for emergency power in the hydraulic system for lowering of the alighting gear, flaps or other devices, that the gas can be vented from the system(s) after use, and the hydraulic system(s) readily restored to normal configuration.

8 FLIGHT TESTS

8.1 Satisfactory system(s) operation shall be demonstrated over the full flight envelope and manoeuvre range of the rotorcraft. Generally the objective is to confirm the results of rig and ground testing and to explore operating regimes and duty cycles that cannot readily be produced on the ground. It is of particular importance that the extremes of aerodynamic and acceleration loads, including zero and negative "g" (where applicable) are experienced. Particular checks and measurements shall be made as follows:-

(i) Pressure throughout the system(s) including transients to ensure that the margins between supply pressures and actuator, motor, etc., requirements are adequate.

(ii) Temperatures throughout the system, particularly of components which may be affected by recirculation of exhaust gases or cooling air during prolonged hover, sideways and rearwards flight (including need to hover other than head into wind).

(iii) Vibration (whether mechanically impressed or system excited) throughout the system(s) or evidence of any form of hydraulic instability.

8.2 Failure simulations shall be explored to the extent that it is considered acceptable and necessary in flight tests.

8.3 Checks shall be made to ensure that adequate and correct indication of system(s) or equipment failure or malfunction takes place and that reversionary or emergency functions are available as required.

8.4 The tests of para 8.2 and 8.3 shall include the operation of all hydraulic accessory equipment and an assessment of effects of such system failure on these items shall be made, together with the possible adverse effect(s) on the main system of any individual equipment failures.

8.5 On completion of a flight test and after engine/APU shut down and rotor stop, hydraulic system(s) are to be checked as follows:-

(i) Reservoir fluid levels are correct, and in multi-system installations there has been no fluid transfer between systems.

(ii) That there has been no unacceptable loss of fluid from the system(s) nor any component or accessory leakage (either seals, couplings, etc.,) or excessive venting of fluid.

(iii) Any depressurisation method operates satisfactorily and after depressurisation of the system(s) that the initial gas charge of all accumulators is satisfactory.

Note: Much of the flight testing of this chapter can be readily carried out in association with the tests required by Chapters 1008, 1010, 1016 and 1017 and those required by Part 9 (Handling).

LEAFLET 1004/1

HYDRAULIC SYSTEMS

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are considered to be the minimum for the tests detailed in Chapter 1004. Such other parameters as are necessary to establish the ability of the Hydraulic System(s) to function and perform to specification in the rotorcraft environment should be determined by consultation with the equipment manufacturer(s)/supplier(s) and the Official Trials Establishments and agreed with the Rotorcraft Project Director.

ITEM No	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
5	Altitude (pressure)
7	Total Temperature (or Ambient Temp/OAT)
9	Pitch Attitude
10	Roll Angle
12	Heading
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
20	Landing Gear Position
25	Stick Position (Pitch)
26	Stick Position (Roll)
27	Yaw Control Pedal Position
36	Yaw Control Positions
37	Port Brake Pressure
38	Starboard Brake Pressure
39	FCS State/Mode
100	Collective Pitch Control Position
103	Main Rotor Speed
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
121	Rotor Brake Temp
125	Fore and Aft Servo Position
126	Lateral Servo Position
127	Collective Servo Position
128	Yaw Servo Position
129	Pitch Series Actuator Position
130	Roll Series Actuator Position
131	Collective Series Actuator Position
132	Yaw Series Actuator Position
	Hydraulic Pump Fluid Outlet Temperatures
	Hydraulic Reservoir Fluid Temperatures
	Hydraulic Pump(s) Fluid Inlet Pressure
	Hydraulic Pump(s) Fluid Delivery Pressure
	Hydraulic Accumulator Fluid Pressure
	Main Servo(s) Axial Vibration
	Secondary Servo(s) Axial Vibration
	Tail Servo Axial Vibration

CHAPTER 1005

FUEL SYSTEMS
GENERAL FLIGHT TEST REQUIREMENTS

1. OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the fuel system installed in the Rotorcraft is suitable for service use and has satisfactorily provision for:-

a) Ground refuelling and defuelling

b) Supply of fuel to engine(s), APU(s) and other auxiliary device(s) where applicable.

c) Jettison of fuel and fuel tanks where applicable.

d) Air-to-Air/Ship-to-Air refuelling (where applicable) in accordance with Rotorcraft Specification requirements.

2. RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 VOLUME 2 Chapters 701, 702, 712 and 806.

3. APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new Rotorcraft fuel systems and shall also be considered for application where modifications have been made which are likely to affect the Rotorcraft and/or system characteristics and hence the results of the tests. The extent of such testing shall be the subject of discussion and agreement with the Rotorcraft Project Director during the Project Definition Phase and be incorporated in the Rotorcraft Specification.

3.2 Much of the testing required by this chapter can and should be carried out in association with the tests of Chapters 906, 1001 and 1002 since many of the test and instrumentation requirements are both common and interactive.

4. INSTRUMENTATION

4.1 The standard of instrumentation shall be such that a continuous recording is obtained of the flight conditions (eg. IAS, Altitude, OAT, rotorcraft attitude, etc.) fuel system conditions and performance (eg. pressure(s), temperature(s), tank contents, flow rates, quantity used etc) and, to a limited extent, interfacing systems - in particular engine(s), APU(s) and auxiliary devices supplied with fuel.

4.2 The suggested parameters to be recorded to meet the requirements of this chapter are detailed in leaflet 1005.1.

5 LOADING

5.1 The tests may be made at convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification, taking into account limitations imposed by the carriage of external stores or other loads (ie. maximum normal acceleration, attitude, speed etc.). Where the Rotorcraft Specification requires the ability to manage the C of G position by fuel transfer, the specific C of G positions should be nominated and the ability to achieve them by redistribution of fuel demonstrated.

5.2 Where analysis and/or rig tests show that the aircraft attitude change can adversely affect unusable fuel levels, the aircraft should be loaded to the worst CG position for the flight tests. Furthermore, for flight safety reasons it would be advisable to keep the AUM to a minimum commensurate with conducting the flight test.

5.3 On rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial (eg. the effect of rocket/missile launch or gun firing on engine behaviour).

6. GENERAL TEST CONDITIONS

6.1 Ground and flight conditions shall be representative of the Service roles for the Rotorcraft and shall cover the whole of the operational flight envelope and the environmental conditions as required by the rotorcraft specification.

6.2 The test shall be conducted on systems which are fully representative of the final Service standard, and where the fuel system interfaces with other systems, they shall also be fully representative of the Service Standard.

6.3 When the use of more than one type of fuel is specified, test shall be made with the normal fuel to demonstrate satisfactory operation of the fuel system and the limitations, if any, which must apply. These tests must include the use of fuel which has been heated or cooled to the maximum and minimum extremes which are defined in the Rotorcraft Specification. Additional tests shall be carried out with the other specified fuels at particular flight conditions where any characteristics of the fuel could conceivably have limiting effects.

6.4 For new fuel systems it is essential that comprehensive testing of a fully configured and representative system ground rig in accordance with DEF STAN 05-123 Chapter 230 has been completed and satisfactory system characteristics established prior to commencement of the tests which are the subject of this chapter.

6.5 The need for further testing shall also be assessed where modifications are made to systems which interface directly with the fuel system(s). In particular engine(s) and engine control system(s) which make direct “demand” on the fuel system must be taken into account.

6.6 The following shall be taken into consideration prior to testing:-

a) The standard of the fuel system(s) and the suitability for the test(s) proposed.

b) Natural frequency (rapping tests) on pipes and components of the system(s).

c) Fire detection and suppression system(s) tests.

7. GROUND TESTS

7.1 REFUELLING

7.1.1 It shall be demonstrated that refuelling using both open orifice and pressure methods (where applicable) can be carried out satisfactorily. Generally, ease of operation and compatibility of refuelling equipment are to be assessed. In the case of pressure refuelling a major objective is to establish that the time for complete refuelling is within that required by the Rotorcraft Specification and Chapter 701 paras 3.2.1. Tests shall also ensure compliance with Chapter 701 paras 3.2.2 and 3.2.3. It is important to check whether refuelling can be safely carried out with engines running without rotors turning and then with rotors turning. Other aspects are to be checked as follows:-

a) Open orifice refuelling - draining of any fuel spilt around orifice(s)

b) Pressure refuelling - Sealing of connector(s)
Pressure of fuel supplied
Sequence of filling tanks
Tank pressures (including vents)
Operations of shut off valve(s) (including measurement of surge pressures)
Quantity of fuel supplied (including behaviour of contents of gauging system)
Drainage of defuelling lines
Refuelling overflow (including tank pressures, fuel drainage, tank vents etc.)

On completion of pressure refuelling tests all pipes joints, couplings, tank attachments, interconnections, vents, etc are to be checked for security and leakage.

7.2 DEFUELLING

7.2.1 Defuelling is to be demonstrated using whatever means are applicable to the rotorcraft including gravity (utilising tank and system drains) jettison facilities (gravity and/or pumped) and suction (where provision is made). Effectiveness is to be assessed by determining the time to defuel (and reinstate system) and the quantity and location of residual fuel within the rotorcraft system, when defuelling in accordance with the laid down servicing procedures.

7.2.2 In the case of suction defuelling, assessment shall also include satisfactory operation at suction pressures defined in Chapter 701 para 3.3.1 establishing that the depression throughout the system and particularly the tank(s) is acceptable and that a steady continuous defuel flow is maintained until the tanks are empty. Attention shall be paid to the behaviour of the contents gauging system during suction defuelling and any anomalous behaviour investigated. Tank and vent pressures shall be monitored throughout the tests. Tank couplings, interconnections and structural attachments shall be inspected on completion of tests to ensure integrity is retained and any tendency for tanks to collapse is to be investigated.

7.2.3 General ease of operation and compatibility of defuelling equipment are also to be assessed.

7.3 FUEL SUPPLY TO ENGINE(S), APU(S) ETC

7.3.1 Tests shall be carried out to ensure that the fuel contents gauging system(s) and the fuel flow metering system(s) can be calibrated and read to the required accuracy over the full range of rotorcraft attitudes which are achievable in steady state flight.

7.3.2 The system(s) shall be operated to meet the full range of fuel flow demand on the engine(s), APU(s) and auxiliary devices. Generally it shall be established that the system(s) operate(s) in the way intended and as previously demonstrated on the system rig. In particular the following shall be monitored:-

a) Pressures and temperatures throughout the system, particularly supply pressure to each of the demanding units.

b) Operation of the fuel transfer and crossfeed system, including auxiliary and ferry tanks.

c) Low level contents gauging accuracy and useable fuel contents.

7.3.3 Satisfactory operation is to be demonstrated using:-

a) Standard external ground power supply units.

b) Rotorcraft standard internal electrical supplies (engine or APU driven generator(s)).

c) Rotorcraft auxiliary internal supplies as provided eg craft main, standby, auxiliary emergency batteries, etc.

8. FLIGHT TESTS

8.1 The following tests shall be carried out to demonstrate that the fuel system(s) is (are) capable of providing adequate supply of fuel to the Engines, APU, etc under all the operational conditions required by the Rotorcraft Specification throughout the flight envelope, including the use of maximum and minimum performance and where achievable, zero and/or negative g levels:-

8.1.1 A maximum rate climb from ground level to maximum altitude using the standard fuel pre-heated to +50^{0}C to check the adequacy of the venting system(s) and the susceptibility of the fuel supply system to vapour locks. This shall be carried out with booster pump(s) on and off.

8.1.2 A low level flight at maximum power or maximum torque (where the transmission is the limiting factor) to determine the adequacy of the fuel supply, the fuel pressure above vapour pressure, the maximum fuel flow rate(s) and the efficiency of the fuel tank venting system(s).

8.1.3 A maximum rate of descent, with engine(s) throttled to flight idle, from maximum altitude down to 2000ft pressure altitude to check the effectiveness of the tank(s) inward venting system.

8.1.4 Manoeuvring flight tests and where applicable take-offs and landings in various directions on sloping ground to establish that the fuel system will continue to provide the required fuel flow(s) to the engine(s), with an adequate pressure margin (21 kpa - 3 psi).

8.1.5 Where pressure is tapped or bled from an engine compressor for the prevention of surge or for use in other systems such as cabin conditioning, intake de-icing, etc these systems, bleeds, etc, shall be operated in flight to determine that there are no adverse effects on fuel tanks pressurisation and/or air operated fuel transfer system(s). Where such bleed etc are normally automatic in operation consideration should be given to installing manual controls for these tests.

8.2 Tests shall be carried out to assess fuel contents/gauging during all flight conditions, in particular low level indication (eg adequate margin from point at which fuel supply is no longer satisfactory due to onset of air ingestion, allowing for fuel transfer where relevant). These tests are to take into account the use of auxiliary or ferry tanks where relevant.

8.2.1 In conjunction with rig testing, the flight tests should be designed to identify:

a) The minimum safe fuel contents (per tank) for flight and any associated limitations (eg operations on sloping ground, rapid transitions, extreme aircraft pitch or roll attitudes etc.). This testing may need to take into account the roles of the aircraft.

b) Any peculiar to type fuel management activity required to ensure best use of fuel at low levels (eg manual transfer, crossfeeding).

8.3 The tests called up in Chapter 906 shall be considered as part of the fuel system(s) clearance, since in the case of rotorcraft the engine fuel system has a direct influence on the rotor speed control, which in turn can affect flying and handling qualities.

8.4 FUEL AND FUEL TANKS JETTISON

8.4.1 Tests shall be conducted to establish:-

a) That fuel can be jettisoned safely at an adequate rate whilst maintaining an adequate supply of fuel to the engine(s) at a power setting capable of sustaining level flight at the rotorcraft maximum all up mass and is automatically stopped at the specified minimum level. These tests shall also ensure that fuel so jettisoned does not cause contamination of airframe, rotor(s) or external stores to the extent that performance or function is impaired. Where fuel could come into contact with hot surfaces and present a fire hazard (eg IRCM devices, exhausts etc.), specific jettison tests may be necessary with areas of the airframe, in the vicinity of such heat sources, covered in a white water soluble paint to indicate the likelihood of contamination. Arrangement must be made for such surfaces to be cold during these tests.

b) That external auxiliary fuel tanks can (where applicable) be jettisoned at any fuel state, without interruption to the fuel supply to the engine(s), significant fuel spillage or strikes on the rotorcraft structure or external stores. These tests shall also confirm that tanks so jettisoned cannot, under any reasonable circumstances, become a hazard to the rotor(s). (Wind tunnel tests of tank jettison shall be taken into consideration when planning this part of the flight trials).

c) Fuel jettison is satisfactory when undertaken during a maximum rate decent at a speed judged to represent the worst condition.

8.5 AIR-TO-AIR/SHIP-TO-AIR REFUELLING (where applicable)

8.5.1 Rotorcraft which have a capability to receive fuel whilst in flight shall be tested to demonstrate:-

a) On the fuel rig, prior to any in flight refuelling:

i) that the rotorcraft C of G is not moved significantly due to the uneven distribution of fuel entering the tanks.

ii) that the surge pressures resulting from refuelling valve closure are within the design limits (refer Chapter 710, para 3.2).

iii) that the vent system(s) is adequate to prevent an excessive build up of pressure in any of the tanks, even under failure conditions.

b) For Hover In Flight Refuelling, compatibility with specified in-service UK and allied rotorcraft carrying ships and shore installations, and to establish procedures.

c) i) For Air-to-Air Refuelling, compatibility with specified in-service UK and allied tanker aeroplanes/rotorcraft establishing the speed/altitude envelope for possible transfer of fuel.

ii) The In Flight Refuelling procedure, establishing adequacy of the probe and signalling lights. When night refuelling is specified, adequacy of the probe illumination shall be established.

iii) In Flight Refuelling Flow rates, both at full rate as required in the rotorcraft specification, and at half rate.

d) That on separation of the probe from the drogue (or hose coupling from the rotorcraft) the fuel spillage is not such as to cause significant deterioration in aircrew vision by obscuring transparencies, does not enter engine, APU intake(s) in sufficient quantities to cause the engine(s) or APU to surge, does not contaminate external stores so as to affect their performance or function or enter any part of the rotorcraft where lodging fuel could become a hazard. This test also includes emergency breakaways.

LEAFLET 1005/1

FUEL SYSTEMS
GENERAL FLIGHT TEST REQUIREMENTS
TEST INSTRUMENTATION PARAMETERS

1. INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc, are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are proposed for the tests detailed in Chapter 1005. Other parameters may be necessary to establish the ability of the Fuel Systems to function and perform to specification in the rotorcraft environment.

ITEM NO.	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
5	Altitude (pressure)
7	Total Temperature (or ambient Temp/OAT)
9	Pitch Angle
10	Roll Angle
11	Sideslip Angle
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
51	Fuel Flow
52	Fuel Temp at Flowmeter
100	Collective Pitch Control Position
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
	HP Fuel Pump Inlet Temperature
	Fuel Tank Temperature
	Fuel Pressure at Engine/Airframe Interface
	Fuel Pressure at APU/Airframe Interface
	Booster Pump Outlet Pressure
	Tank Vent Pipe(s) Pressure
	Transfer Pump Outlet Pressures
	Pressure Fuelling Valve Inlet Pressure
	Low Press Fuel Valve Position
	Fuel Transfer Valve Position
	Pressure Refuelling Valve Position
	Jettison Valve Position

CHAPTER 1006

ICE PROTECTION SYSTEMS
GENERAL FLIGHT TEST REQUIREMENTS

1. OBJECT

1.1 The object of the tests of this Chapter is to demonstrate, to the maximum extent practical, that any ice-protection systems installed on the Rotorcraft will protect it in all the icing, snow and mixed conditions as laid down in the Rotorcraft Specification.

1.2 Where the evidence gathered during flight test does not correspond precisely to the range of conditions specified, best use shall be made of modelling, analysis and rig tests to predict the system capability in the specified conditions. The object of this further analysis is to satisfy the Rotorcraft Project Director that, when the results of such analysis are taken in combination with the flight test evidence, there is confidence that the ice protection systems will provide protection throughout the range of required conditions.

1.3 If the combination of flight tests and other data is insufficient to fulfil this objective, a recommendation for a limited CA release will be defined based upon the available evidence. The extension of this release will depend upon further flight tests and in-service experience.

1.4 There will be an initial phase of testing during which the rotorcraft's basic capability for flight in icing is assessed, with particular emphasis being placed on establishing the adequacy of anti-icing and de-icing in relation to vulnerable areas. From this may arise a requirement for some secondary protection (eg aerials, areas outside the designed intake coverage, etc).

1.5 Tests will confirm the integration of protection system with the rotors, airframe and other systems.

1.6 Testing does not cover general clearance/handling in icing conditions, whether or not the rotorcraft is fitted with ice protection systems. These aspects of testing will be covered by (an amendment to) Part 9. Testing must, however, cover the consequences, or effectiveness, of ice protection systems failure or degraded mode operation.

2. RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 101, 703, 705 to 708, 710 to 713, 715 to 718, 800, 806.

3. APPLICABILITY

3.1 The tests in this Chapter must conclude with assessment on a rotorcraft with an ice protection system representative to the Service standard with all relevant ice protection systems operative, i.e. engine air intakes, rotor anti/de-icing, etc.

3.2 Protected areas and areas requiring specific assessment regardless of protection may include:-

a) Aerofoils - including rotor blades, wings, stabilisers, etc which may be part of a particular rotorcraft type, ie compound helicopter, tilt rotor/wing.

b) Propellers, rotor heads and associated mechanisms.

c) Moving surfaces, ie slats, flaps, controls, trim tabs, undercarriage and weapon bay doors.

d) Engine, auxiliary and cooling intakes and exhausts.

e) Air data system and associated sensors including pitot and static heads/ports and masts.

f) External protuberances such as aerials, external sensors and wire-strike protection cutters.

g) Weapons and weapon carriers/launchers.

h) Transparencies, including those of weapon sensors and sighting systems.

j) Vents and drains.

k) Other specialised items not covered by the above, including any external role(s) equipment such as underslung or external load equipment.

3.3 The two basic types of active ice protection systems are as follows:-

a) Anti-icing: Systems used to prevent formation of ice on part of the surface of a rotorcraft, usually achieved through continuous and/or rapid cyclic heating of the relevant part of the rotorcraft surface, but can be based on the use of freezing point depressant fluids/pastes in certain limited applications.

b) De-icing: Systems used to periodically remove ice from part of the surface of a rotorcraft, before it reaches the point where it would cause rapid and unacceptable degradation of the rotorcraft and/or system(s) functioning/performance or would hazard the rotorcraft during shedding.

4. EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 Instrumentation will, to some extent, depend upon the assessment of risk related to flying the particular rotorcraft in icing conditions. This will be based upon calculation, modelling and rig tests (icing tunnel, etc.), previous experience with the rotorcraft and the extent of the icing envelope to be investigated. Many of the parameters required for these tests are common with those of Part 9 and of several chapters of Part 10. [Refer particularly to Chapters 1001, 1003, 10015 and 1016, also to the relevant accompanying leaflets]. The instrumentation specific to this Chapter is discussed in detail in Leaflet 1006/1.

4.1.2 All normal flying and general instruments shall be fitted and serviceable for the tests of this Chapter. In particular the instruments and their standard of calibration shall be such that the crew can monitor all critical aircraft and atmospheric parameters. Special test instrumentation shall be provided where the standard cockpit flight instruments are considered inadequate for this purpose, eg OAT, Liquid Water Content (LWC), rotor control loads/stresses, etc. Obtaining, installing and calibrating these instruments, shall be the responsibility of the contractor in accordance with the requirements of Def Stan 05-123 Chapter 240.

4.1.3 Icing on main and tail rotor blades can result in significant and rapid increases in rotor head/blade stresses and control loads. It is important that the instrumentation includes appropriate load monitoring in all critical areas and that on-board visual indicators are provided. Whilst telemetry may be employed, if available, to reduce the number of channels in need of monitoring by aircrew on the test vehicle, the most critical parameters must still be available for continuous monitoring by aircrew in the event of loss of telemetry during critical phases of the test programme.

4.1.4 The effects of icing, ice/slush/snow ingestion or intake blockage upon engine behaviour is an area which may become evident in the form of torque instability, and/or mismatching in the event of multi engineered vehicles, so that appropriate supplementary engine instrumentation shall be considered.

NOTE: Every effort shall be made during the design phase of the instrumentation to ensure that the sensors and other external instrumentation such as cameras and their mountings are unlikely to introduce an extra element of hazard into the trials or to appreciably modify the basic Rotorcraft ice accretion characteristics. Where possible, these problems should be investigated using icing simulation methods prior to commencement of trials and, in extreme cases, simulated shedding techniques may be used such as wax block methods (see para 6.5) to determine the degree of extra hazard introduced. Where the accretion of ice cannot otherwise be prevented then either heating or chemical anti/de-icing should be considered to ensure that no restriction upon trials flight is introduced.

4.2 VISUALISATION

4.2.1 The use of rotor/engine intake visualisation is considered important and essential to the understanding of the variable characteristics of the ice accretion which will be encountered in natural icing conditions.

4.2.2 Wherever possible, visual monitoring of the accumulation of snow/ice in all critical areas shall be provided together with video or cine-recording where warranted. Where visual monitoring is not practical, flight trials must be restricted to accommodate possible hazards with a progressive expansion of activities subject to regular satisfactory post flight inspections. To achieve this, it will be necessary to make every effort to ensure all potentially hazardous accretions of ice/slush or snow are retained by the airframe until visual inspection can be accomplished.

4.2.3 Where rotor head and other limitations permit, camera(s) shall be mounted to record the build-up, shedding and run-back of ice on upper and/or lower surfaces of main rotor blades. These may be configured to view all blades simultaneously or singly, and shall have a means of correlating data with other on-board monitoring systems. These cameras shall be subject to development testing to ensure that they do not add to the hazards of the trial by aggravating the overall vibration levels, the airframe/rotor stresses and loads or to the rotor-head icing. They shall also be subject to EMC investigation, both as to possible cause and effect.

4.2.4 Prevailing ambient atmospheric conditions shall be recorded (the relevant list should include outside air temperature, liquid water content, droplet size, snow severity and altitude).

4.2.5 An ice accretion meter should be provided for the pilot to enable identification of the onset of airframe icing, ice type and amount. If a calibrated Vernier Accretion Meter is not practical on the aircraft type, an appropriate part of the airframe shall be identified for this purpose. Consideration shall be given to the clearance of the meter once it has accreted the maximum it is capable of measuring.

5. LOADING

5.1 The tests must be made over a range of loading and centre-of-gravity positions to ensure that the achievable flight envelope is not significantly more restrictive than the maximum envelope for which release is sought taking into account the limitations imposed by the carriage of external stores or other loads, ie maximum normal acceleration, attitude, speed, etc.

5.2 On rotorcraft powered by a single engine, external loads should not be carried if the tests may either result in or require the engine to be shut down in flight, unless the load is specifically required for the trial .

6. GENERAL TEST CONDITIONS AND REQUIREMENTS

6.1 The service operational requirements need to be clearly defined in terms of, severity of snow and icing encounters to be considered, the duration of any and each icing encounter and the number of icing encounters per flight. The level of acceptable rotorcraft or system performance degradation shall be specified in the requirement.

6.2 The ice protection systems under test shall be representative of the production standard rotorcraft. Ground and flight conditions shall be representative of the Service role of the rotorcraft and the range of climatic conditions for which clearance is required.

6.3 In particular, the following shall be taken into account prior to testing:-

a) The evidence from ground rig tests, together with any previous experience of operation with similar equipment or installations.

b) The protection systems manufacturers limitations on the equipment under test and evidence from any previous experience with the rotorcraft in icing conditions with or without protection.

c) Test evidence from the Engine manufacturer as to the engine's liquid water, snow and ice ingestion tolerance and ice protection system qualification.

d) The various snow and icing conditions to be examined, including blown, recirculating snow and freezing rain/drizzle.

e) The status and condition of any ice protection system including failure and degraded modes where appropriate.

f) The need to test different configurations of other aircraft systems that may affect the ability of the aircraft to operate in snow or icing e.g. Environmental Control System ON/OFF

6.4 Datum engine and aircraft performance and stress/load measurements shall be undertaken, on the fully equipped trials rotorcraft, out of icing conditions, and should be completed and assessed in advance of any icing sortie.

6.5 In areas where the likelihood of large accretions is judged as high, shedding tests may be necessary before the aircraft is exposed to significant levels of ice accretion. Such tests, possibly using wax blocks, can identify the probability of ice shedding and impacting vulnerable areas.

6.6 Icing trials with a hitherto untried rotorcraft have to be approached with the understanding that it is a high risk area. The choice of icing trials venue must be made carefully to ensure that the following important factors are taken into account:-

a) The best probability of icing or snow conditions at the temperatures and severities required.

b) Good servicing facilities including heated hangarage to ensure fast turn around between icing sorties and safe, efficient maintenance in a hostile environment.

c) Effective logistic support for rotorcraft and trials equipment spares and for personnel.

d) Good air traffic facilities with an efficient meteorological service, radar cover and safety services and ideally a dedicated area for icing encounters.

e) Safe, uncluttered terrain against the possibility of a forced landing.

f) Line features, e.g railways or roads, (ideally connecting with the airfield) to give good visual cues and enable safe snow flying to take place.

g) Flat terrain to increase the probability of non-icing air between the icing level and ground.

7. GROUND TESTS

7.1 Validation of qualification evidence of windscreen services shall be provided by cold soak ground tests where the windscreen and cockpit transparencies have been allowed to mist up or ice over, followed by operation of:-

a) Normal windscreen/transparency demist/ice protection systems to assess the adequacy of clear vision throughout the pre-take off period.

b) Engine(s) and/or APU operating in accessory drive (where appropriate), air conditioning to maximum heat setting and windscreen/transparency heat to maximum boost to assess adequacy of clear vision for take off and hover.

c) Any standby demist/de-ice system which may be fitted.

7.2 Engine and APU (where appropriate) running should be carried out over the widest range of power possible on the ground to establish any condition at which the onset of intake icing may occur for any intake configuration for which release is sought. All configurations of engine air intake shall be qualified by evidence gained using appropriate wind tunnel icing facilities or an equivalent simulation. Consideration must be shown to be given to the validation of the qualification by means of test evidence gathered from full scale aircraft tests over the range of conditions required by the Rotorcraft Specification. This is of particular importance in conditions for which accurate simulations are not available. Note that under some conditions this could be at ambient temperatures as high as $+5^0$C. These tests shall also establish lower temperature ground running procedures for Service use and any necessary operational restrictions.

7.3 Taxying and ground running operations should be assessed to determine the extent of airframe and/or rotor contamination from the effects of spray/slush thrown up from the ground by wheels (if fitted) and rotor down wash. A preliminary assessment of the effects of recirculating snow on engines, rotors and pilot/crew vision may also be made during this phase of testing. The assessment should encompass both take off and landing configurations if different. Consideration shall then be given to the possible effects of the resultant spray/slush freezing and causing interference with normal operation of undercarriage, undercarriage doors, adjustable control surfaces and flying controls or other systems during and after take off.

7.4 Following operation of ice protection systems there shall be a post-flight assessment of the extent of run-back of melted snow/ice on rotor blades, engines, flying controls and control surfaces where subsequent refreezing could occur to the detriment of their performance. It must be remembered that similar hazards may occur, particularly as a result of flight in precipitating and recirculating snow, due to snow melt-water draining from the aircraft drains, bays and panels. Particular attention must be paid to resultant effects upon flush-fitted vents such as employed for fuel and air data systems.

7.5 Where the ice protection system has built-in-test equipment (BITE) then the performance of this equipment should be assessed against the known status of the protection system throughout the tests.

7.6 The design of the ice protection systems should be such as to preclude potentially damaging effects in the event of deliberate or inadvertent operation outside the range of the design envelope. However, where system operation is controlled only by pilot action and no other protection or restriction is applied, tests may be necessary to establish the effect on the airframe structure, rotors and other systems resulting from such circumstances. These tests shall, where the design of the Rotorcraft electrical or protection systems permit and an argument of possible effect can be sustained, be undertaken both with and without engines/rotors running as applicable.

7.7 In order to minimise risk during subsequent flight testing, consideration shall be given to conducting a series of tests on the ground using appropriate wind tunnel icing facilities or an equivalent simulation. Aspects that might usefully be addressed are:-

a) The ability of the ice detector(s) to detect icing conditions and its (their) response time to those conditions.

b) Functioning of pitot/static head and mast heaters under icing conditions.

c) Operation of low airspeed sensing systems and heating (where provided) under icing conditions.

d) Functioning of the ice protection systems of the rotor(s) under normal and partial failure conditions.

e) Functioning of engine and APU intake ice protection systems.

f) Ice shedding from unprotected parts of rotors, airframe and intakes to assess the possibility of significant quantities of ice being ingested by engine(s)/APU and general FOD risk to the rotorcraft.

g) The effects of ice accretion on primary and secondary flying controls.

h) Assessment of clear vision for the pilot(s) over the specified operational envelope of the rotorcraft.

j) Assessment of the effects on rescue winch(es), MAD winches, weapon carrier installations, etc and the effect this has on operation and jettison capability.

k) Initial assessment of the possible effects of frozen slush on the operation of undercarriage and undercarriage/weapon bay doors.

m) An assessment of the effects of ice or snow blocking engine or APU screens, fuel system vents, cooling air ducts, etc. Such an assessment should include the effects on the mechanical integrity of the screens

8. FLIGHT TESTS

8.1 Flight in icing conditions with an untried rotorcraft and/or ice protection system(s) is an area to be approached with extreme caution with appropriately restricted initial sorties into icing conditions and progressing slowly to a less restrictive envelope. (see Leaflet 1006/2 for notes on Flight Safety aspects of these trials).

8.2 The extent and scope of test in natural snow and ice conditions will depend upon the rotorcraft configuration and required operational envelope sought. The level of test necessary to substantiate qualification evidence previously gained by rig testing of specific ice protection system and components may also vary and the following specific areas of investigation may thus be reviewed on merit subject to agreement with the Rotorcraft Project Director.

8.3 The gathering of meteorological data throughout all icing encounters, especially OAT, liquid water content, droplet size and ice accretion is essential to the support of the qualification and substantiation programme and a critical aid in the evaluation of any possible limiting conditions and interpretation of data gathered throughout the trials.

8.4 Aspects to be addressed in the initial phase should encompass:

a) Establishment of electrical power demands and adverse effects of operation of ice detection and protection systems on airframe, rotor and other systems by operation in a range of conditions including clear air at temperatures from $+5^{0}$C down to -10^{0}C and progressively colder.

b) Establish by flight in a range of conditions, the correct functioning of ice detection systems (if fitted) and the relationship with airframe ice build up. Observation shall be maintained to establish all cues that the rotorcraft has entered icing conditions, eg; ice formation on unheated areas of windscreen/transparencies, windscreen wipers (where fitted) and/or structure in the immediate view of pilot/crew.

c) The extent of ice formation on the intake protection systems along with the associated effects of intake blockage on engine operation.

8.5 Having established a degree of confidence in the basic capabilities of the aircraft systems and any areas of the airframe that may result in restrictive or limited activities, testing should then be extended progressively to cover the following:-

a) Flight in icing conditions to substantiate the performance of all ice protection fitted including engine air intakes, rotors and general airframe icing.

b) Flight in precipitating and recirculating snow.

c) To demonstrate that adequate pilot/crew vision is maintained under all required operational conditions by use of available transparency demisting/ice protection and other windscreen services.

d) The effects of relevant partial and total system failures including, if required, cautious assessment of the unprotected rotorcraft. This shall include an assessment of the effects of load shedding following an electrical generation system failure if relevant to continued operation in icing conditions. Note that basic electrical ice protection systems are considered to be essential services in terms of the electrical system power capacity (Chapter 711 para 2.8) if the functioning of the Ice Protection system, is a prerequisite to the performance of the Operational Role of the Rotorcraft (Chapter 706 para 2.7.2 iii).

e) The effects of icing on the operation of any sensors critical to the operational role of the Rotorcraft, eg Pitot-static heads/ports, low airspeed sensing system, etc.

f) Assessment of mechanical side effects arising from icing, eg run-back from heated surfaces restricting sliding panels, movable controls or items such as blade stops on rotor heads.

g) Effects of the accumulation of ice/snow/slush on equipment air cooling intakes and exhausts and ancillary vents and drains such as employed on fuel systems.

h) Rates, position and significance of any ice accretion and effect on rotorcraft. It must be noted that the level of significance of any ice accretion will vary dependent upon rotorcraft and configuration and the location of accretion.

j) Operation on, from or to ice and snow covered surfaces including ground and hover taxying, rotor engagement and use of deck-lock or haul-down equipment (where appropriate).

k) An assessment of the effect on Rotorcraft performance of the operation of ice protection systems, eg net effects upon available engine power (both direct and in-direct), airframe and rotor drag effects, and the significance of any failures in terms of comparative data as derived by testing outlined in (iv) of this paragraph.

LEAFLET 1006/1

ICE PROTECTION SYSTEMS
GENERAL FLIGHT TEST REQUIREMENTS

TEST INSTRUMENTATION PARAMETERS

1. INTRODUCTION

1.1 Details of parameter ranges, accuracy, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters should be considered and recorded for the tests detailed in Chapter 1006. Other parameters may be necessary to establish the ability of any individual Ice Protection System to function and perform in the Rotorcraft environment and to enable the Rotorcraft to operate in all the climatic conditions laid down in the Rotorcraft Specification. Dependent upon rotorcraft configuration, some of the described parameters may not be valid whilst others may be derived from the parameter list.

ITEM	**PARAMETER**
	GENERAL
1	Time Base
2	Manual Event Marker
3	Crew Speech
	FLIGHT CONDITION
4	Airspeed (Each Independent System)
5	Altitude (Pressure and Radio)
6	Outside Air Temperature (OAT)
7	Rotorcraft Attitude
8	Pitch Rate
9	Roll Rate
10	Yaw Rate
11	All Up Weight
	FLIGHT CONTROLS
12	Stick Positions
13	Yaw Control Position
14	Collective Pitch Control Position
15	Control Servo Positions
16	Control Surface Settings
17	Control Surface Positions
18	Control Input Forces
19	Rotor Blade Pitch, Flap and Lag Angles

ITEM	PARAMETER
	PERFORMANCE PARAMETERS
20	Engine and APU Parameters
21	Fuel Flows
22	Fuel Temperatures/Pressures
23	Rotor Speeds
24	Engine Torques
25	Rotor Drive Shaft Torques
26	Delta Torques
	LOADS (Applicable to all rotors fitted)
27	Rotating Control Loads
28	Rotor Head Loads (Flap and Lag)
29	Rotor Blade Loads (Flap, Lag and Torsion)
30	Loads in Flight Critical Static or Dynamic Components
	SYSTEMS
31	Pressures, Temperatures and Flows as Appropriate
32	Electrical Loads
	TEST CONDITION
33	Ice Severity (Liquid Water Content)
34	Vernier Ice Accretion
35	Water Droplet Size
36	Snow Severity (Visibility or Liquid Water Content) and Type
	ICE PROTECTION SYSTEMS
37	Temperatures, Pressures and Flows as appropriate
38	Photographic and/or Video Installations as appropriate

Note: In consideration of the above parameters, sufficient instrumentation should be provided to determine whether there is any detriment to the performance of the Environmental Control System(s), the Electrical System(s), the Engine and Transmission Cooling System(s), etc. caused by ice/snow blockage of air inlets and outlets.

2. TELEMETRY

2.1 The tests of Chapter 1006 are considered to be of a particularly hazardous nature, therefore telemetry may prove advantageous in some instances. Where telemetry is not available for reasons of logistics or whatever, then an extensive provision of real-time on-board monitoring should be made available directly to the pilot and/or to the flight test crew.

2.2 Details of parameter ranges, etc, are given in Leaflet 1000/1, Table 2. Critical parameters must be visually displayed at all times and loss or failure of a parameter may thus result in the abort of tests. If telemetry is necessary, all appropriate parameters required for telemetric monitoring should be defined and agreed with the Rotorcraft Project Director in advance of the trials.

LEAFLET 1006/2

ICE PROTECTION SYSTEMS
GENERAL FLIGHT TEST REQUIREMENTS

FLIGHT SAFETY

1. It is acknowledged that icing is very difficult to forecast accurately and that its severity is generally unknown prior to an actual encounter. Further to this, icing is frequently encountered during the deteriorating phase of a weather pattern. It is therefore important that when icing flights are undertaken that there is adequate clear air below the cloud layer to ensure an area of retreat where further ice build-up will be inhibited, and if necessary a safe precautionary landing can be an available option. High rates of descent from cloud in icing conditions should be used with caution as when entering air which is at positive temperature there will be a possibility of damage to rotors and engines from ice shedding from the airframe and the ability to recover from an auto rotative descent with iced rotors may be unproven. For some testing artificial icing produced by spray tankers may provide a safer and more controllable test environment.

2. Initial penetration into icing with an untried rotorcraft and/or system should only be undertaken in the following conditions:

a) Clear air for 1000 vertical feet (300 metres) below the cloud layer.

b) Horizontal visibility for one nautical mile (2 km) below the cloud layer.

c) Open terrain with low population density and minimal obstruction.

d) Predefined torque, stress, load and vibration limits to be available for initial penetration.

e) If available, air to ground telemetry of flight safety data to minimise work load on pilot/crew.

f) Predefined airframe accretion and exposure limits before evacuating the condition.

g) Predefined aircraft system critical parameter limits to be available prior to all encounters and throughout the test programme.

3. Once an understanding of the rotorcraft’s behaviour in icing conditions is acquitted some of the criteria in para 2 above may be relaxed following discussion and agreement with the Rotorcraft Project Director. An example might be the relaxation of Para 2 (i) to allow IMC recovery in non-icing air.

4. It should be noted that, with or without rotor protection, asymmetric ice shedding can result in severe vibration and or damaging stresses/loads, with the probability of vibration increasing as the temperature decreases. Experience would suggest that caution should be exercised when the conditions are such that the probability of occurrence of any of the following is high: rapid torque rise, natural cyclic shedding, asymmetric shedding, airframe accretion resulting from wet snow versus dry snow, run-back ice resulting from melt on

heated areas, cloud water droplet size etc. To some extent the ice shedding characteristics will vary from one rotorcraft to another, this feature alone calls for caution during the exploratory phase.

5. Full cold weather survival equipment should be carried in the rotorcraft and the crew should carry personal survival equipment together with current cold weather survival and local working instructions. Aircrew will normally carry parachutes when operating in icing conditions above 3000 ft, along with life jacket and dinghy if operating over water. Consideration should be given to the most suitable survival equipment assembly for the likely hazards on each particular trial.

6. Further advisory material on the operation and trials of rotorcraft in icing conditions can be found in Reference 1.

REFERENCES

No.	Title etc.
1	CAA Paper 96009, Advisory Material for Helicopter Limited Icing Clearances, Civil Aviation Authority, London, 1996. (Distributed by Westward Digital Ltd, 57 Windsor St, Cheltenham, GL52 2DG)

CHAPTER 1007

CONDITIONING SYSTEMS

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the Conditioning Systems fitted to the Rotorcraft provide conditioning within defined limits through the environmental range within which the Rotorcraft is required to operate, and are satisfactory in the following respects:

(i) Cabin/cockpit conditioning.
(ii) Equipment bay conditioning and equipment cooling.
(iii) Air ventilated suit supply (AVS) (where applicable).
(iv) Transparency de-misting.
(v) Windscreen rain removal.
(vi) Other specified functions, as applicable.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 101, 703, 709, 710, 715, 725, 1006 and 1014.

3 APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new Rotorcraft cabin and component conditioning systems and to all systems where modifications have been made which are likely to affect the results of the tests unless otherwise stated.

3.2 The need for further testing shall also be considered where modifications are made to other systems or components of the Rotorcraft which interface with and may have an effect on the performance of the conditioning systems. Typically these would be changes to Avionic or Electrical equipment which involve greater heat dissipation and/or restrictions to airflow or structural changes which affect airflow distribution and venting.

4 INSTRUMENTATION

4.1 The recommendations for instrumentation for the trials of Rotorcraft Conditioning Systems and their controls are contained in Leaflet 1007/1. (Refer also to Chapters 900 and 1000, together with the appropriate Leaflets). Many of the parameters required for the tests of this Chapter are common with the requirements of other Chapters of Part 10.

4.2 Instruments which are fitted to monitor the status of Engines and Auxiliary Power Systems shall be serviceable and calibrated. Where the standard instrumentation is not adequate for this purpose then special test instrumentation shall be provided.

5 LOADING

5.1 The tests may be made at the convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification or, if not so defined, as envisaged by the Service user, taking into account the limitations imposed by the carriage of external stores or other loads, i.e. maximum normal acceleration, attitude, speed, etc.

5.2 On rotorcraft, powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial (e.g., the effect of rocket/missile launch or gun firing on engine behaviour).

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions shall be representative of the Service Role of the Rotorcraft and shall cover the whole of the operational flight envelope and the climatic/environmental conditions defined in the Rotorcraft Specification.

6.2 It is important that the Engine(s), Auxiliary Power Systems and other equipment/systems which have a bearing on the effectiveness of the Conditioning System(s) are fully representative of the final Service standard where their characteristics are likely to affect the results of the tests.

6.3 For new Conditioning Systems it is essential that comprehensive rig testing of a fully configured and representative system has been completed and satisfactory characteristics established prior to commencement of the tests which are the subject of this Chapter. In particular the following shall be taken into account:

(i) The evidence from preliminary component and rig tests.
(ii) The range of climatic conditions and solar radiation to be tested.
(iii) The Operational Role(s) of the Rotorcraft.
(iv) The standard of the conditioning system(s) and components and their suitability for the tests proposed.
(v) Natural frequency (rapping tests) on major components and pipe runs.

7 GROUND TESTS

7.1 Tests shall be conducted on the system(s):

(i) External air/conditioning supplies as appropriate to the Rotorcraft.

(ii) All onboard air/conditioning supplies (i.e. engine/APU compressor bleed air, etc.,) throughout the range of system operating configurations provided and for which testing on the ground is practicable.

7.2 Generally it shall be established that the system(s) can be operated in accordance with the requirements laid down in the Rotorcraft Specification and as previously demonstrated on the ground rig(s). In particular it shall be demonstrated that:

(i) Means for selection, control and monitoring of all modes of the system(s) (direct fresh air, bleed air recirculation, refrigeration cooling packs, etc.,) including coupling and decoupling in the case of external suppliers, are satisfactory.

(ii) The accessibility of all components is satisfactory with respect to the ability to replace filter elements, to drain condensate and to carry out checks to establish the state of serviceability of the system(s).

(iii) The system(s) operate satisfactorily after the Rotorcraft has been subjected to:

(a) A hot soak at the upper end of the ambient operating range of the Rotorcraft with clear skies and,

(b) A cold soak at the lower end of the ambient operating range, and that the cabin, cockpit and equipment can be maintained within the required temperatures.

(iv) Cabin, cockpit, equipment and air ventilation suit(s) can be maintained within the temperature range stated in the Rotorcraft Specification during ground operations such as:

(a) Turnround servicing

(b) Standby

(c) Taxy and hold

(v) When the worst design ambient conditions of temperature and humidity are experienced, the system(s) is within the limits stated in the Rotorcraft Specification. Moisture contamination shall be reduced as low as possible and any contamination which does occur shall not affect the performance, life or reliability of either the Conditioning Equipment or the equipment being conditioned.

(vi) The system(s) shall not be damaged by the acquisition of temperatures within the range -40°C to +90°C (or as specified) and that the system(s) are capable of immediate operation when acquired temperatures come within the range -30°C to 70°C (or as specified) regardless of the effects of thermal inertia.

7.3 Tests shall be carried out to demonstrate that the system(s) environmental and duty cycle requirements are capable of being met.

8 FLIGHT TESTS

8.1 Tests shall be carried out to establish that:

(i) Temperature, pressure, flow and distribution throughout the system is as required by the Rotorcraft Specification in relation to Rotorcraft altitude, airspeed, engine power settings, flight profile, environmental conditions and system control settings.

(ii) The stabilised temperature level and distribution within the cabin, cockpit and equipment bays can be controlled in accordance with the requirements of the Rotorcraft Specification throughout the flight envelope and climatic range under both maximum and minimum environmental temperature conditions

(iii) The performance of the Conditioning System(s) after a single engine or APU failure is in accordance with the Rotorcraft Specification. (Not applicable to single engine Rotorcraft).

(iv) The performance of the emergency ventilation and de-mist system is satisfactory when conditioning is not available, either through system or engine failure.

(v) De-misting is such that a safe approach and landing can be made after a maximum rate descent from a prolonged cold soak at altitude into a hot humid environment.

(vi) Hot air rain disposal (if fitted) produces a sufficiently clear area of the windscreen for safe operation of the Rotorcraft at low flight speeds and during takeoff and landing.

(vii) The air supply to the AVS produces an AVS performance within the requirements of the AVS Specification under all conditions of flight.

(viii) The relative humidity of the air in the cabin, cockpit and equipment bays is within the limits laid down in the Rotorcraft Specification under all conditions of flight.

8.3 Measurements shall be made to ensure that noise levels occasioned by airflow in ducts and from outlets do not exceed those laid down in the Rotorcraft Specification.

8.4 EMC testing shall be carried out on the installation to ensure compliance with specification.

LEAFLET 1007/1

CONDITIONING SYSTEMS

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are considered to be the minimum for the tests detailed in Chapter 1007. Such other parameters as are necessary to establish the ability of the Conditioning System to function and perform to specification in the Rotorcraft environment should be determined by consultation with the equipment manufacturer(s)/supplier(s) and the Official Trials Establishments and agreed with the Rotorcraft Project Director.

ITEM No	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
5	Altitude (pressure)
7	Total temperature (or ambient temperature (OAT))
8	Angle of Attack
9	Pitch altitude
10	Roll angle
11	Sideslip angle
12	Heading
44	Engine throttle position(s)
45	Engine rotational speed(s)
103	Main rotor speed
123	Air intake temperature
	Compressor delivery air temperature
	Air bleed temperature(s)
	APU air intake temperature
	APU compressor delivery air temperature
	ECS air inlet temperature(s) Cockpit, cabin and
	ECS air outlet temperature(s) equipment bays, etc.
	Avionic equipment rack air inlet temperatures
	Avionic equipment unit temperatures
	Windscreen surface temperatures
	Engine compressor delivery air pressure
	APU compressor delivery air pressure
	Engine air bleed air pressure
	APU air bleed air pressure
	Air pressure at control valve inlet
	Air pressure at control valve outlet
	Static air pressure at rack(s) inlet(s)
	Static air pressure at rack(s) outlet(s)
	Engine compressor bleed valve position
	APU compressor bleed valve position

CHAPTER 1008

ALIGHTING GEAR

1 OBJECT

1.1 The object of the tests of this chapter is to confirm compliance with the design requirements for alighting gear.

1.2 The tests are designed to cover all factors which can affect the strength, functioning and operating characteristics of the alighting gear while on the ground and in flight, and are required to provide information which can be used as a basis for limitations within which the alighting gear can be used in the service.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 301, 302, 303, 304, 305, 306 and 310.

3 APPLICABILITY

3.1 The tests described in this chapter are applicable to all new wheeled alighting gears and shall also be considered for application where modifications have been made which are likely to affect the Rotorcraft and/or system characteristics and hence the results of the tests. The extent of such testing shall be the subject of discussion and agreement with the Rotorcraft Project Director during the Project Definition phase and incorporated in the Rotorcraft Specification.

3.2 The tests shall be conducted on alighting gear which is fully representative of the final Service standard. In particular tyre and oleo pressures shall be representative.

3.3 Some rotorcraft operate using a skid gear which must be given similar consideration to wheeled gear, but assessment of ground handling wheels must be included.

3.4 Requirements for alternative alighting gear will be included in the Rotorcraft Specification together with a suitable programme of tests.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The standard of instrumentation shall be such that strength, performance and operating characteristics of the alighting gear can be determined and where possible compared with the results of mathematical modelling and ground tests.

4.1.2 Consideration shall be given to the use of telemetry for these tests especially where analogy cannot be drawn from previous ship/rotorcraft configurations.

4.1.3 For details of the instrumentation characteristics and recommended measurands refer to Leaflet 1008/1.

4.2 VISUALISATION

4.2.1 It may be desirable in some circumstances to employ high speed cine photography or high speed video recording equipment to study certain transient phenomena. eg: Tyre distortion, motion of rotorcraft relative to ships deck, wheel shimmy, etc.

5 LOADING

5.1 Test shall be conducted at a range of all-up masses and centre of gravity positions which will be representative of the various roles and store configurations which are required by the Rotorcraft Specification. This will involve the total range of loadings as detailed in Chapter 1000, para 5. In addition extremes which could arise from an inability to cross transfer fuel shall be taken into consideration.

6 GENERAL TEST CONDITIONS

6.1 Ground and flight conditions are to be representative of the service role of the Rotorcraft, in the range of climatic conditions defined in the Rotorcraft Specification. However, for any test where loss of power could lead to a forced landing, the tests shall be conducted in weather conditions such that an engine out landing may be safely made.

6.2 The tests are to be made in full on a representative sample of the rotorcraft. Tests must be conducted on systems and structures which are fully representative of the final Service standard. Where the system under test interfaces with other systems, they shall also be fully representative of the final Service standard. When modifications liable to affect the characteristics of the Rotorcraft have been incorporated, the first Rotorcraft so modified shall be tested at the most adverse conditions established from previous testing. In addition, tests at other conditions shall be made, to ascertain whether the modification is likely to produce more adverse effects under other conditions or loadings.

6.3 The following shall be considered prior to commencement of testing:

(i) the evidence from component and system rig tests to indicate the level of caution to be applied in approaching various extremes, in particular where there are possibilities of ground resonance,

(ii) the range of tyre and oleo pressures to be tested including the possibilities of non-symmetrical configurations,

(iii) the operational role of the rotorcraft, in particular the requirement to test on other than smooth, hard, level surfaces. It may be necessary to test on contoured, irregular, variously sloping surfaces, soft ground and on ships decks subject to motion in six degrees of freedom with possible adverse wind directions,

(iv) the standard of the alighting gear and its suitability for the tests proposed.

6.4 All of the tests detailed in para 7 and 8 shall be approached in a planned manner ensuring that increments of change are such as to maintain adequate control over the rotorcraft at all times. Should telemetry not be available for the critical areas then, either on-board real time monitoring shall be provided or the results from each increment studied before proceeding when tests have reached the point where handling or other characteristics begin to deteriorate. (See Part 9 and Chapter 1016).

6.5 Where appropriate, landings and take-offs shall be made from rough ground along and across the grain and on slopes both longitudinal and transverse up to the safety limit as defined by the rotorcraft's handling (These tests would be considered appropriate for the use of telemetry to monitor flight safety and to indicate handling margins) (See Chapter 908).

7 GROUND TESTS

7.1 Ground tests with rotor stationary or turning as appropriate shall include the following as relevant to the case:

(i) ground handling tests on various surfaces including where appropriate both rough and smooth, hard and soft ground and grass, rough and smooth runways, hangar floors, all in both wet and dry conditions,

(ii) taxying tests; all surfaces as (i),

(iii) brake and tyre tests including the effects of scrubbing,

(iv) steering and manoeuvring tests, all surfaces as (i) (See Chapter 302),

(v) deck handling, including the use of mechanical handling or towing equipment which may be used to hangar the rotorcraft. This will need to assess the difficulty or otherwise of disengaging any form of deck lock or acquisition equipment which may be due to alighting gear characteristics (See Chapter 1009 and refer back to Para 7.1 (i),

(vi) ground resonance testing; this can be affected by oleo pressures and damping, by tyre pressures, by main rotor lag plane damper characteristics and rotorcraft all up mass. (Note: These tests would be considered appropriate for use of telemetry and remote video monitoring and recording to ensure that catastrophically critical conditions are not entered without warning).

8 FLIGHT TESTS

8.1 The following flight tests shall be performed as appropriate to the case:

(i) shimmying tests during running take-offs, landing and during taxying. (See Leaflet 301/2).

(ii) retraction and lowering tests including normal, stand-by and emergency systems (See Chapter 306). These tests shall be carried out in increments either to the maximum speeds required by the Rotorcraft Specification or to that speed, angle of attack, angle of yaw which may be established by loading or handling limitations. (See Chapter 902),

(iii) landing gear loading during normal operation as required by the Rotorcraft Specification, including where appropriate deck landing on representative ships with varying wind strengths and directions and sea states up to the maximum considered safe. (Note: These tests would be considered appropriate for the use of telemetry to monitor flight safety).

LEAFLET 1008/1

ALIGHTING GEAR

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc, are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are proposed for the tests detailed in Chapter 1008. Other parameters may be necessary to establish the ability of the Alighting Gear to function and perform to specification in the rotorcraft environment

ITEM No.	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
4A	Ground Speed
5	Altitude (pressure)
6	Altitude (radio-altimeter)
7	Total Temperature (or Ambient Temp/OAT)
8	Angle of Attack
9	Pitch Attitude
10	Roll Angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
19	Alighting Gear Position
37	Port Brake Pressure
38	Starboard Brake Pressure
39	FCS state/mode
44	Throttle Position(s)
45	Rotational Speed(s)
69	Brake Temperature (Port and Starboard)
70	Type Temperature
71	Nosewheel Angle
100	Collective Pitch Control Position
101	Collective Pitch Trip Position
102	Collective Pitch Stick Force
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
107	Main Rotor Head Moment

ITEM No.	PARAMETER
108	Tail Rotor Head Moment
109	Main Rotor Blade Flap Bending
110	Main Rotor Blade Lag Bending
112	Tail Rotor Blade Flap Bending
113	Tail Rotor Blade Lag Bending
114	Main Rotor Blade Pitching Moments/Rotating Control Load
115	Tail Rotor Blade Pitching Moments/Rotating Control Load
116	Main Rotor Blade Pitch Angle
117	Main Rotor Blade Flap Angle
118	Main Rotor Blade Lag Angle
119	Tail Rotor Blade Pitch Angle
120	Tail Rotor Blade Flap Angle
122	Alighting Gear Strut Extension/Compression
124	Tail Rotor Drive Shaft Torque
	Hydraulic Accumulator Fluid Pressure.
	Harpoon Actuator Pressure.
	Wheel Brake(s) Actuator Pressure.
	Nose Wheel Steering Actuator Pressure.
	Alighting Gear Selector Position.
	Harpoon Selector Position.
	Nose Wheel Steering Actuator Position.
	Alighting Gear Oleo Extension(s).
	Whole Body Accelerations in Six Degrees of Freedom.
	Vertical Distance from Deck at each Alighting Gear, Leg, Normal to Rotorcraft Datums
	Bending and shear loads in alighting gear legs
	Bending and shear loads in alighting gear pintles
	Bending and shear loads in alighting gear axles
	Bending and shear loads/stresses in local attachment points of pintles, axles, retraction and steering actuators, etc.
	Loads/stresses in alighting gear retraction and steering actuators
	Bending/torsion in alighting gear leg torque restraints.
	Vertical velocity at each alighting gear leg relative to deck and normal to Rotorcraft datum (doppler radar or cameras)
	Bending/torsion in alighting gear leg torque restraints.
	Vertical velocity at each alighting gear leg relative to deck and normal to Rotorcraft datum (doppler radar or cameras).

2 TELEMETRY

2.1 The tests of Chapter 1008 are considered sufficiently hazardous to justify the use of telemetry and recommendations are made accordingly. Where for any reason for the use of telemetry proves not to be feasible then provision should be made for rapid recovery and examination of data during trials in order to minimise delays through the need to monitor safety. The following parameters should be telemetered. Details of parameter ranges, etc, are given in Leaflet 1000/1, Table 2.

ITEM No.	**PARAMETER**
	GENERAL
1	Timebase
2	Manual Event
3	Crew Speech
	FLIGHT CONDITIONS
4	Indicated Airspeed
6	Incidence
7	Pitch Attitude
8	Roll Angle
9	Sideslip Angle
10	Heading
11	Pitch Rate
12	Roll Rate
13	Yaw Rate
14	Longitudinal Acceleration
15	Lateral Acceleration
16	Normal Acceleration
	CONTROL INPUTS
17	Pitch Control Position
18	Yaw Control Position
19	Roll Control Position
100	Collective Pitch Control Position
101	Main Rotor Speed
102	Input Torque
	ENGINE (EACH)
21	Power Turbine Rotational Speed
103	Main Rotor Head Moment
104	Tail Rotor Head Moment
111	Tail Rotor Drive Shaft Torque
	Vertical velocity at each alighting gear leg if doppler radar used.

CHAPTER 1009

DECK SECURING SYSTEMS

1 OBJECT

1.1 The tests required by Chapter 1009 are to demonstrate satisfactory functioning and operation of systems installed in the Rotorcraft to facilitate operation from ships decks or platforms.

1.2 The object of the tests is to obtain data from which any limitations can be determined and operating procedures formulated. The tests shall demonstrate that stresses, loads and control margins in particular remain within limits throughout all operations recommended in the release to Service.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 303, 304, 308, and 309.

3 APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new rotorcraft, deck-securing and acquisition systems. The tests are also applicable when modifications are made to either Rotorcraft or ship's equipment which are likely to affect the test results.

3.2 The tests are part of a comprehensive design and development programme to provide safe operation of Rotorcraft from ships. Tests are applicable to a combination of ship and Rotorcraft for the expected range of parameters for that combination. Refer also to Chapter 908 para 7.5.1 (v).

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The standard of instrumentation shall be such that a continuous recording can be obtained of:

(i) flight conditions (eg IAS, Altitude, OAT, Rotorcraft attitudes, motion relative to ship, etc),

(ii) system conditions (eg Hydraulic pressures, including transients, harpoon loads, etc),

(iii) structural loading (eg Alighting gear, transmission torques, rotor moments, etc),

(iv) ship's and ship's equipment conditions (eg motion in six degrees of freedom, haul cable angles, loads etc). Some means of correlating this recorded data with that recorded on the Rotorcraft will be required.

4.1.2 For cases where trials are required to be performed by the MOD the instrumentation parameters shall be agreed with the Official Trials Establishment and approved by the Rotorcraft Project Director.

4.1.3 Consideration shall be given to the use of telemetry for these tests especially where analogy cannot be drawn from previous ship/rotorcraft configurations.

4.1.4 For details of the instrumentation parameters refer to Leaflet 1009/1.

4.2 VISUAL MONITORING EQUIPMENT

4.2.1 Visual monitoring and recording equipment shall be provided on both the Rotorcraft and on the ships deck/platform, positioned so that a detail view of the securing/hauldown equipment is possible, together with an overall view of Rotorcraft and ship/platform local structure in order to determine what hazards are present and to give an overall picture of relative motions.

5 LOADING

5.1 Tests shall be conducted at a range of all-up masses and centre of gravity positions which will be representative of the various roles and store configurations which are required by the Rotorcraft Specification(s). This will involve the total range of loadings as detailed in Chapter 1000, para 5. In addition extremes which could arise from an inability to cross transfer fuel shall be taken into consideration.

6 GENERAL TEST CONDITIONS

6.1 The tests shall be carried out on Rotorcraft and equipment to a fully representative standard and the clearance must be formulated around the specific classes of ship and platform from which the Rotorcraft is required to be operated.

6.2 These tests require test conditions similar to many of the tests of Chapters 1004 and 1008 and the ship trials of Chapter 908, and shall be planned accordingly.

6.3 Ground and flight conditions are to be representative of the Service role of the Rotorcraft. During that part of the trials which takes place at sea with designated ships or platforms the wind speeds/directions, sea states and ship's motion are to be within the range of conditions laid down in the Rotorcraft Specification. Extremes are to be approached incrementally and with caution.

6.4 The following shall be considered prior to commencement of testing:

(i) the evidence from static strength tests and preliminary ground rig tests, including component and system tests to indicate the level of caution to be applied in approaching extremes,

(ii) the standard of the securing and/or acquisition gear together with the Rotorcraft alighting gear and their suitability for the tests proposed,

(iii) the intended operational use of the securing or acquisition gear,

(iv) the Rotorcraft's unassisted slope landing limits, bearing in mind that the use of cyclic control for slope landing calls for a different technique from that required on a ship's deck or platform in the presence of varying amounts of roll, yaw, pitch, heave and sway.

7 GROUND TESTS

7.1 The tests to be carried out with the systems installed in the Rotorcraft shall include the following:

(i) static release and engagement tests of the deck securing equipment, followed where practicable by rolling platform tests. These tests shall include checks on security with the system engaged,

(ii) rotation of the Rotorcraft into and out of wind with securing system engaged, with and without rotor turning,

(iii) acquisition tests where applicable, e.g. with hauldown,

(iv) deck handling tests, including the use of deck handling or towing equipment which may be used to manoeuvre or hangar the Rotorcraft (see Chapter 1008 para 7.1.(v)), also,

(v) assessment of integrity of securing system with the Rotorcraft tied down/secured, rotors stopped, over prolonged periods with the system exposed to ship manoeuvring with wind and deck motions.

8 FLIGHT TESTS

8.1 The following flight tests shall be performed as appropriate to the case:

(i) assessment of deck markings and other cues to ensure touch down within the deck/platform engagement area and safe avoidance of any structural hazards,

(ii) release and engagement tests in winds and sea states up to the maxima laid down in the Rotorcraft Specification,

(iii) checks on security of gear with rotor turning during ship manoeuvring,

(iv) acquisition tests (where applicable) from maximum operating height and variable offsets up to the maximum permitted by the system and/or operating conditions laid down in the Rotorcraft Specification,

(v) assessment of integrity of securing system under prolonged exposure to the whole range of ship's operating conditions as laid down in the Rotorcraft Specification.

8.2 Where the Rotorcraft Specification requires, the above tests shall be repeated under night conditions.

Note: Some of the above tests would be considered appropriate for the use of telemetry to monitor flight safety and consideration shall therefore be given to combining these tests with those of Chapters 908 and 1008.

LEAFLET 1009/1

DECK SECURING SYSTEMS

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are proposed for the tests detailed in Chapter 1009. Other parameters may be necessary to establish the ability of the Deck Securing system to function and perform to specification in the rotorcraft environment.

ITEM No.	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
4A	Ground Speed
6	Altitude (radio-altimeter)
7	Total Temperature (or Ambient Temp/OAT)
8	Angle of Attack
9	Pitch Attitude
10	Roll Angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
19	Alighting Gear Position
37	Port Brake Pressure
38	Starboard Brake Pressure
39	FCS state/mode
44	Throttle Position(s)
45	Rotational Speed(s)
70	Tyre Temperature
71	Nosewheel Angle
100	Collective Pitch Control Position
101	Collective Pitch Trip Position
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
107	Main Rotor Head Moment
108	Tail Rotor Head Moment
109	Main Rotor Blade Flap Bending
110	Main Rotor Blade Lag Bending
112	Tail Rotor Blade Flap Bending
113	Tail Rotor Blade Lag Bending
114	Main Rotor Blade Pitching Moments/Rotating Control Load

ITEM No.	PARAMETER
115	Tail Rotor Blade Pitching Moments/Rotating Control Load.
116	Main Rotor Blade Pitch Angle
117	Main Rotor Blade Flap Angle
118	Main Rotor Blade Lag Angle
119	Tail Rotor Blade Pitch Angle
120	Tail Rotor Blade Flap Angle
122	Alighting Gear Strut Extension/Compression
124	Tail Rotor Drive Shaft Torque
	Hydraulic Accumulator Fluid Pressure.
	Harpoon Actuator Press.
	Nose Wheel Steering Actuator Pressure.
	Alighting Gear Selector Position.
	Harpoon Selector Position.
	Nose Wheel Steering Actuator Position.
	Alighting Gear Oleo Extension(s).
	Rotorcraft Whole Body Accelerations In Six Degrees of Freedom.
	Vertical Distance from Deck at each Alighting Gear, Leg, Normal to A/C Datums.
	Relative Vertical Velocity at each Landing Gear, Leg normal to Rotorcraft Datums (Doppler radar or cameras).
	Main Wheel Angle(s) Relative to Rotorcraft.
	Hauldown Cable Load.
	Hauldown Cable Angle(s).
	Hauldown Cable Velocity.
	Ship's motion, roll attitude.
	Ship's motion, pitch attitude.
	Ship's motion, surge.
	Ship's motion, heave.
	Ship's motion, sway.

2 TELEMETRY

2.1 The tests of Chapter 1009 are considered sufficiently hazardous to justify the use of telemetry and recommendations are made accordingly. Where for any reason the use of telemetry proves not to be feasible then provision should be made for rapid recovery and examination of data during trials in order to minimise delays through the need to monitor safety. The following parameters should be telemetered. Details of parameter ranges, etc, are given in Leaflet 1000/1 Table 2.

ITEM NO.	PARAMETER
	GENERAL
1	Timebase
2	Manual Event
3	Crew Speech
	FLIGHT CONDITIONS
4	Indicated Airspeed
6	Incidence
7	Pitch Attitude
8	Roll Angle
9	Sideslip Angle
10	Heading
11	Pitch Rate
12	Roll Rate
13	Yaw Rate
14	Longitudinal Acceleration
15	Lateral Acceleration
16	Normal Acceleration
	CONTROL INPUTS
17	Pitch Control Position
18	Yaw Control Position
19	Roll Control position
100	Collective Pitch Control Position
101	Main Rotor Speed
102	Input Torque
	ENGINE (EACH)
21	Power Turbine Rotational Speed
	STRESS MONITOR
103	Main Rotor Head Moment
104	Tail Rotor Head Moment
	Hauldown Cable Load

CHAPTER 1010

POWERED FLYING CONTROLS

1 INTRODUCTION

1.1 The primary object of the tests of this Chapter is to confirm that the Rotorcraft is safe in the event of malfunction of the powered flying control system. The tests shall also provide data to the stress engineer for the assessment of the effects on the fatigue life of the airframe, controls, rotor(s) and transmission of all normal, emergency and failure cases of the Powered Flying Control System (PFCS), Automatic Flying Control System (AFCS) and the stability Augmentation System (SAS). Limited consideration is given to the possible effects of the introduction of active control technology to Rotorcraft.

1.2 The tests listed in this Chapter include the effects of AFCS and SAS and malfunctions of such systems, and are additional to the tests required by Part 9 and in particular Chapter 909.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 107, 203, 204, 207, 603, 604, 900 and Leaflet 900/1.

3 APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new Rotorcraft, powered flying control systems, stability augmentation systems and active flying control systems and to all systems where modifications have been made which are likely to affect the results of the tests unless otherwise stated.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The standard of instrumentation shall be such that the functional characteristics of the powered flying control system can be recorded and analysed. Control and transmission loads and stresses, vibration, electrical signalling and system/component temperatures shall be included where appropriate. Provision shall also be made for the measurement of structural loads and responses which may result from control system inputs.

4.1.2 The recommendations for the instrumentation for the trials of powered control systems are contained in Leaflet 1010/1.

4.1.3 The cockpit instruments and their standard of calibration shall be such that the crew can monitor accurately all the significant flight parameters. Special test instrumentation shall be installed where the standard cockpit instruments are considered inadequate for this purpose.

4.1.4 Some of the tests of this Chapter are likely to result in significant hazard to the Rotorcraft so that the provision of telemetry shall be considered. Where for any reason this should be found impracticable then real time monitoring facilities shall be provided for the crew and data analysis shall take place concurrently with the tests.

4.2 TEST EQUIPMENT

4.2.1 A hazard assessment shall be carried out which shall be used to guide the selection of the malfunctions which should be flight tested. Such a programme shall be agreed with the Rotorcraft Project Director. On the basis of the malfunctions to be flight tested test equipment shall be designed, if necessary, which is capable of progressively testing the severity of the malfunction. It may also be necessary to add additional selectable valves to simulate ballistic damage and other failures which the Rotorcraft has been designed to survive. All such test equipment shall have the facility for immediate recovery of the System(s) to normal status and shall have provisions for indication of test unit and system status to the crew.

Note: The equipment for the tests described above shall have calibrated controls capable of producing repeatable signals and shall have in addition outputs capable of being simultaneously recorded.

4.2.2 Consideration shall be given to the possible need to change the system characteristics (gain, time-constants, gearing etc) quickly between tests where this might lead to savings in elapsed time.

5 LOADING

5.1 The required loadings consist of the full range as specified in paragraph 5 Chapter 1000.

6 GENERAL TEST CONDITIONS

6.1 The tests shall be carried out on Rotorcraft and systems which are fully representative of the final Service standard, and where the systems interface with other systems, these shall also be to a fully representative standard.

6.2 Some of the tests of this Chapter could result in significant hazard to the Rotorcraft and its crew if they are undertaken in an uncontrolled manner. No failure testing should take place in flight unless the consequences have been studied by calculation, simulation and other ground testing. Any simulation used should preferably be of the "pilot in the loop" type. Flight testing should be progressive in nature and constrained such that any hazardous effect can be estimated, allowing for the total time exposed to the risk, to be no more than extremely remote.

6.3 These tests require test conditions and test equipment similar to many of the tests of Chapters 909 and 1015 and shall be planned accordingly.

6.4 The following shall be considered prior testing:-

(i) evidence from ground rig testing in accordance with Chapter 704.

(ii) evidence from fatigue testing in accordance with Chapter 704.

(iii) evidence from mathematical modelling studies of the power control system and AFCS/SAS.

(iv) information on system characteristics and performance from simulator and/or 'hot rig' including preliminary EMC testing.

(v) evidence from static ground tests on fully configured systems in a representative airframe.

(vi) a Hazard Assessment including Power Supply Generation and Distribution, AFCS and Hydraulic System.

(vii) the standard of the powered flying control system and the AFCS/SAS and their suitability for the tests.

(viii) the status of the Active Control System, where relevant, and it's software.

7 GROUND TESTS

7.1 Tests shall be conducted on the system(s) utilising:-

(i) external hydraulic and electrical supplies as appropriate to the Rotorcraft.

(ii) all on-board power sources, both hydraulic and electrical, as appropriate, ie main supplies (primary, secondary if so designated) and emergency supplies (where applicable) throughout the range of system configurations provided and for which testing on the ground is practicable.

7.2 Generally it shall be established that the systems can be operated in accordance with the requirements laid down in the Rotorcraft Specification and as previously demonstrated on the ground rigs and the simulator. In particular it shall be demonstrated that:

(i) means for selection control and monitoring of the systems is satisfactory.

(ii) the accessibility of all components is satisfactory with respect to adjustment and checks necessary to establish the serviceability of the systems.

(iii) the systems and emergency selection of alternative systems, whether automatic or manual operate satisfactorily after cold and hot soaks of the Rotorcraft in the inoperative condition, at the extremes of the environmental range as specified in the Rotorcraft Specification.

(iv) operation of the power control system and any other equipment or system powered from the same hydraulic supply(ies), in the manner which results in the greatest flow demands on the hydraulic system(s), does not reveal any limitations which are likely to inhibit control responses or ranges of movement.

(v) each actuator, servo, etc operates satisfactorily when initiated and only then and that there is no detrimental interaction between actuators. The actuators shall operate smoothly and without hesitation or instability over their full operating ranges.

7.3 Tests shall be carried out to demonstrate that such failures as can readily be represented result in correct warning indication, automatic changeover to alternative power sources and continuing satisfactory operation of systems (or specified parts of system) in accordance with specification requirements.

8 FLIGHT TESTS

8.1 Satisfactory systems operation shall be demonstrated over the full flight envelope and manoeuvre range of the Rotorcraft. Generally the objective is to confirm the results of analysis, rig and ground testing and to explore operating regimes and duty cycles that cannot readily be produced on the ground. For normal operation of controls it is of particular importance that the flight envelope limits of aerodynamic and inertia forces, including zero and negative "g" (where applicable) should be experienced. It should also be shown that operation of emergency systems will also be satisfactory in these conditions but this may be achieved be a combination of analysis, ground testing and flight testing in more moderate conditions than for the systems in normal operation.

8.2 Vibration measurements shall be made through the systems [whether mechanically impressed or system excited, and any evidence of hydraulic or other system instability shall be investigated.

8.3 Failure simulations shall be explored to the extent that it is considered acceptable and necessary in flight tests.

8.4 Checks shall be made to ensure that there are adequate and correct indication of system(s) or equipment failure or malfunction and that reversionary or emergency functions are available as required.

8.5 Throughout the testing of the powered control systems, the automatic flight control system, the stability augmentation system and the active control system there shall be continuous recording of control loads and stresses, rotor blade and hub loads and stresses and transmission torques and torsional oscillations. Some of the loads, etc at the extremes of the flight envelope may be high enough to cause substantial fatigue damage if maintained for any appreciable length of time, it is therefore essential that careful monitoring of loads and stresses is carried out throughout the testing in these regimes of flight and telemetry shall be provisioned to this end. Where for any reason telemetry cannot be provisioned then the fringes of the flight envelope shall be explored cautiously with the aid of an on-board monitoring system. Data analysis shall be concurrent with the testing to ensure minimum hazard to the Rotorcraft. Where any doubt occurs as to the propriety of continuing with testing then a new certificate for Flight testing shall be required in accordance with DEF STAN 05-123 Chapter 202.

LEAFLET 1010/1

POWERED FLYING CONTROL SYSTEMS

TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of ranges, accuracies, resolutions etc are given in Leaflet 1000/1 Table 1.

1.2 The following parameters should be considered for recording for the tests detailed in Chapter 1010.

Other parameters may be necessary to establish the ability of the Powered Flying Control System(s) to function and perform in the Rotorcraft environment and to enable the Rotorcraft to operate in all the flight regimes laid down in the Rotorcraft Specification.

ITEM No.	PARAMETER
	GENERAL
1	Time Base
2	Manual Event Marker
3	Crew Speech
	FLIGHT CONDITIONS
4	Indicated Airspeed
4A	Ground Speed
5	Altitude (pressure)
6	Altitude (radio altimeter)
7	Total Temperature (or Ambient Temp/OAT)
8	Angle of Attack
9	Pitch Attitude
10	Roll Attitude
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
	AIRFRAME CONFIGURATION AND STATE
19	Flap Setting
20	Landing Gear Position

ITEM No.	**PARAMETER**
	CONTROL INPUTS
25	Stick Position (Pitch)
26	Stick Position (Roll)
27	Yaw Control Pedal Position
28	Stick Force (Pitch)
29	Stick Force (Roll)
30	Yaw Control Pedal Force
31	Pitch Trim Position
32	Roll Trim Position
33	Yaw Trim Position
34	Elevator Positions
35	Aileron Positions
36	Yaw Control Positions
	SIGNAL SENSORS
39	FCS State/Mode
40	ILS deviation (elevation and azimuth)
41	Flight director demand (elevation and azimuth)
	ENGINE (EACH)
44	Throttle Position(s)
45	Rotational Speed(s)
100	Collective Pitch Control Position
101	Collective Pitch Trim Position
102	Collective Pitch Stick Force
103	Main Rotor Speed
104	Input Drive Torque(s)
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
107	Main Rotor Head Moment
108	Tail Rotor Head Moment
109	Main Rotor Blade Flap Bending
110	Main Rotor Blade Lag Bending
111	Main Rotor Blade Torsion
112	Tail Rotor Blade Flap Bending
113	Tail Rotor Blade Lag Bending
114	Main Rotor Blade pitching moments/Rotating control loads.
115	Tail rotor blade pitching moments/Rotating control loads.
116	Main rotor blade pitch angle
117	Main rotor blade flap angle
118	Tail rotor blade flap angle
119	Tail rotor blade pitch angle
120	Tail flap angle
124	Tail rotor drive shaft torque and torsional oscillation

ITEM No.	PARAMETER
	ENGINE (EACH)
125	Fore and aft (main) servo pos'n
126	Lateral (main) servo pos'n
127	Collective (main) servo pos'n
128	Yaw servo pos'n
129	Pitch series actuator(s) pos'n(s)
130	Roll series actuator(s) pos'n(s)
131	Collective series actuator(s) pos'n(s)
132	Yaw series actuator(s) pos'n(s)
	Pitch parallel actuator pos'n
	Roll parallel actuator pos'n
	Collective parallel actuator pos'n
	Yaw parallel actuator pos'n
	Sonar cable F&A angle
	Sonar cable lateral cable angle
	Sonar cable length
	Underslung load cable angles
	Hover hold signal
	Height hold signal
	Heading deviation/error signal
	Main servo(s) input signal(s)/pos'n(s)
	Doppler longitudinal velocity
	Doppler lateral velocity
	Doppler vertical velocity
	High sensitivity radio altimeter
	Hydraulic system(s) temperatures and pressures
	Stationary Flying control loads

Provision should be made for passive interfaces to all Digital Data Highways (MIL STD 1553 and ARINC 429 for instance) in order to acquire data from within the PCS, AFCS, SAS, and ACS.

2 TELEMETRY

2.1 The tests of Chapter 1010 are considered to be sufficiently hazardous for telemetry to be a requirement, as is a comprehensive on-board monitoring system.

2.2 The following telemetry parameters are considered to be the minimum for the tests of Chapter 1010. Details of parameter ranges, etc, are given in Leaflet 1000/1, Table 2.

ITEM No.	PARAMETER
	GENERAL
1	Time Base
2	Manual Event
3	Crew Speech
	FLIGHT CONDITIONS
4	Indicated Airspeed
5	Pressure Altitude
6	Incidence
7	Pitch Attitude
8	Roll Attitude
9	Sideslip Angle
10	Heading
11	Pitch rate
12	Roll Rate
13	Yaw Rate
14	Longitudinal Acceleration
15	Lateral Acceleration
16	Normal Acceleration
	CONTROL INPUTS
17	Pitch Control Position
18	Yaw Control Position
19	Roll Control Position
20	Throttle position(s)
21	Power Turbine Rotational Speed
100	Collective Pitch Control Position
101	Main Rotor Speed
	STRESS MONITOR
102	Input Drive Torque(s)
103	Main Rotor Head Moment
104	Tail Rotor Head Moment
105	Main Rotor Blade Torsion
106	Tail Rotor Blade Torsion
107	Main Rotor Blade Flap Bending
108	Main Rotor Blade Lag Bending
109	Main Rotor Rotating Control Load
110	Tail Rotor Rotating Control Load
111	Tail Rotor Drive Shaft Torque

CHAPTER 1011

ELECTROMAGNETIC COMPATIBILITY OF SAFETY CRITICAL SYSTEMS

1 OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the electronic/electrical systems of the rotorcraft which are safety critical operate satisfactorily and with an adequate margin of safety (as defined by the Main Contractor and agreed by Project Office on advice from EMC specialists) in:

(i) The electromagnetic environment generated by the rotorcraft (systems of the rotorcraft are self compatible), and

(ii) The external electromagnetic environment corresponding to the operational requirement.

2 RELEVANT DESIGN REQUIREMENTS

2.1 EMC Control Document/Portfolio/Design Guide written by the main contractor for the specific project as agreed by DAES, (for non-armament aspects) or DA Arm (for air armament and air armament control aspects) Project Office and the test authority (usually A&AEE/Engineering Division/Armament Division); external environment as defined in NWS 6 or as agreed in the Contract; EMC qualification test requirements for the installed equipment to RAE Tech Memo FS(F) 510 and Ordnance Board Proceeding (41273) "Principles of Design and Use for Electrical Circuits Incorporating Explosive Components". The documents must include the agreed statement defining the limits of satisfactory operation, criteria against which satisfactory operation shall be judged, methods of monitoring and limits of responsibility of Contractor and Project Office. The documents must also contain agreed statements concerning the levels of electrical transients on power, signal and/or control leads (see also Chapter 1003).

3 APPLICABILITY

3.1 The rotorcraft tests described in this chapter are applicable to all new electronic/electrical safety critical systems of rotorcraft and all such systems where modifications have been made which are likely to affect the results of the tests unless otherwise stated.

3.2 A safety critical system is defined as a system (or one of a collection of systems) of the rotorcraft in which a disturbance (or combination of disturbances) could result in a direct hazard to the rotorcraft, aircrew, people or property on the ground.

4 EQUIPMENT

4.1 The basic equipment required for these tests is:

(i) Source(s) of electromagnetic fields.

(ii) Antennae/receivers to measure strength of electromagnetic fields .

(iii) Current/voltage probes for injecting and measuring interference induced during trials and fibre optic instrumentation for signal transmission.

(iv) Such instrumentation as is necessary to indicate system malfunction/ degradation of performance.

Note: The standard of instrumentation and installed test equipment in the test rotorcraft must be such that it does not affect the rotorcraft's electromagnetic compatibility.

5 GENERAL TEST CONDITIONS AND REQUIREMENTS

5.1 All electrical/electronic systems are to be fully representative of the production rotorcraft and working. No additional wiring to rotorcraft or to stores is to be present except that required for the trial. (See also Note, para 4).

5.2 Soundly based information on the likely EMC characteristics of the systems under test is an important pre-requisite to effective testing of the rotorcraft and this shall be obtained by the main contractor. The information shall be acquired during Qualification and Rig Testing or by special tests agreed with the Test Authority. The information shall be presented in the form of a malfunction signature (in a form in which measurements on rotorcraft can be related) showing the minimum voltages/currents/field strengths required to prevent satisfactory operation or to cause a malfunction as a function of frequencies as agreed between the Contractor and the Test Authority. Particular susceptibilities at certain frequencies are to be clearly identified together with levels at which malfunction/unacceptable degradation of performance occurs. Such information should be available in the EMC portfolio compiled through pursuance of the EMC Control Plan for the systems under test. In order to ensure the relevance of the data, care shall be taken to ensure that the tests used to generate the malfunction signature are made under conditions which are as representative of the final installation as is practicable to achieve. These conditions shall be agreed between the Main Contractor, Rotorcraft Project Director and the Test Authority.

6 TEST DETAILS

6.1 Each system to be tested shall be exercised through or in all (or at least most) susceptible modes while the other rotorcraft systems are brought into operation and any effect of operation, including effects of electrical transients, noted and where practical and sensible the emission from intentional transmitters e.g. VHF radio should be enhanced in power) as agreed in the EMC Portfolio. Subsequently, the systems are to be exercised while the rotorcraft is illuminated from a source (or range of sources) representative of the required external electromagnetic environment. If there are limitations to the extent to which the required external environment can be generated, for example with respect to frequency, power or orientation in relation to the axes of the rotorcraft then additional tests should be considered using substitutional techniques such as direct voltage or current injection.

6.2 Induced interference is to be continuously monitored during the trial together with performance of the systems under test. Serious loss or degradation of performance is likely to take the form of either an unwanted output or as a failure to respond to a wanted signal. Because the former is easier to detect and because the mechanisms generating the two forms of degradation are similar, the interferences shall be modulated at a frequency which is within the operating bandwidth of the system under examination.

6.3 Where the test is completed without system malfunction or serious degradation of performance occurring, the margin of safety is to be established by comparison of the induced interference recorded from the test with the malfunction signature established previously (see para 5.2). In the case of armament systems, the margin of safety to be applied shall be in accordance with Ordnance Board Proceeding (41273).

CHAPTER 1012

ESCAPE SYSTEMS AND FLOTATION GEAR
GENERAL FLIGHT TEST REQUIREMENTS

1. OBJECT

1.1 The object of the tests of this chapter is to demonstrate that the Escape Systems and Flotation Gear installed in the Rotorcraft function and perform in accordance with specification and are suitable for service use. In particular to:-

a) Demonstrate the effective functioning of the escape equipment on the ground, in flight, floating on the water and submerged.

b) Establish satisfactory procedures for the effective use of escape equipment in those same four regimes.

c) Demonstrate the satisfactory operation of the flotation gear in flight (where applicable) and in the water by means of full scale simulation.

2. RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 102, 307, 703, 718 and 719.

2.2 DEF STAN 00-970 Volume 1 Chapters 102 and 106, in lieu of Design requirements specific to Rotorcraft.

3. APPLICABILITY

3.1 The tests described in this Chapter are applicable to all new canopies, doors, hatches and flotation gear, etc and shall also be considered for application where modifications have been made which are likely to affect the Rotorcraft and/or system characteristics and hence the results of the tests. The extent of such testing shall be the subject of discussion and agreement with the Rotorcraft Project Director, during the Project Definition Phase and be incorporated in the Rotorcraft Specification.

4. EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The standard of instrumentation, including telemetry where appropriate, shall be such that the functioning and performance of the escape system and flotation gear can be monitored.

4.1.2 The recommendations for instrumentation for the trials of Rotorcraft escape systems and flotation gear are contained in Leaflet 1012/1.

4.2 VISUAL MONITORING EQUIPMENT

4.2.1 Video and/or high speed cine cameras shall be fitted to the test Rotorcraft, on the ground and on chase aircraft so that the functioning and performance of the escape equipment and flotation gear can be monitored and recorded.

4.3 TEST EQUIPMENT

4.3.1 Crew ejection is not a feature generally associated with Rotorcraft but against the contingency that future Rotorcraft may incorporate such systems (eg tilt rotor/wing rotorcraft) some allowance must be made for testing such provisions. The test to determine the functioning and performance of the ejection seat(s) require the provision of a special test Rotorcraft incorporating modifications to permit the ejection of the seat(s) for test purposes.

4.3.2 Maximum possible use shall be made of available facilities such as the Official Trials Establishment blower tunnel to establish system performance characteristics prior to flight testing.

5. LOADING

5.1 The tests may be made at any convenient loading and centre-of-gravity position, provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification, taking into account limitations imposed by the carriage of external stores or other loads (ie. maximum normal acceleration, attitude, speed etc.).

5.2 On rotorcraft powered by a single engine, external stores or other loads must not be carried on any test which may result in, or require the engine to be shut down, unless the store or load is specifically required for the trial.

6. GENERAL TEST CONDITIONS

6.1 Ground and flight conditions shall be representative of the Service Role of the Rotorcraft and shall cover the whole of the operational flight envelope and the climatic/environmental conditions defined in the Rotorcraft Specification.

6.2 For new escape systems and flotation gear it is essential that comprehensive rig testing of fully configured and representative equipment has been completed and satisfactory characteristics established prior to commencement of the test of this Chapter. In particular the following shall be taken into account:-

a) The evidence from preliminary ground rig testing of the ability to operate MDC systems or jettison canopies, doors, hatches, etc under simulated aerodynamic and hydrodynamic loading.

b) The evidence from full scale mock-up, capable of being set in various attitudes, or rotorcraft, that access to and egress from all escape apertures is clearly marked and escape possible under all lighting and environmental conditions. These tests shall have been carried out with full normal aircrew and passenger clothing and equipment appropriate to the role of the Rotorcraft.

c) The evidence from ground rig testing of the ability of the flotation bags to inflate symmetrically (where applicable) whether fed from single or multiple gas sources at normal, maximum and minimum ambient temperatures as specified by the Rotorcraft Specification, and with faults as specified by Chapter 307 Para 2.4.

d) Natural frequency (rapping) tests on major components and pipe runs.

e) Evidence from ditching model tests and flotation model tests to establish ditching procedures and stability in adverse weather and sea states. Flotation model tests should include faults as specified in Chapter 307 Para 2.4.

f) Ejection seat tests from test rigs, including blower tunnel test, including the ability to carry out single ejections (where applicable) without impairing the ability of other crew member(s) to escape.

g) Simulator data (if available) to determine what limitations (if any) need to be applied for service use, of in flight inflations of the flotation gear (where applicable) over the range of speeds defined in the Rotorcraft Specification. Symmetrical and Asymmetrical inflations need to be considered.

7. GROUND TESTS

7.1 Tests shall be conducted on the systems as follows:-

a) Full scale escape tests using a representative fuselage in a water tank facility. Opportunity should be taken to undertake live underwater escape demonstration and if applicable would include any underwater ejection system if standard to type.

b) Inflation tests of flotation gear/bags to demonstrate free inflation characteristics uninhibited by bay covers or local structure. Where weapons or other external stores are capable of being carried by the Rotorcraft tests shall be carried out to ensure that in the event that such stores have not been jettisoned prior to ditching that damage is not caused to flotation gear to the extent that it is rendered inoperative.

c) EMC testing shall be carried out to demonstrate that the flotation gear is not inadvertently deployed in flight or on the ground/deck by operation of aeroplane/rotorcraft, ground or ships equipment.

8. FLIGHT TESTS

8.1 Satisfactory systems operation shall be demonstrated over the flight envelope required by the Rotorcraft Specification. Generally the objective is to confirm the results of rig and ground testing and to explore operating regimes which cannot readily be produced on the ground. Use of simulator data may be used in lieu of in flight inflation testing. Particular tests if required shall be carried out as follows:-

a) In flight inflation of flotation gear (where applicable) over the range of speeds defined in the Rotorcraft Specification to determine what limitations (if any) need to be applied for Service use.

b) Tests to determine the effects of inadvertent in-flight inflation (symmetrically and either side asymmetrically) of flotation gear at speeds up to the Rotorcraft Specification operational maximum. These tests shall proceed with caution making small incremental increases in speed at the same time monitoring control, rotor, airframe and bag attachment loads. Visual monitoring/recording facilities shall be provided so that the motions of inflating and inflated bags may be determined, together with an assessment of the risk of damage to the bags through abrasion and/or impact with local structure, external stores, etc. The use of telemetry for these last tests would be appropriate since it could enhance flight safety and reduce the probability of catastrophic damage to bags and attachments.

8.2 Where the type of ejection system fitted necessitate flight tests, the tests will be defined in the Rotorcraft Specification.

LEAFLET 1012/1

ESCAPE SYSTEMS AND FLOTATION GEAR GENERAL FLIGHT TEST REQUIREMENTS

TEST INSTRUMENTATION PARAMETERS

1. INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc, are given in Leaflet 1000/1.

1.2 The following parameters are proposed for the tests detailed in Chapter 1012. Other parameters may be necessary to establish the ability of the Escape Systems and Flotation Gear to function and perform in the Rotorcraft environment. Note that Item No. below relates to Leaflet 1000/1.

ITEM NO.	PARAMETER
1	Time Base
2	Manual Event Marker
3	Crew Speech
4	Indicated Airspeed
4A	Ground Speed
5	Altitude (pressure)
6	Altitude (radio-altimeter)
7	Total Temperature (or ambient Temp/OAT)
8	Angle of Attack
9	Pitch Angle
10	Roll Angle
11	Sideslip Angle
12	Heading
13	Pitch Rate
14	Roll Rate
15	Yaw Rate
16	Longitudinal Acceleration
17	Lateral Acceleration
18	Normal Acceleration
19	Alighting Gear Position
39	FCS state/mode
71	Nosewheel Angle
100	Collective Pitch Control Position
101	Collective Pitch Trip Position
102	Collective Pitch Stick Force
103	Main Rotor Speed
104	Input Drive Torque
105	Main Rotor Collective Pitch
106	Tail Rotor Pitch
107	Main Rotor Head Moment

ITEM NO.	PARAMETER
108	Tail Rotor Head Moment
109	Main Rotor Blade Flap Bending
110	Main Rotor Blade Lag Bending
112	Tail Rotor Blade Flap Bending
113	Tail Rotor Blade Lag Bending
114	Main Rotor Blade Pitching Moments/Rotating Control Load
115	Tail Rotor Blade Pitching Moments/Rotating Control Load
116	Main Rotor Blade Pitch Angle
117	Main Rotor Blade Flap Angle
118	Main Rotor Blade Lag Angle
119	Tail Rotor Blade Pitch Angle
120	Tail Rotor Blade Flap Angle
122	Alighting Gear Strut Extension/Compression
124	Tail Rotor Drive Shaft Torque

2. TELEMETRY

2.1 Some of the tests of Chapter 1012 are considered sufficiently hazardous to justify the use of telemetry and recommendations are made accordingly. Where for any reason the use of telemetry proves not to be feasible then provision should be made for rapid recovery and examination of data during trials in order to minimise delays through the need to monitor safety. The following parameters should be considered for telemetry. Details of parameter ranges, etc, are given in Leaflet 1000/1, Table 2. Note that Item No. below relates to Leaflet 1000/1.

ITEM NO.	PARAMETER
	<u>GENERAL</u>
1	Timebase
2	Manual Event
3	Crew Speech
	<u>FLIGHT CONDITIONS</u>
4	Indicated Airspeed
6	Incidence
7	Pitch Attitude
8	Roll Angle
9	Sideslip Angle
10	Heading
11	Pitch Rate
12	Roll rate
13	Yaw Rate

ITEM NO.	PARAMETER
14	Longitudinal Acceleration
15	Lateral Acceleration
16	Normal Acceleration
17	Pitch Control Position
18	Yaw Control Position
19	Roll Control Position
100	Collective Pitch Control Position
101	Main Rotor Speed
102	Input Torque
	STRESS MONITOR
103	Main Rotor Head Moment
104	Tail Rotor Head Moment

CHAPTER 1013

WATERPROOFING

1 OBJECT

1.1 The object of the tests of this Chapter is to ensure the satisfactory watertightness of the cockpit, cabin and other parts of the Rotorcraft containing electronic, electric, or other equipment, electrical wiring, controls, and control runs, the efficiency of which may be impaired by water, and also to check the satisfactory drainage ability of internal bays and compartments where water may collect.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 101 and 407.

3 APPLICABILITY

3.1 Ground and flight tests shall be made in full on a development rotorcraft, and on the first of any subsequent production rotorcraft which embody alterations or modifications likely to affect watertightness.

4 GROUND TESTS

4.1 The tests shall be made on the Rotorcraft:

(i) after satisfactory completion of the cabin pressure proving tests, where appropriate,

and where practical,

(ii) before the fitment of interior trimming and upholstery,

(iii) before painting.

4.2 During the test all sliding canopies, windows, direct vision openings, inspection covers, doors and hatches shall be closed. When the rotorcraft has components which may be folded when the rotorcraft is parked, it shall be tested in both configurations.

4.3 Each part of the rotorcraft shall be subjected to a water spray of the following intensity for a period of at least twenty minutes:

Rotorcraft designed for World wide use	**Precipitation Intensity mm/hr**	**Average Droplet dia mm**	**Velocity of fall m/sec**
	100	3.0	7

During the test the direction of the spray shall be varied to simulate the effect of driven rain from any direction on the parked Rotorcraft. The spray shall also be directed upwards from ground level to simulate normal Rotorcraft taxying, take-off, and landing on wet concrete/tarmac/ships deck. Water spray shall also be directed into the undercarriage bays and fuel filler areas.

4.4 After the spray test all parts of the Rotorcraft shall be thoroughly inspected for leaks and accumulations of water. Particular attention shall be given to the sealing of access panels and doors to electrical and avionics bays where ingress and retention of water could have adverse effects on equipment performance. All external doors and access panels shall be opened for the inspection.

Note: Where there is difficulty in tracing the source of a leak the use of fluorescine dye powder technique is recommended.

4.5 When it is impossible to prevent the ingress of water into compartments, the inspection shall ensure that the drainage arrangements satisfy the requirements of Chapter 407.

5 FLIGHT TESTS

5.1 A flight test shall be made on a Rotorcraft which has satisfactorily completed the ground test. After standing in the rain the Rotorcraft shall be flown from a wet surface in rain of an intensity in which it is required to operate for a period sufficient to ensure a thorough check of watertightness being made.

5.2 During the test direct vision openings if fitted shall be opened at an appropriate time to demonstrate that there is no unacceptable ingress of water. Components such as landing lights and ram air turbines which can be extended and retracted shall be operated.

Note: The opportunity will need to be taken to fit the flight test in with other trials when suitable weather conditions prevail.

5.3 As far as is practicable during the flight all accessible compartments shall be inspected for leakage. As soon as possible after landing a full inspection as specified in para 4 shall be made. The undercarriage bays and any other areas exposed to spray thrown up from the wet surface shall also be inspected.

5.4 Where the inspection reveals the need for remedial action such action shall be followed by a repeat of the ground test on the affected parts and the need for a repeat of the flight test shall be discussed and agreed with the Rotorcraft Project Director.

CHAPTER 1014

ARMAMENT INSTALLATIONS

1 INTRODUCTION

1.1 The flight test objectives laid down in this Chapter are intended to ensure that the armament installations are safe and likely to satisfy the operational requirements for the rotorcraft concerned. Armament installations, for the purpose of this requirement, covers the whole range of ballistic, propelled or dropped offensive/defensive weapons or stores which may be carried by rotorcraft.

2 GENERAL

2.1 Flight trials are required to demonstrate that the design of the rotorcraft armament installation is sound and that the limiting environmental factors are not exceeded. A sufficient number of store releases or weapon firings must be made to achieve this objective.

2.2 Instrumentation and photographic or kinetheodolite coverage of the rotorcraft and weapon is required to record the critical parameters on which the design is based and validate any mathematical models and wind tunnel or ground trials.

3 APPLICABILITY

3.1 The coverage of this chapter shall include the following:

(i) Fixed guns.

(ii) Pintle mounted or remotely laid guns.

(iii) Bombs or depth charges.

(iv) Torpedoes

(v) Sonobuoys.

(vi) Rockets.

(vii) Guided weapons.

(viii) Mine dispensers.

(ix) Flares

(x) Chaff dispensers.

4 WEAPONS OR STORES CARRIAGE AND RELEASE

4.1 The flight trials are to provide evidence of satisfactory carriage, release and jettison of weapons thus the contractors' trials must explore the extremities of the flight envelope if there is any doubt as to flight safety. C.A., in conjunction with the rotorcraft designer is responsible for recommending the limits to which service pilots may fly. The flight trials shall cover:

(i) Carriage - Weapon carriage flights are to be flown within the required operational envelope of the rotorcraft. Sufficient in-flight data is to be obtained to validate the design calculation of dynamic, vibration and stress loads on rotorcraft weapon carrier, weapon structure and where considered applicable on rotorcraft structure and dynamic components. Any heat or contamination effects on the weapon installation and any adverse effects on rotorcraft handling or performance shall be noted.

(ii) Release - The evidence provided by the releases or firings must be sufficient to validate design calculations, mathematical models and wind tunnel tests. The tests should demonstrate beyond any reasonable doubt, that safe release can be achieved under any condition within the operational envelope with a safe margin for inadvertent release beyond these limits. There must also be clear evidence that all safeguards against premature priming before or during release are effective.

(iii) Jettison - Single store jettison tests at representative heights and speeds in level flight must be made. Multiple store jettison tests may be required. The jettison evidence must be sufficient to validate design calculations, mathematical models and wind tunnel tests on all stores, including dispensers, fuel tanks and E.C.M. or guidance pods when these are capable of jettison.

(iv) Hang up procedures shall be assessed.

5 GUN FIRING TESTS

5.1 SCOPE OF TESTS

5.1.1 The integral nature of the gun firing tests that are required on a new rotorcraft gun installation result in considerable overlap between the responsibilities of the contractor and A.&A.E.E. The safety and functioning tests that are the responsibility of the contractor are in many cases identical with tests required by A.&A.E.E. in the C.A. Release test programme. Tests of recent gun installations have been conducted as joint contractor/A.&A.E.E. trials and it is envisaged that all future gun installations will be the subject of similar, joint trials. Thus the following flight test objectives for gun firing tests are intended to ensure that the installation is safe, satisfies the operational requirement and is suitable for C.A Release.

5.2 FUNCTIONING TESTS

5.2.1 The correct functioning of all guns shall be established by firing them both symmetrically and asymmetrically at airspeeds, normal accelerations (both positive and negative) and heights representative of the range of conditions likely to be met operationally. The following features shall be examined:

(i) Gun installation air conditioning.

(ii) Gun installation thermal environment.

(iii) Evidence of engine malfunction during and as a result of firing.

(iv) Interference with the flying characteristics of the rotorcraft.

(v) Effect of vibration on the rotorcraft structure, stores and equipment, including cockpit instruments and H.U.D.'s, also compass detector units in which errors are present due to changes in permanent magnetism of structure, guns or equipment induced by vibration.

(vi) Risk of damage to the rotorcraft structure or stores for ,which carriage clearance is required, by link, case or complete rounds strikes if these are not collected.

(vii) Gun re-cocking system if fitted.

(viii) Gun purging systems.

(ix) Effectiveness of travel limiting stops on pintle mounted or remotely laid guns.

(x) E.M.C. clearance.

(xi) Sight harmonisation.

5.3 INTENSIVE GUNNERY TESTS

5.3.1 A series of tests shall be made on a production rotorcraft which has not previously undergone gun firing, to assess the following:

(i) Functioning, reliability and durability of the gun installation under conditions of intensive use.

(ii) Effect of blast and vibration on the rotorcraft structure and equipment and any store for which carriage clearance is required.

(iii) Suitability of special-to-type servicing equipment used with the installation.

(iv) Misfire procedures.

6 ROCKETS AND GUIDED WEAPONS

6.1 COPE OF TESTS

6.1.1 The comments regarding the relationship between the contractor and A.& A.E.E. for gun firing tests are equally applicable for rockets and guided weapons (para 4.1 refers).

6.2 FUNCTIONING TESTS

6.2.1 The correct functioning of all rockets or missiles shall be established by firing them both symmetrically and asymmetrically at airspeeds, normal accelerations (both positive and negative) and heights representative of the range of conditions likely to be met operationally. The following features shall be examined:

(i) Weapon installation air conditioning.

(ii) Weapon installation thermal environment.

(iii) Evidence of engine malfunction during and as a result of firing.

(iv) Interference with the flying characteristics of the rotorcraft.

(v) Damage or contamination due to efflux debris.

(vi) Effect of vibration on the rotorcraft structure, stores and equipment, including cockpit instruments and H.U.D.'s, also compass detector units in which errors are present due to changes in permanent magnetism of structure, weapons or equipment induced by vibration.

(vii) Assessment of sighting, acquisition and pre launch switching procedures associated with locking on to target and energising or pre programming weapons prior to launch.

6.3 INTENSIVE WEAPON TESTS

6.3.1 A series of tests shall be made on a production rotorcraft which has not previously undergone missile or rocket firing, to assess the following:

(i) Functioning, reliability and durability of the missile or rocket installation under conditions of intensive use.

(ii) Effect of blast and vibration on the rotorcraft structure and equipment and any store for which carriage clearance is required.

(iii) Assessment leading to operational procedures regarding post launch tracking or manoeuvres.

(iv) Suitability of special-to-type servicing equipment used with the installation.

(v) Hang up and misfire procedures.

7 INTERNAL DISPENSERS

7.1 SCOPE OF TESTS

7.1.1 These are generally concerned with fairly innocuous stores such as sonobuoys or smoke floats but may include flares or target markers. Contractors and A.&A.E.E. trials may be carried out separately. The flight tests are intended to ensure that the installation is safe and that the onboard handling of the store in the cabin environment prior to release can be carried out with safety and that the installation as a whole satisfies the operational requirement and is suitable for C.A. release.

7.2 FUNCTIONING TESTS

7.2.1 The installation shall be assessed over the whole range of rotorcraft operating conditions with representative loads of stores. The following features shall be examined:

(i) Dispenser thermal environment.

(ii) Effect of vibration oil dispenser structure and stores.

(iii) In flight loading of dispenser (where applicable).

(iv) E.M.C.

(v) Release tests should confirm that all classes of stores capable of carriage and release can be safely operated with assurance that safe release can be achieved at any condition within the flight envelope with adequate margin to cater for inadvertent release beyond these limits.

7.3 INTENSIVE DROPPING TESTS

7.3.1 A series of tests shall be made on a production rotorcraft to assess the following:

(i) Functioning, reliability and durability of the dispenser installations under conditions or intensive use.

(ii) Suitability of special-to-type servicing equipment used with the installation.

(iii) Hang up procedures.

7.4 CHAFF DISPENSERS

7.4.1 Tests with chaff dispensers can generally be covered by the tests called up in this section but testing must include rig or ground tests to ensure that chaff is dispersed well clear of the rotorcraft. The ingress of chaff to avionic bays can have serious consequences and in the cabin environment can present a health risk to the crew. Final clearance of chaff must include discharge at operational speeds.

8 AIR CARRIED MINE DISPENSERS

8.1 This section concerns mine dispensers which may be carried as an underslung load. The mechanical and handling clearance of these can be treated like any other underslung load and should meet the requirements of Chapter 1017. The following tests are required to meet the armament aspects of the assessments:

(i) E.M.C.

(ii) Drop tests over the allowable speed range with permutations of release sequences and various stores.

9 ICING CLEARANCE

9.1 Where the armament includes the air carriage of external stores and use of weapons bays with doors functioning in flight, icing and climatic trials should include such stores and fittings which will be part of the testing called up in Chapter 1006.

10 CLIMATIC CLEARANCE

10.1 Clearance of all the above systems as part of the hot or cold climatic trials shall be considered. The extent of firing, dropping or jettison trials should be agreed with the Rotorcraft Project Director and DA Arm MOD(PE).

CHAPTER 1015

STRUCTURES

1 INTRODUCTION

1.1 This Chapter states the requirements for:

(i) The measurement of loads on the rotorcraft structure, rotor system and controls on the ground an in flight with application of the results to the static strength and fatigue tests and to substantiate the declared fatigue life of vital dynamic components such as rotor blades, gearboxes, controls and transmission.

(ii) The demonstration of structural strength by flight tests up to the specified design conditions.

2 BASIC REQUIREMENTS

2.1 The scope of the load measurement programme, including the conditions to be covered, the measurements to be taken and the techniques to be employed, shall be discussed and agreed with the Rotorcraft Project Director on the advice of the Airworthiness Division, R.A.E. This shall be done at a sufficiently early stage in the design to ensure adequate planning of the test programme, both technical and financial.

2.2 An indication of the conditions and measurements to be considered is given in paras 4 and 5. The scope of these conditions and measurements should be carefully considered for each project and may be increased or decreased, according to the needs of the project concerned. The necessary instrumentation and equipment should be built into the rotorcraft from the start.

2.3 The coordination of the load measurement programme and the static and fatigue strength test programmes shall also be agreed with the Rotorcraft Project Director on the advice of the Airworthiness Division, R.A.E., and discussion continued throughout the programme.

2.4 It is important to recognise that in Rotorcraft there is a large number of dynamic components all of which can be subject to very high vibratory stresses with the result that there are generally a large number of lifed items. Many of the major components such as rotor blades are the subject of extensive fatigue testing in their own right with specially designed rig tests which include dynamic inputs. The whole procedure of static fatigue tests, vibration survey, resonance tests in flight stress measurements and intensive running shall be brought together in an integrated programme and agreed with the Rotorcraft Project Director on the advice of the Airworthiness Division, R.A.E., and discussion continued throughout the programme.

3 APPLICABILITY

3.1 The tests agreed under para 2.1 shall be carried out on an early development rotorcraft. Further tests shall be made on later development rotorcraft, as far as is considered necessary after discussion with the Rotorcraft Project Director on the advice of the Airworthiness Division R.A.E., if major structural or dynamic component alterations have been made or flying limitations have been raised.

4 TEST REQUIREMENTS

4.1 The magnitude, distribution and time history of the loads shall be determined in the agreed flight conditions and appropriate rotorcraft configurations. In selecting the flight conditions the following cases shall be considered, and also any other static or fatigue loading cases arising from the particular role of the rotorcraft.

(i) The symmetric manoeuvres of Chapter 202.

(ii) Flight in turbulence levels up to the maximum called up in the Specification.

(iii) Special flight manoeuvres such as 'nap of the earth' flying and unusual manoeuvres that are likely to be performed during both operational and training sorties.

(iv) The take off and landing cases of Part 3 including ship operation and sloped site operation.

(v) Turns autorotations, transition to and from the hover.

In all manoeuvre cases, the loads, and their distribution and time histories during the manoeuvre, shall be determined over a range of severity of the manoeuvre up to the design conditions or to any lower limit set by safety or other consideration.

4.2 The structural strength of the rotorcraft shall be demonstrated in particular manoeuvres by flights to the limits of the design conditions or to any lower limit set by safety or other consideration and agreed with the Rotorcraft Project Director. Thorough inspection of the structure shall be made after the flight and there shall be no signs of proof failure.

4.3 Stress monitoring in vital dynamic components shall be included for all parts of the programme where it is considered possible that design limits can be approached or even inadvertently exceeded, examples are:

(i) A.F.C.S. runaway tests.

(ii) Icing trials.

(iii) Engine off landing trials.

(iv) Deck landing or rejected take off tests.

(v) Type tests

4.4 The above tests where applicable may be combined with the relevant flight tests required by Part 9.

5 MEASUREMENTS

5.1 Measurements shall be made at sufficient stations to establish the loads in the relevant parts of the structure with reasonable accuracy. The number and position of the stations will depend on the project concerned, on the quantities being measured and on the practical aspect of accessibility.

5.2 AIRFRAME STRUCTURE

5.2.1 The following quantities should be considered for measurement:

(i) Shear and bending moment on the main cabin frames associated with gearbox support.

(ii) Shear and bending moment at one or more fuselage sections including tail boom break fin joint (where applicable).

(iii) Shear, bending moment and torque at the horizontal stabiliser root and fins (where applicable).

(iv) Landing gear loads.

(v) Loads on external stores.

(vi) Main, intermediate and tail gearbox attachment loads.

5.3 DYNAMIC COMPONENTS AND CONTROLS

(i) Main and tail rotor blade flap and lag bending and torsion at critical stations.

(ii) Main rotor head shaft bending.

(iii) Blade attachment arm loads.

(iv) Control pitch change lever or spider arm bending.

(v) Control pitch change rod bending.

(vi) Tail rotor shaft torsion (where applicable).

(vii) Front and rear main gearbox interlink shaft bending where applicable.

(viii) Effect of thermal stresses which may arise from cyclic heating with electro thermal de-icing systems.

(ix) Environmental measurements where applicable i.e. icing severity when high loads may be expected.

5.4 The rotorcraft will be subjected to an extensive vibration survey and the outcome of this work has a direct relevance to the tests call up in this Chapter. Vibration is the subject of Chapter 1016 and in many cases vibration and stress measurements will be required from the same sorties.

5.5 A continuous record shall be made of the relevant flight parameters necessary to define the particular manoeuvres in which the load measurements are being taken. The following quantities shall be considered for measurements.

(i) Airspeed.

(ii) Altitude.

(iii) Normal acceleration.

(iv) Main rotor speed.

(v) Position of collective, cyclic and yaw controls.

(vi) Rates of roll, pitch and yaw.

(vii) Engine torque.

(viii) Tail rotor torque.

(ix) Rotorcraft weight and distribution.

(x) Incidence/Vertical Airspeed or Sideslip/Lateral Acceleration.

6 FLIGHT LIMITATIONS AND THE RELATION BETWEEN STATIC, STRENGTH TEST, RIG TESTS, GROUND RUNNING TESTS AND FLIGHT TESTS

6.1 A programme shall be agreed with the Rotorcraft Project Director on the advice of the Airworthiness Division, R.A.E., demonstrating the logic by which the fatigue and strength rig tests will be related to the flight programme to substantiate component lives and overall structural strength.

6.2 The initial flight testing shall conform to the limitations recommended in Leaflet 900/1, para 3.2 for development rotorcraft. This limits flying, prior to completion of static strength tests, to those manoeuvres in which the loads do not exceed 80% of the unfactored design loads. In the case of new designs on which there is less experience a lower figure may be desirable.

6.3 As information on flight loads and temperatures is obtained the ground and rig test loading conditions shall, if practicable, be adjusted as considered necessary to conform to the measured flight loads (see Leaflet 200/2, para 3.3 (i) Note).

6.4 All structural and dynamic component design alterations found necessary as a result of static tests and fatigue rig running should be incorporated in the flight rotorcraft as the tests proceed.

6.5 When the static strength tests have been completed satisfactorily, the flight tests required by para 4.2 shall be made.

6.6 The rotorcraft shall be subjected to a proof speed test on the Transmission and Rotor system in a representative rotorcraft under ground running condition or if available in a rotor rig. During such testing the dynamic component stresses called up in para 5.3 shall be measured. The test shall include running up to 15% in excess of the Rotor Maximum R.P.M. (Power On).

6.7 Stress monitoring of key stresses in dynamic components will continue during intensive flying and ground running tests which will be an integral part of the overall structural test programme called up in para 6.1.

CHAPTER 1016

VIBRATION AND DYNAMIC STABILITY

1 OBJECT

1.1 The measurement and where possible control of vibration is an important aspect in the development programme for all rotorcraft. The tests called up in this Chapter are concerned with the measurement of vibration levels throughout the structure and dynamic components to demonstrate that they do not reach structurally damaging levels and that the vibration levels at crew stations and equipment bays do not exceed specification levels and that function and performance are unaffected.

1.2 Leaflet 1016/1 contains information on the qualitative assessment of vibration.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 500, 501 and 600.

3 APPLICABILITY

3.1 See Chapter 1000 para 3.

4 MEASURING EQUIPMENT

4.1 Measuring equipment and the number and positioning of vibration sensors should be agreed with the Rotorcraft Project Officer.

4.2 Airborne Vibration Recorder equipment shall be provided capable of measuring all orders of main and tail rotor speed normally associated with the rotorcraft including harmonies and heterodyne frequencies which may arise.

4.3 Provision should be made to ensure that vibration measurements to include engine orders and high speed accessory frequencies as part of investigations.

4.4 In isolated cases hand held vibration recording equipment may be used as an aid to investigation.

4.5 Rotor blade tracking equipment may be used as a means of assessing and controlling this basic cause of vibration.

5 TESTS

5.1 In the course of the flight test programme the rotorcraft will be subjected to a comprehensive vibration survey which will progress from a theoretical analysis through rig and ground running tests, terminating with flight tests and a comprehensive vibration survey.

5.2 From these tests an in depth knowledge of the vibration characteristics of the rotorcraft can be expected to evolve and also an understanding of a level of main rotor excitation which can be used as a basis for all vibration assessment in service including airframe, transmission, dynamic components, power units, control systems and equipment.

5.3 Vibration testing will include the following:

(i) Vibration measurements throughout the flight envelope and ranges of weight, C of G and rotorspeed.

(ii) Equipment vibration.

(iii) Vibration absorber assessment and tuning.

(iv) Main rotor blade tracking and blade interchangeability.

(v) Tail rotor balancing.

(vi) Resonance and stabiliser flutter monitoring.

(vii) Vibration clearance for all operational configurations (e.g. external stores).

(viii) Natural frequency ('bonk') checks on all new equipment, followed by flight checks if shown necessary.

(ix) Testing should be extended to give consideration to the effects of in service wear in transmission components on airframe life.

5.4 GROUND RESONANCE

5.4.1 The test programme shall include tests to demonstrate that the rotorcraft is free from ground resonance. Such tests shall be undertaken prior to first flight of prototype Rotorcraft and following the introduction of major changes which may affect the dynamic characteristics of the Rotorcraft. The ground running programme shall cover the following:

(i) Rotor RPM from 0 to max.

(ii) Changes of wheel configuration, tyre and oleo pressures.

(iii) Control inputs, including collective pitch.

(iv) AFCS/SAS engagements.

The ground resonance programme and procedure shall be agreed with the Rotorcraft Project Director.

LEAFLET 1016/1

VIBRATION AND DYNAMIC STABILITY

QUALITATIVE ASSESSMENT OF VIBRATION

1 INTRODUCTION

1.1 This leaflet considers the extent to which subjective opinion can be used to assess rotorcraft vibration.

2 ADVISORY INFORMATION

2.1 The presence of vibration to a perceptible level is unlikely to be avoided in rotorcraft. Vibrations are usually referred to in terms of rotorcraft revolutionary speed and generally only the lower frequency orders are perceptible. Frequently these are not so much structurally damaging but intrusive to the crew environment. It is important that this distinction is understood and that subjective opinion cannot be taken as a basis to decide whether a vibration is structurally damaging. In any case individual tolerance varies considerably. Structural integrity and equipment or system environmental clearances must always be based on quantitative measurements.

2.2 To avoid the possibility of idiosyncratic judgements, qualitative assessments should be based (in all but minor matters) on a number of individual opinions. Minority views should not be disregarded, but the reasons for them resolved. Note should be made of the prior experience of those involved in the assessment, in terms of their flying hours in total, on the type or on similar types of rotorcraft and systems, and in the particular aircrew function involved.

2.3 Subjective assessment of vibration should not be restricted solely to that of the pilot, the intensity and overall impression can change considerably around the cabin. In many cases the concentration required for an observer to perform his role for example on radar or anti-submarine duties may mean that a vibration will be much more intrusive although considered tolerable by the pilot.

2.4 To facilitate consistent expression of subjective opinions regarding the effect of vibration on crew environment, a rating scale was devised by A. & A. E. E. This scale offers an easily understood, common approach to vibration reporting and is included in this Leaflet as Table 1. The scale allows the crew to express the effect that is experienced during a short test, and is not unlike some other Rating Scales, intended to permit a judgement as to the acceptability of the vibration over an extended period of time, or for a particular mission. This restriction reflects the difficulty of making such judgements based on short tests, and the acceptability of vibration for extended periods is best determined from the known, acceptable characteristics of existing in-Service rotorcraft performing similar roles (again the rating scale of Table 1 should be used), or where such information does not exist, from extended tests simulating Service use of the rotorcraft.

2.5 Vibration rating assessments should always be accompanied by an estimate of the vibration order in terms of N x R (N = Number of occurrences per rotor revolution). At the higher orders of vibration frequency, it is not easy to deduce multiples of NR and the assessment may have to be against a general norm, i.e. low, medium or high frequency. It should also be appreciated that in some cases, a subjective assessment of two frequencies beating together can result in a mistaken impression as regards the order. Where any doubt exists the frequency should be confirmed by measurement.

2.6 The proposed rating scale as included in this leaflet places emphasis on the opinions being that of aircrew. This does not mean the casual passenger opinions should be discounted but such opinions may be influenced by a lack of understanding as to what is normal for a given regime of flight.

TABLE 1

RATING SCALE FOR QUALITATIVE ASSESSMENT OF VIBRATION

SCALE No.	**LEVEL**	**SUBJECTIVE VIBRATION ASSESSMENT**
		No perceptible vibration
1 2 3	Light	Not apparent to experienced aircrew occupied by their tasks, but noticeable if attention is directed to it or not otherwise occupied
4 5 6	Moderate	Experienced aircrew, fully occupied by their tasks, are aware of vibration but it does not intrude so that work is affected, at least over a short period.
7 8 9	Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task may be affected, but control of the rotorcraft or aircrew duties are not seriously impaired if, when necessary, secondary tasks are neglected.
10	Intolerable	Physical discomfort or concern for the safety of the rotorcraft. Sole preoccupation to reduce vibration.

Note: Assessment using the above scale should include an estimate of the predominant frequency which may be expressed in relation to the rotor speed e.g. IR = One per rotor revolution.

CHAPTER 1017

RESCUE HOISTS, EXTERNAL CARGO AND ROLE EQUIPMENT

1 OBJECTIVE

1.1 The object of the test of this Chapter is to ensure that the internally and externally carried and actuated items of Role Equipment are capable of fulfilling the tasks defined in the Rotorcraft Specification without hazard to the Rotorcraft.

1.2 This Chapter covers the whole range of internally and externally carried role equipment except for Armament role equipment, which is covered by Chapter 1014.

1.3 The following paragraphs of this Chapter require consultation between the Contractor and the Rotorcraft Project Director to determine and agree the extent of testing required. The agreed testing requirement shall then be incorporated into either the Rotorcraft Specification or the Contract.

2 RELEVANT DESIGN REQUIREMENTS

2.1 DEF STAN 00-970 Volume 2 Chapters 100, 102, 105, 111, 205, 501, 600, 714, 721, 723 and 900.

3 APPLICABILITY

3.1 The tests in this Chapter apply to all new Rotorcraft and new Role Equipment, and to all Rotorcraft and Role Equipment where modifications have been made which are likely to affect tests which ensured compliance with the original Specification. However, when a Rotorcraft is fitted with Role Equipment designed and approved to foreign military or civil requirements, the extent to which the Rotorcraft shall then be tested to the requirements of this Chapter shall be defined by the Rotorcraft Project Director.

4 EQUIPMENT

4.1 INSTRUMENTATION

4.1.1 The standard of instrumentation shall be such that a continuous recording can be obtained of flight conditions (e.g. IAS, Altitude Rotorcraft attitudes and accelerations, etc.) and role equipment installation conditions together with data for critically interfacing systems. It is important particularly where data is to be recorded on more than one medium that timing and event data are able to be closely correlated for meaningful analysis.

4.1.2 The Rotorcraft Design Authority is generally responsible for determining the parameters to be recorded as appropriate to the trials. Where particular parameters are required to be recorded during joint MoD/contractor trials or during MoD trials these will be defined in the Rotorcraft Specification. The parameters recommended for consideration are detailed in Leaflet 1017/1.

4.1.3 Some of the tests of this Chapter may present significant hazards to the Rotorcraft; therefore, the use of telemetry for such tests should be considered.

4.1.4 Where the position of the external equipment relative to the Rotorcraft is critical for safety or other reasons then this shall be recorded by photographic or video equipment wherever possible, utilising on-board, ground based or chase aircraft facilities. This should be so positioned as to be capable of determining position and motion in six degrees of freedom. Where small items of equipment are involved and resolution may become critical then only photographic recording should be considered.

4.2 EQUIPMENT

4.2.1 The requirements of paragraph 4.2 of Chapter 1000 (General Flight Test Requirements Systems and Structures) shall apply to the tests of this Chapter.

4.2.2 Use of Dummy loads and stores having representative mass, moments of inertia and aerodynamic characteristics would be acceptable for certain of the tests where this would be expedient.

4.2.3 Safety equipment and functional test equipment shall be available as necessary to minimise risk to both the Rotorcraft and the Trials Objectives.

4.2.4 An anthropomorphic dummy may be required for rescue hoist trials.

5 LOADING

5.1 The tests of this chapter shall be made at the mass and centre-of-gravity position appropriate to the particular role under test. Where the role is such that a range of mass and CG positions are possible, then that range should be explored to the full, otherwise the tests may be made at any convenient mass and centre-of-gravity position provided that this does not result in the achievable flight envelope being more restrictive than the maximum envelope as defined by the Rotorcraft Specification taking into account limitations imposed by the carriage of external stores or other loads (i.e. maximum normal acceleration, altitude, etc.).

5.2 On Rotorcraft powered by a single engine, external stores or other loads must not be carried on any tests which may result in, or require the engine be shut down, unless the store or load is specifically required for the trial. Such stores should be jettisonable, wherever possible.

6 GENERAL TEST CONDITIONS

6.1 When a Rotorcraft has one ore more Role Equipment installation with similar characteristics, then the tests required by this Chapter shall be carried out on one typical installation with consideration being given to supplementary tests on the others. When the Rotorcraft configuration is to any degree non-symmetrical, then tests shall be carried out to determine the effects on otherwise similar installations, unless it can be shown from wind tunnel or other test results that such effects are likely to be minimal.

6.2 The tests shall be conducted on installations which are fully representative of the final Service standard(s) and where the installation(s)/system(s) interface with others, they shall also be representative of the Service standard.

6.3 When tests to this Chapter have been carried out on prototype or development Rotorcraft it may be necessary to repeat the tests in whole or in part on a production installation, to ensure that production methods and/or modification standards have not degraded the characteristics of the installation(s) to an unacceptable degree. The amount of re-testing will depend on the nature of any changes and the effects particularly on the dynamic, vibration and handling characteristics of the installation(s).

6.4 The requirements of this Chapter first cover the general case and then state more detailed requirements for specific cases as follows:

(a) Rescue hoists and SAR equipment

(b) Trailed Magnetic Anomaly Detectors (MAD)

(c) Dipping SONAR.

(d) External cargo hooks, single and multi point

(e) Abseiling and roping equipment

(f) Internal and external stretchers

(g) Airborne surveillance equipment

(h) Searchlights

(i) Internal cargo winches

(j) Parachuting (static line and free drop) for personnel and cargo and free dropping of both retarded and non-retarded loads.

6.5 It should be noted that other Rotorcraft role applications do exist, not currently on the UK inventory and not covered by the above list. An example would be mine sweeping.

Should new specification requirements arise, a schedule of test requirements shall be discussed and agreed with the Rotorcraft Project Director during the Project Definition phase and incorporated in the Rotorcraft Specification.

6.6 Ground and flight test conditions are to be representative of the Service roles of the Rotorcraft, in the range of climatic and environmental conditions defined in the Rotorcraft Specification. However for any test where any loss of power could lead to a forced landing, the tests shall be done in weather conditions such that an engine out landing may be safely made.

6.7 Tests shall be made within the Operational limits laid down in the environmental and technical specifications for the Role Equipment and stores specified. The authority of the Rotorcraft Project Director shall be obtained before attempting to carry out any tests which might involve inadvertent exceedance of these limits.

6.8 The aerodynamic standard of the Rotorcraft, in so far as it is likely to affect the results of the tests, shall be fully representative of the in-service standard. Should this not be so, the Rotorcraft Project Director must be informed of any deficiencies and the probable effect these will have on the Rotorcraft's operational clearance. Testing shall include such other role equipment as may affect the performance of either the Rotorcraft or the equipment under test.

6.9 Consideration shall be given to evidence from component and system rig test of both Rotorcraft, winches, role equipment and release units to plan the ground and flight tests to indicate the level of caution to be applied in approaching tests at the extremes of the flight envelope.

6.10 Where the role equipment includes the fitment of external equipment such as winches, underslung load carriers, etc. which are required to function in flight, icing trials shall include an investigation of the effects of such equipment on the ice accretion of the Rotorcraft. These tests will be called up as part of the testing required by Chapter 1006. Clearance of role equipment as part of hot and/or cold climatic trials shall be given consideration.

6.11 Some of the tests of this Chapter have requirements in common with other Chapters of Part 9 and Part 10 and efforts should be made to integrate those tests in the interests of economy of time and effort.

6.12 Note shall be taken of the content of AP101A-1105-1A, 1B and 1C (respectively Carriage of Cargo by Helicopters - General Information, Underslung load clearances - Internal load clearances) when planning the tests of this chapter.

7 GROUND TESTS

7.1 The need for functional testing on the ground of all release and jettison systems shall be discussed by the Contractor with the Rotorcraft Project Director. Wherever possible these functional tests should be carried out over the appropriate range of Rotorcraft attitudes and at such a height above ground that the store or load can be seen to clear the Rotorcraft structure and other stores and equipment.

7.2 Tests in accordance with Chapter 238 of DEF STAN 05-123 shall be carried out to determine the suitability of special to type servicing equipment provided for use with all the installations and equipment.

7.3 Electromagnetic compatibility (EMC) testing in accordance with Chapter 1011 and Radio Frequency Hazard tests in an appropriate radio frequency environment shall be carried out to establish that inadvertent release or jettison of stores does not occur.

7.4 Ground testing shall be carried out to demonstrate that no item of Role Equipment, where stowed or deployed in the operational mode, inhibits crew escape or the automatic/manual deployment of life rafts and other safety equipment.

8 FLIGHT TESTS

8.1 GENERAL

8.1.1 The nature of the testing of hoists, external cargo and other role equipment on a Rotorcraft is such that there is often a need for specialist test facilities to be used. In the case of MAD, Dipping Sonar, Airborne Surveillance Equipment, and Airborne Searchlights, such tests require a comprehensive avionic test programme and could call for trials against ships, other representative targets, and use facilities such as surveyed ranges.

8.1.2 The effect of role equipment on Rotorcraft vibration shall be investigated; as shall the effect of Rotorcraft vibration on the equipment carriers, and Rotorcraft structure local to the attachments of the role equipment.,

8.1.3 The EMC testing of paragraph 7.3 shall be supplemented by flight testing where appropriate.

8.1.4 Stresses/strains shall be measured in the Rotorcraft structure local to the role equipment attachments in order to facilitate determination of static and fatigue strengths.

8.1.5 Tests involving the external deployment of personnel or equipment must be approached with caution. Rescue hoist operators involved in prototype or trial installations shall have local approval based on tuition and experience. All personnel involved in abseiling, roping or freefall parachuting shall hold approvals to the satisfaction of the Rotorcraft Project Director.

8.1.6 Where loads and/or role equipment are suspended from the Rotorcraft by means of cables or strops, tests shall be made to demonstrate that there is no coupling of the Automatic Flight Control System (AFCS) with the vibratory modes of the cables(s)/strop(s) and load(s)/equipment(s). This shall be carried out for the full range of allowable loads and cable/strop lengths in conjunction with the tests of Chapter 1010.

8.2 RESCUE HOISTS

8.2.1 Tests shall be carried out to establish that the rescue hoist(s) can be safely and effectively operated throughout the required flight envelope with a small margin for occasional use beyond these limits.

8.2.2 Measurement of extension and recovery times shall be made to ensure compliance with the Specification. This shall be carried out with appropriate loads and utilising representative duty cycles. The following parameters shall be recorded during these tests:

(a) Temperature of hoist motor(s)

(b) Temperature of hydraulic fluids, where appropriate

(c) Hydraulic pressures, where appropriate

(d) Voltage and current of supply to hoist monitor, where appropriate

(e) Cable velocity

8.2.3 Ease of ingress and egress through cabin door(s) and other apertures to be assessed, including the manipulation of stretchers or casualties who may be partially or wholly incapacitated.

8.2.4 Ease of deployment and retraction of hoist(s), where applicable.

8.2.5 Lifts with anthropomorphic dummies and live subjects shall take place, as appropriate.

8.2.6 Assessment shall be made of the effectiveness of auxiliary hover control, where appropriate.

8.2.7 The effectiveness of any cable cutting facility, whether mechanical or utilising Electroexplosive devices (EEDs), shall be assessed at no load, with varying loads, and with varying cable angles and extensions.

8.3 MAGNETIC ANOMALY DETECTORS (MAD)

8.3.1 An assessment shall be made of the ability to deploy and recover/retract the Sensor head throughout the required flight envelope. Assessment shall be made of any available emergency means for recovery and stowage/locking of the MAD ‘Bird’.

8.3.2 Measurement of extension and recovery times shall be made to ensure compliance with the Specification Representative duty cycles shall be used and the following parameters shall be recorded during these tests:

(a) Temperature of hoist motor(s)

(b) Temperature of hydraulic fluids, where appropriate

(c) Hydraulic pressures, where appropriate

(d) Voltage and current of supply to hoist motor, where appropriate

(e) Cable velocity

8.3.3 An assessment shall be made of the effectiveness of any jettison/cable cutting facility, whether mechanical or EED at varying deployments/cable lengths throughout the specified flight envelope and as judged appropriate after consideration of results of rig/ground tests which might indicate the possibility of problems.

8.3.4 The operation and performance of the MAD shall be assessed, to ensure compliance with the Rotorcraft Specification.

8.4 DIPPING SONAR

8.4.1 An assessment shall be made of the ability to deploy and recover the sensor body throughout the required flight envelope. Assessment shall be made of any available emergency means for recovery and stowage/locking of the Sonar body.

8.4.2 Measurement of deployment and recovery times shall be made to ensure compliance with the Specification. Representative duty cycles shall be used and the following parameters shall be recorded during these tests:

(a) Temperature of hoist motor(s)

(b) Temperature of hydraulic fluids, where applicable

(c) Hydraulic pressures, where applicable

(d) Voltage and current of supply to hoist motor, where applicable

(e) Temperature of sensor body (to determine propensity for ice formation on initial immersions)

(f) Cable velocity

8.4.3 An assessment shall be made in conjunction with the tests of Chapter 1010, of the AFCS coupled modes.

8.4.4 The operation and performance of the Sonar shall be assessed, to ensure compliance with the Rotorcraft Specification.

8.5 EXTERNAL CARGO

8.5.1 An assessment shall be made of the ability to attach and release single loads to and from individual and multiple suspension points as appropriate.

8.5.2 Tests shall be carried out to establish that the external load carrying facilities can be safely and effectively operated throughout the required flight envelope.

8.5.3 There shall be a handling assessment, through the specified flight envelope with representative loads and strop/cable lengths, on any and all of the suspension points as appropriate.

8.5.4 Stresses and loads shall be measured in both hooks and attachments, together with loads and effective angles in strops and cables.

8.5.5 There shall be an assessment of the effects and hazards of emergency release, jettison and such failure cases as may be judged appropriate. This shall include an assessment of the possible effects of release malfunction which could result in partial release of multipoint suspended single load or an unplanned distribution of individual loads on single attachment points.

8.5.6 Where underslung loads are of a size or character likely to affect the performance of RF antennae installed in the lower part of the Rotorcraft, then tests shall be carried out to determine any effects on the performance of Doppler and Radio Altimeter and on any communications equipment wholly or partially dependent on such antennae.

8.5.7 Where the requirement for the carriage of underslung loads results in prolonged flight at substantially greater nose down attitudes than normal, then tests shall be carried out to determine any adverse effects on Rotorcraft structure, systems, and equipment; in particular fluid systems including:

(a) The fuel system in terms of fuel flow rates and unusable fuel

(b) Lubrication systems for the Rotorcraft's transmission, engines and auxiliary power units

(c) Hydraulic systems in terms of pressures and fluid temperatures

8.6 ABSEILING AND ROPING

8.6.1 Techniques for use shall be assessed, including ease and safety of egress from the Rotorcraft. The Dispatcher's role and his ability to operate effectively as controller and safety monitor shall also be included in the assessment.

8.6.2 There shall be an assessment of effects on the Rotorcraft's handling during abseiling/roping.

8.7 INTERNALLY AND EXTERNALLY CARRIED STRETCHERS (LITTERS)

8.7.1 An assessment shall be made of the ability to install and remove the removable fittings and stretchers and the time taken. There shall also be an assessment of the ease of ingress and egress for stretchers and attendants together with the ability to use specified items of medical or other equipment.

8.7.2 There shall be an assessment of the total environment of the stretcher compartment(s) including vibration, noise, temperatures and ventilation, to determine compliance with the Specification.

8.7.3 There shall be a general handling assessment of the Rotorcraft with any external stretcher housings installed to determine whether any adverse effects are incurred or whether limitations need to be imposed during their use.

8.7.4 There shall be a live qualitative assessment of both internal and external installations in terms of the comfort of occupants and the ability of attendants to perform their tasks in the Rotorcraft environment.

8.8 AIRBORNE SURVEILLANCE EQUIPMENT (CAMERAS, FLIR, ETC.)

8.8.1 A general assessment shall be made of any change in the handling characteristics of the Rotorcraft where the equipment is carried externally.

8.8.2 Vibration and stress measurements shall, where applicable, be made on both equipment and its attachments to the Rotorcraft structure.

8.8.3 Temperature measurements shall be made to ensure that ventilation and cooling are to the Specification requirements.

8.8.4 There shall be an assessment of the functioning and general performance of the equipment in the Rotorcraft environment, to ensure compliance with the Rotorcraft Specification (refer para 8.1.1). Where such equipment has a telemetry data link facility, the assessment shall include full transmit/receive tests and EMC testing as appropriate to the role.

8.9 AIRBORNE SEARCHLIGHTS

8.9.1 A general assessment shall be made of any change in the handling characteristics of the Rotorcraft where the equipment is carried externally or deployed for use.

8.9.2 Vibration and stress measurements shall, where applicable, be made on both equipment and its attachments to the Rotorcraft structure.

8.9.3 Temperature measurements shall be made on both the equipment and local structure to ensure that ventilation and cooling are adequate to keep both within the Specification limits.

8.9.4 An assessment shall be made of the effects of the searchlights switching and operating loads on the Rotorcraft electrical system in accordance with the requirements of Chapters 1003 and 1011.

8.9.5 There shall be an assessment of the functioning and general performance of the equipment in the Rotorcraft environment, to ensure compliance with the Rotorcraft Specification.

8.10 INTERNAL CARGO WINCHES

8.10.1 Tests shall be carried out to establish that the cargo winch(es) can be safely and effectively operated throughout the required flight envelope.

8.10.2 Measurement of raising and lowering speeds/times shall be made to ensure compliance with the Specification. This shall be carried out with appropriate loads and utilising representative duty cycles. The following parameters shall be recorded during these tests.

(a) Temperature of winch motor(s)

(b) Temperature of hydraulic fluids, where appropriate

(c) Hydraulic fluid pressures, where appropriate

(d) Voltage and current of supply to winch motor(s), where appropriate

8.10.3 An assessment of the ability to handle both high and low density loads in the cabin environment shall be made, together with ingress and egress through cabin doors and other apertures.

8.10.4 An assessment shall be made of the effectiveness of any jettison/cable cutting facility, whether mechanical or EED, at varying cable extension and load types throughout the specified flight envelope.

8.10.5 An assessment shall be made of the Automatic Flight Control System coupled modes, in conjunction with the tests of Chapter 1010.

8.10.6 Where winches are electrically powered, there shall be an assessment of the switching and operating loads on the Rotorcraft electrical system in accordance with the requirements of Chapters 1003 and 1011.

8.10.7 There shall be an assessment of emergency cut-out/stop facilities to ensure adequate warning of potential hazards to personnel and Rotorcraft, together with demonstration that effective safety margins are embodied in the systems and procedures.

8.10.8 Vibration and stress measurements shall be made, where appropriate, on equipment and attachments to rotorcraft structure.

8.10.9 There shall be a general handling assessment of the rotorcraft during winching operations in order to determine what limitations, if any, may need to be imposed on the Rotorcraft's flight operations.

8.11 PARACHUTING AND LOAD DROPPING

8.11.1 Tests shall be carried out to establish that all specified equipment for both personnel and parachutable loads, performs adequately and safely, including ensuring that static lines, etc. do not hazard either Rotorcraft crew or other personnel.

8.11.2 Tests shall be carried out to establish that all specified loads, whether free falling or retarded, can be ejected/extracted from the Rotorcraft and do not hazard either Rotorcraft, crew or other personnel.

LEAFLET 1017/1

RESCUE HOISTS, EXTERNAL CARGO AND ROLE EQUIPMENT TEST INSTRUMENTATION PARAMETERS

1 INTRODUCTION

1.1 Details of parameter ranges, accuracies, resolutions, etc., are given in Leaflet 1000/1, Table 1.

1.2 The following parameters are typical of those which may need to be recorded during the tests of Chapter 1017. The first part details those which will be required to define the Rotorcraft performance and motions, the second part those which are required to determine the behaviour and performance of the various systems and role equipment.

(i) Parameters defining rotorcraft behaviour:

Item No	Parameter
1	Time base
2	Manual event marker
3	Crew speech
4	Indicated airspeed
4A	Ground Speed
5	Altitude (pressure)
6	Altitude (radar altimeter)
7	Total temperature (or Ambient/Outside air (OAT))
9	Pitch attitude
10	Roll angle
11	Sideslip angle
12	Heading
13	Pitch rate
14	Roll rate
15	Yaw rate
16	Longitudinal acceleration
17	Lateral acceleration
18	Normal acceleration
22	Failure state
24	Fuel contents
25	Stick position (Pitch)
26	Stick position (Roll)
27	Yaw control pedal position
31	Pitch trim position
32	Roll trim position
33	Yaw trim position
39	FCS State/mode

Item No	Parameter
100	Collective pitch control position
101	Collective pitch trim position
103	Main rotor speed
104	Input drive torque
107	Main rotor head moment
123	Air intake temperature
	Longitudinal and Lateral doppler velocity
	Vertical velocity doppler
	Angle of tilt of rotor/wing relative to wing/ fuselage

(ii) Parameters specific to role(s):

Item No	Parameter
For the tests of 8.2,8.3, 8.4,8.5 and 8.10	Hoist/winch motor body temperature(s)
	Hoist/winch motor hydraulic fluid temperature(s)
	Hoist/winch motor hydraulic fluid pressures
	Hoist/winch motor electrical supply voltage, including transients
	Hoist/winch motor electrical supply current, including transients
	Hoist/winch body/structure vibration in three axes
	Stresses/loads at hoist/winch attachments to Rotorcraft
	Steady/transient loads in cables/strops
	** Angles of cables/strops relative to hoist/winch head or attachment points
	** Position/motion of loads and equipment relative to Rotorcraft and externally carried stores

** Note that it may be necessary to use optical sensors or cameras appropriately configured to determine these parameters since many types of sensor are liable to errors caused by acceleration affects.

	Length of cable paid out from hoist/winch
	Velocity of hoist/winch cable
	Event for cable cut or load release/jettison

Item No	Paramater
For the tests of 8.6 and 8.11	Cameras shall be installed to record the motion/position of personnel and loads during exit and departure from the Rotorcraft and behaviour of static lines, etc.
For the tests of 8.8 and 8.9	Three axis vibration at attachment points of equipment to Rotorcraft structure Loads/stresses at attachment points Temperatures of equipment and local structure to determine adequacy of cooling/ventilation system
For the tests of para 8.5	Fuel flow rate(s) to engine(s) and APU(s) Fuel contents APU(s) and transmission Oil temperatures on pump inlet, outlet and on cooling systems for engine(s), APU(s) and transmission Hydraulic reservoir(s) fluid levels Hydraulic pump inlet and outlet fluid pressures and temperatures

2 TELEMETRY

2.1 Several of the tests of Chapter 1017 could result in hazard to the rotorcraft so that for these telemetry would be considered appropriate. Where telemetry is not, for any reason, feasible, then real time monitoring of the critical parameters should be provisioned for the flight tests engineer. The following parameters should be considered for telemetry, details of ranges, etc., are given in Leaflet 1001, Table 2.

Item No	Parameter
1	Time base
2	Manual event
3	Crew speech
4	Indicated airspeed
5	Altitude (pressure)
7	Pitch attitude
8	Roll angle
10	Heading
11	Pitch rate
12	Roll rate

Item No	Parameter
13	Yaw rate
16	Normal acceleration
17	Pitch control position
18	Yaw control position
19	Roll control position
100	Collective pitch control position
101	Main rotor speed
102	Input torque
103	Main rotor head moment
111	Tail rotor drive shaft torque
	Altitude (radar altimeter)
	Ground speed, longitudinal
	Ground speed, lateral
	Vertical velocity, doppler
	Angle of title of rotor/wing relative to wing/fuselage
	Load in cable/strop
	Event for load release/jettison

PART 10 APPENDIX No.2
FLIGHT TESTS - INSTALLATIONS AND STRUCTURES
U.S. MILITARY SPECIFICATIONS, STANDARDS AND HANDBOOKS

CHAPTER 1000: GENERAL FLIGHT TEST REQUIREMENTS - SYSTEMS AND STRUCTURES

1000	MIL-STD-490	SPECIFICATION PRACTICES
	MIL-STD-961	MILITARY SPECIFICATIONS AND ASSOCIATED DOCUMENTS
	MIL-STD-962	MILITARY STANDARDS, HANDBOOKS, AND BULLETINS, PREPARATION OF
	MIL-STS-2074	FAILURE CLASSIFICATION FOR RELIABILITY TESTING
	MIL-STD-2124	FLIGHT DATA RECORDER, FUNCTIONAL STANDARDS FOR
	MIL-D-8708	DEMONSTRATION, AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-25381	FLIGHT TESTING, ELECTRIC SYSTEM, PILOTED AIRCRAFT AND GUIDED MISSILE, GENERAL REQUIREMENTS FOR
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1001: ENGINES

1001	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-2069	REQUIREMENTS FOR AIRCRAFT NONNUCLEAR SURVIVABILITY PROGRAM
	MIL-D-8708	DEMONSTRATION - AIRCRAFT, WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, - AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1002: AUXILIARY POWER SOURCES

1002	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-462	ELECTROMAGNETIC INTERFERENCE, CHARACTERISTICS, MEASUREMENT OF
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-P-8686	POWER UNIT, AIRCRAFT AUXILIARY, GAS TURBINE TYPE, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT
	MIL-P-85573	POWER UNIT, AIRCRAFT AUXILIARY GAS TURBINE, GENERAL SPECIFICATION FOR
	MIL-E-87145	ENVIRONMENTAL CONTROL AIRBORNE

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1003: ELECTRICAL SYSTEMS

1003	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-461	ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR THE CONTROL OF ELECTROMAGNETIC INTERFERENCE
	MIL-STD-462	ELECTROMAGNETIC INTERFERENCE, CHARACTERISTICS, MEASUREMENT OF
	MIL-STD-705	GENERATOR SETS, ENGINE DRIVEN, METHODS OF TEST AND INSTRUCTIONS
	MIL-HDBK-705	GENERATOR SETS, ELECTRIC, MEASUREMENTS AND INSTRUMENTATION METHODS
	MIL- -5088	WIRING, AEROSPACE VEHICLE
	MIL-E-6051	ELECTROMAGNETIC COMPATIBILITY REQUIREMENTS, SYSTEM
	MIL-E-7080	ELECTRIC EQUIPMENT, AIRCRAFT, SELECTION AND INSTALLATION OF
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-I-8700	INSTALLATION AND TEST OF ELECTRONIC EQUIPMENT IN AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-G-21480	GENERATOR SYSTEM, 400 HERTZ ALTERNATING CURRENT, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-25381	FLIGHT TESTING, ELECTRIC SYSTEM, PILOTED AIRCRAFT AND GUIDED MISSILE, GENERAL REQUIREMENTS FOR
	MIL-E-25499	ELECTRICAL SYSTEMS, AIRCRAFT, DESIGN AND INSTALLATION OF, GENERAL SPECIFICATION FOR
	MIL-M-25500	MOCK-UP TESTING, ELECTRIC SYSTEM, PILOTED AIRCRAFT AND GUIDED MISSILE, GENERAL REQUIREMENTS FOR
	MIL-T-28800	TEST EQUIPMENT FOR USE WITH ELECTRICAL AND ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-E-87145	ENVIRONMENTAL CONTROL, AIRBORNE

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	MIL-E-87219	ELECTRICAL POWER SYSTEMS, AIRCRAFT
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1004: HYDRAULIC SYSTEMS

1004	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-H-5440	HYDRAULIC SYSTEMS, AIRCRAFT, TYPES I AND II, DESIGN AND INSTALLATION REQUIREMENTS FOR
	MIL-T-5522	TEST REQUIREMENTS AND METHODS FOR AIRCRAFT HYDRAULIC AND EMERGENCY PNEUMATIC SYSTEMS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS

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SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1005: FUEL SYSTEMS

1005	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-F-7872	FIRE AND OVERHEAT WARNING SYSTEMS, CONTINUOUS, AIRCRAFT: TEST AND INSTALLATION OF
	MIL-F-8615	FUEL SYSTEM COMPONENTS, GENERAL SPECIFICATION FOR
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-F-17874	FUEL SYSTEMS: AIRCRAFT, INSTALLATION AND TEST OF
	MIL-T-18847	TANKS, FUEL, AIRCRAFT, AUXILIARY, EXTERNAL DESIGN AND INSTALLATION OF
	MIL-A-19736	AIR REFUELLING SYSTEMS, GENERAL SPECIFICATION OF
	MIL-E-22285	EXTINGUISHING SYSTEM, FIRE, AIRCRAFT, HIGH-RATE-TYPE, INSTALLATION AND TEST OF
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-23447	FIRE WARNING SYSTEMS, AIRCRAFT, RADIATION SENSING TYPE: TEST AND INSTALLATION OF

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1006: ICE PROTECTION SYSTEMS

1006	MIL-T-5842	TRANSPARENT AREAS, ANTI-ICING, DEFROSTING AND DEFOGGING SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-S-6625	SPRAY EQUIPMENT, AIRCRAFT WINDSHIELD ANTI-ICING
	MIL-D-8181	DETECTOR, ICE, AIR INTAKE DUCT, AIRCRAFT ENGINES AND AIRFRAME SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-E-008593	ENGINES, AIRCRAFT, TURBOSHAFT AND TURBOPROP, GENERAL SPECIFICATION FOR
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-D-8804	DE-ICING SYSTEM, PNEUMATIC BOOT, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-A-9482	ANTI-ICING EQUIPMENT FOR AIRCRAFT, HEATED SURFACE TYPE, GENERAL SPECIFICATION FOR
	MIL-T-18607	THERMAL ANTI-ICING EQUIPMENT, WING AND EMPENNAGE
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-B-81365	BLEED AIR SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-P-81655	PITOT-STATIC TUBE, L-SHAPED, COMPENSATED, ELECTRICALLY HEATED, GENERAL SPECIFICATION FOR

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

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CHAPTER 1007: CONDITIONING SYSTEMS

1007	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-STD-472	ELECTROMAGNETIC INTERFERENCE, CHARACTERISTICS, MEASUREMENT OF
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-H-18325	HEATING AND VENTILATING SYSTEMS, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-T-18606	TEST PROCEDURES FOR AIRCRAFT ENVIRONMENTAL SYSTEMS
	MIL-E-18927	ENVIRONMENTAL CONTROL SYSTEMS, AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-B-81365	BLEED AIR SYSTEMS, GENERAL SPECIFICATION FOR

CONTROLLED DISTRIBUTION:

	MIL-E-87145	ENVIRONMENTAL CONTROL, AIRBORNE
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT

1007 1.

2. RELEVANT DESIGN REQUIREMENTS

1007 2. MIL-H-18325
MIL-E-18927
MIL-B-81865

3. APPLICABILITY

1007 3.

4. INSTRUMENTATION

1007 4.

5. LOADING

1007 5. PARA: 3.17.2

6. GENERAL TEST CONDITIONS

1007.6 MIL-D-8708 PARA: 3.17.2
MIL-T-18606
MIL-D-23222

7. GROUND TESTS

1000 7. MIL-D-8708 PARA: 3.17.2
MIL-T-18606
MIL-D-23222

8. FLIGHT TESTS

1007 8.	MIL-D-8708 MIL-T-18606 MIL-D-23222	PARA: 3.17.2

CHAPTER 1008: ALIGHTING GEAR

1008	MIL-W-5013	WHEEL AND BRAKE ASSEMBLIES, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-T-6053	TESTS, IMPACT, SHOCK ABSORBER LANDING GEAR, AIRCRAFT
	MIL-B-8075	BRAKE CONTROL SYSTEMS, AIRCRAFT WHEELS, GENERAL SPECIFICATION FOR
	MIL-S-8552	LANDING GEAR, AIRCRAFT SHOCK ABSORBER (AIR-OIL TYPE)
	MIL-B-8584	BRAKE SYSTEMS, WHEEL, AIRCRAFT, DESIGN OF
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-S-8812	STEERING SYSTEM, AIRCRAFT, GENERAL REQUIREMENTS FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS

CONTROLLED DISTRIBUTION:

	MIL-L-87139	LANDING GEAR SYSTEMS
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT
1008 1.

2. RELEVANT DESIGN REQUIREMENTS
1008 2.

3. APPLICABILITY
1008 3.

4. EQUIPMENT
1008 4.

5. LOADING
1008 5.

6. GENERAL TEST CONDITIONS
1008 6.

7. GROUND TESTS
1008 7.

8. FLIGHT TESTS
1008 8.

CHAPTER 1009: DECK SECURING SYSTEMS

1009	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-T-81259	TIE-DOWNS, AIRFRAME DESIGN, REQUIREMENTS FOR
	MIL-T-85075	TIE-DOWN ASSEMBLY, HELICOPTER BLADE, GENERAL SPECIFICATION FOR

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1. OBJECT
1009 1.

2. RELEVANT DESIGN REQUIREMENTS
1009 2.

3. APPLICABILITY
1009 3.

4. EQUIPMENT
1009 4.

5. LOADING
1009 5.

6. GENERAL TEST CONDITIONS
1009 6.

7. GROUND TESTS
1009 7.

8 FLIGHT TESTS
1009 8.

CHAPTER 1010: POWERED FLYING CONTROLS

1010	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND HELICOPTER
	MIL-F-9490	FLIGHT CONTROL SYSTEMS - DESIGN, INSTALLATION AND TT OF PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-C-18244	CONTROL AND STABILISATION SYSTEMS: AUTOMATIC, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-F-18372	FLIGHT CONTROL SYSTEMS: DESIGN, INSTALLATION AND TEST OF, AIRCRAFT, (GENERAL SPECIFICATION FOR)
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION

1010 1.

2. RELEVANT DESIGN REQUIREMENTS

1010 2.

3. APPLICABILITY

1010 3.

4. EQUIPMENT

1010 4.

5. LOADING

1010 5.

6. GENERAL TEST CONDITIONS

1010 6.

7. GROUND TESTS

1010 7.

8. FLIGHT TESTS

1010 8.

CHAPTER 1011: ELECTROMAGNETIC COMPATIBILITY OF SAFETY CRITICAL SYSTEMS

1011	MIL-HDBK-235	ELECTROMAGNETIC (RADIATED) ENVIRONMENT CONSIDERATIONS FOR DESIGN AND PROCUREMENT OF ELECTRICAL AND ELECTRONIC EQUIPMENT, SUBSYSTEMS AND SYSTEMS
	MIL-HDBK-237	ELECTROMAGNETIC COMPATIBILITY MANAGEMENT GUIDE FOR PLATFORMS, SYSTEMS AND EQUIPMENT
	MIL-HDBK-241	DESIGN GUIDE FOR ELECTROMAGNETIC INTERFERENCE (EMI) REDUCTION IN POWER SUPPLIES
	MIL-HDBK-253	GUIDANCE FOR THE DESIGN AND TEST OF SYSTEMS PROTECTED AGAINST THE EFFECTS OF ELECTROMAGNETIC ENERGY
	MIL-STD-461	ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR THE CONTROL OF ELECTROMAGNETIC INTERFERENCE
	MIL-STD-462	ELECTROMAGNETIC INTERFERENCE, CHARACTERISTICS, MEASUREMENT OF
	MIL-STD-469	RADAR ENGINEERING DESIGN REQUIREMENTS, ELECTROMAGNETIC COMPATIBILITY
	MIL-STD-810	ENVIRONMENTAL TEST METHODS AND ENGINEERING GUIDELINES
	MIL-B-5087	BONDING, ELECTRICAL, AND LIGHTNING PROTECTION, FOR AEROSPACE SYSTEMS
	MIL-E-6051	ELECTROMAGNETIC COMPATIBILITY REQUIREMENTS, SYSTEM
	MIL-I-8700	INSTALLATION AND TEST OF ELECTRONIC EQUIPMENT IN AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-T-28800	TEST EQUIPMENT FOR USE WITH ELECTRICAL AND ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR

CONTROLLED DISTRIBUTION:

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT

1011 1.

2. RELEVANT DESIGN REQUIREMENTS

1011 2.

3. APPLICABILITY

1011 3.

4. EQUIPMENT

1011 4.

5. GENERAL TEST CONDITIONS AND REQUIREMENTS

1011 5.

6. TEST DETAILS

1011 6.

CHAPTER 1012: ESCAPE SYSTEMS AND FLOATATION GEAR

1012	MIL-STD-872	TEST REQUIREMENTS AND PROCEDURES FOR AIRCRAFT EMERGENCY GROUND AND DITCHING ESCAPE PROVISIONS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-S-87328	SURVIVAL AND FLOATATION SYSTEM, AIRBORNE, SPECIFICATION FOR

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT

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2. RELEVANT DESIGN REQUIREMENTS

1012 2.

3. APPLICABILITY

1012 3.

4. EQUIPMENT

1012 4.

5. TEST DETAILS

1012 5.

CHAPTER 1013: WATERPROOFING

1013	MIL-W-006729	WATERTIGHTNESS OF AIRCRAFT, TESTING, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT
1013 1.

2. RELEVANT DESIGN REQUIREMENTS
1013 2.

3. APPLICABILITY
1013 3.

4. GROUND TESTS
1013.4.

5. FLIGHT TESTS
1013.5.

CHAPTER 1014: ARMAMENT INSTALLATIONS

1014	MIL-STD-461	ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR THE CONTROL OF ELECTROMAGNETIC INTERFERENCE
	MIL-STD-1289	GROUND FIT AND COMPATIBILITY TESTS OF AIRBORNE STORES, PROCEDURE FOR TESTING, STORE SUSPENSION AND RELEASE EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-STD-1763	AIRCRAFT/STORES CERTIFICATION PROCEDURES
	MIL-E-5400	ELECTRONIC EQUIPMENT, AEROSPACE, GENERAL SPECIFICATION FOR
	MIL-T-7743	TESTING, STORE SUSPENSION AND RELEASE EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-A-8591	AIRBORNE STORES, SUSPENSION EQUIPMENT AND AIRCRAFT-STORE INTERFACE (CARRIAGE PHASE) ; GENERAL DESIGN CRITERIA FOR
	MIL-I-8670	INSTALLATION OF FIXED GUNS AND ASSOCIATED EQUIPMENT IN NAVAL AIRCRAFT
	MIL-I-8671	INSTALLATION OF DROPPABLE STORES AND ASSOCIATED RELEASE SYSTEMS
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-8708	DEMONSTRATION, AIRCRAFT WEAPON SYSTEMS GENERAL SPECIFICATION FOR
	MIL-H-18325	HEATING AND VENTILATING SYSTEMS, AIRCRAFT, GENERAL SPECIFICATION FOR
	MIL-L-22769	LAUNCHER, WEAPONS, AIRBORNE AND ASSOCIATED EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-D-81980	DESIGN AND EVALUATION OF SIGNAL TRANSMISSION SUBSYSTEMS, GENERAL SPECIFICATION FOR
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION
1014 1.

2. GENERAL
1014 2.

3. APPLICABILITY
1014 3.

4. WEAPONS OR STORES CARRIAGE AND RELEASE
1014 4.

5. GUN FIRING TESTS
1014 5.

6. ROCKETS AND GUIDED WEAPONS
1014 6.

7. INTERNAL DISPENSERS
1014 7.

8. AIR CARRIED MINE DISPENSERS
1014 8.

9. ICING CLEARANCE
1014 9.

10. CLIMATIC CLEARANCE
1014 10.

CHAPTER 1015: STRUCTURES

1015	MIL-STD-1530	AIRCRAFT STRUCTURAL INTEGRITY PROGRAM, AIRPLANE REQUIREMENTS
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-A-8860	AIRPLANE STRENGTH AND RIGIDITY, GENERAL SPECIFICATION FOR
	MIL-A-8861	AIRPLANE STRENGTH AND RIGIDITY, FLIGHT LOADS
	MIL-A-008862	AIRPLANE STRENGTH AND RIGIDITY, LANDPLANE LANDING AND GROUND HANDLING
	MIL-A-8863	AIRPLANE STRENGTH AND RIGIDITY, GROUND LOADS FOR NAVY ACQUIRED AIRPLANES
	MIL-A-8865	AIRPLANE STRENGTH AND RIGIDITY, MISCELLANEOUS LOADS
	MIL-A-8866	AIRPLANE STRENGTH AND RIGIDITY, RELIABILITY REQUIREMENTS, REPEATED LOADS, FATIGUE, AND DAMAGE TOLERANCE
	MIL-A-8867	AIRPLANE STRENGTH AND RIGIDITY, GROUND TESTS
	MIL-A-8868	AIRPLANE STRENGTH AND RIGIDITY, DATA AND REPORTS
	MIL-A-8870	AIRPLANE STRENGTH AND RIGIDITY, VIBRATION, FLUTTER AND DIVERGENCE
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

CONTROLLED DISTRIBUTION:

	AFGS-87221	AIRCRAFT STRUCTURES - GENERAL SPECIFICATION FOR
	SD-24	DESIGN AND CONSTRUCTION OF AIRCRAFT WEAPON SYSTEMS. VOL II - ROTARY WING AIRCRAFT
	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. INTRODUCTION
1015 1.

2. BASIC REQUIREMENTS
1015 2.

3. APPLICABILITY
1015 3.

4. TEST REQUIREMENTS
1015 4.

5. MEASUREMENTS
1015 5.

6. FLIGHT LIMITATIONS AND THE RELATION BETWEEN STATIC, STRENGTH TEST, RIG TESTS, GROUND RUNNING TESTS AND FLIGHT -TESTS
1015 6.

CHAPTER 1016: VIBRATION AND DYNAMIC STABILITY

1016	MIL-STD-1530	AIRCRAFT STRUCTURAL INTEGRITY PROGRAM, AIRPLANE REQUIREMENTS
	MIL-H-8501	HELICOPTER FLYING AND GROUND QUALITIES, GENERAL REQUIREMENTS FOR
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-8708	DEMONSTRATION: AIRCRAFT WEAPON SYSTEMS, GENERAL SPECIFICATION FOR
	MIL-A-8860	AIRPLANE STRENGTH AND RIGIDITY, GENERAL SPECIFICATION FOR
	MIL-A-8868	AIRPLANE STRENGTH AND RIGIDITY, DATA AND REPORTS
	MIL-A-8870	AIRPLANE STRENGTH AND RIGIDITY, VIBRATION FLUTTER AND DIVERGENCE
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-F-83300	FLYING QUALITIES OF PILOTED V/STOL AIRCRAFT

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	SD-8706	GENERAL SPECIFICATION FOR DESIGN EXAMINATIONS, ENGINEERING, AIRCRAFT WEAPON SYSTEMS

1. OBJECT
1016 1.

2. RELEVANT DESIGN REQUIREMENTS
1016 2.

3. APPLICABILITY
1016 3.

4. MEASURING EQUIPMENT
1016 4.

5. TESTS
1016 5.

CHAPTER 1017: RESCUE HOIST EXTERNAL CARGO AND ROLE EQUIPMENT

1017	MIL-STD-250	AIRCREW STATION CONTROLS AND DISPLAYS FOR ROTARY WING AIRCRAFT
	MIL-T-8679	TEST REQUIREMENTS, GROUND, HELICOPTER
	MIL-D-23222	DEMONSTRATION REQUIREMENTS FOR HELICOPTERS
	MIL-H-23599	HOOK HELICOPTER RESCUE
	MIL-T-28800	TEST EQUIPMENT FOR USE WITH ELECTRICAL AND ELECTRONIC EQUIPMENT, GENERAL SPECIFICATION FOR
	MIL-D-81980	DESIGN AND EVALUATION OF SIGNAL TRANSMISSION SUBSYSTEMS, GENERAL SPECIFICATION FOR

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1. OBJECT

1017 1.

2. RELEVANT DESIGN REQUIREMENTS

1017 2.

3. APPLICABILITY

1017 3.

4. EQUIPMENT

1017 4.

5. SCOPE OF TRIALS

1017 5.

6. TESTS REQUIRED

1017 6. MIL-H-23599A PARA: 4.5.4

7. GENERAL NOTE

1017 7.

VARIATION NOTE

NOTE:

This electronic copy of Defence Standard 00-970 (Vol 1, Amdt 4 and Vol 2, Amdt 12) is at variance with the hard copy master due to minor editorial corrections. These corrections shall be taken into account during the current revision of the Standard.

VOLUME 1

PART 1 - CHANGES

LEAFLET 101/2

Page 6	-	Table 2, 'Temperature' column - should be all minus values

LEAFLET 101/4

Page 1	-	Specific Humidity, line 2, 'meteorology' spelled incorrectly

LEAFLET 101/5

Page 1	-	Para 1.2, line 7, should read 'an immediate emergency', not 'am'
Page 3	-	Para 2.3.3, line 4, should read 'para 3.2 'not 'pars 3.2'
	-	Para 2.4, line 2, brackets are not closed

CHAPTER 102

Page 9	-	Para 7, first letter missing from heading, para 7.1, 7.1.1.(ii) and (iii)

LEAFLET 102/0

Page 1	-	R & M 2644, remove carrier return after 'cockpit'

LEAFLET 102/4

Page 1	-	Para 3.1, line 1, 'look' should read 'lock'
		Para 4.1, lines 3 & 6, 'look' should read 'lock'
Page 2	-	Para 5.2, line 1, should read 'geometric lock', not 'look'.
	-	Para 5.3, line 2, 'prints' should read 'points'
	-	Para 5.4, line 4, 'look' should read 'lock'
	-	Check 'hook', 'hood', 'latch', etc
Page 3	-	Para 6.2, line 1, 'looks' should read 'locks'
	-	Para 8.1, line 6, should read 'whole locking', not 'looking'

LEAFLET 103

Page 4	-	Para 7.4, line 1, 'far interior' should read 'for interior'

LEAFLET 104/1

Page 1/2	-	Incorrect Header - should read 'Leaflet 104/1'

LEAFLET 105/1

Page 9 - ‘Present Position - pp’ - should read ‘PP’

LEAFLET 105/4

Page 3 - Para 2.1.3.(v), line 9, ‘IJND’ should read ‘LJND’

CHAPTER 106

Page 1 - Para 1.2, line 4, ‘e1equipment’ should read ‘equipment’

CHAPTER 107

Page 2 - Para 2.2.2, line 3, ‘spectable’ should read ‘spectacle’
Page 4 - Para 2.7.2, line 2, should read ‘shall’, not ‘shah’
Page 9 - Para 11.3.4, line 4, ‘altitude’, not ‘attitude’ ?
Page 10 - Para 11.3.9, line 3, ‘pen-nit’ - should read ‘permit’

CHAPTER 107

Page 24 - Item No. 5, location, ‘alternatively’ hyphenated, but ‘n’ missing

LEAFLET 108/2

Page 6 - Para 2.1, line 8, ‘sensitivies’ should read ‘sensitivities’

LEAFLET 108/4

Page 2 - Para 2.2.1, line 4, ‘wig’ should read ‘will’
Page 3 - Para 3.4.1, line 7, brackets closed, but not opened

CHAPTER 110

Page 3 - Para 2.6.1, line 1, capital ‘E’ at beginning of line

VOLUME 1

PART 2 - CHANGES

CHAPTER 200

1.11	‘judgements’ corrected
1.6	‘i’ changed to ‘I’ at beginning of sentence

CHAPTER 201

5.7	‘judgements’ corrected
Fig 1	‘DESIGNINGTO’ changed to ‘DESIGNING TO’
1.1	‘maintainwith’ changed to ‘maintain with’
1.3	‘componentsincluding’ changed to ‘components including’

LEAFLET 201/1

1.5	‘thatwhere’ changed to ‘that where’
1.6	‘lifea’ changed to ‘life a’

LEAFLET 201/3

5.1.4	‘40%,according’ changed to ‘40%, according’
5.1.6	‘1.5on’ changed to ‘1.5 on’
5.1.6	‘leading,and’ changed to ‘leading, and’
P9 Ref 2	‘structura’ changed to ‘structural’
	‘loadings.Royal’ changed to ‘loadings. Royal’
Table 1	‘CONTRUCTION’ changed to ‘CONSTRUCTION’

LEAFLET 201/4

Pages2, 4 and 6 Header	‘204/1’ changed to ‘201/4’

LEAFLET 201/8

TABLE 1	Col 5 ‘GUAGING’ changed to ‘GAUGING’
	‘SPECIF.IT’ changed to ‘SPECIF. IT’

LEAFLET 203/1

P3 and 5	Header ‘Admt’ changed to ‘Amdt’

CHAPTER 204

P2 and 4	Header ‘Admt’ changed to ‘Amdt’

CHAPTER 205

P2 and 4	Header 'Amdt5' changed to 'Amdt 5'

CHAPTER 206

P10 TABLE 1 (conversion from 50,000 lb)	'22,6800kg' changed to '22,680 kg'
3.1.3	'shag' changed to 'shall'
6.1.	6.1.1 missing, or paras misnumbered
8.2.1	'personnel dozing' changed to 'personnel doing'

LEAFLET 206/3

3.3.1 (iii)	repeated 'by' - one deleted
4.2.1	'cheeks' changed to 'checks'

LEAFLET 206/6

2.3.2	'beaning' changed to 'bearing'

LEAFLET 207/2

2.2 (I)	'ssentially' changed to 'Essentially'

CHAPTER 208

6.2.2	'performance,both' changed to 'performance, both'
8.16	'e.g.,control' changed to 'e.g. , control'
P17 Header	'Issud' changed to ' Issued'
3.2.4	'significant) ' changed to 'significantly'
3.2.5	where should brackets close?

LEAFLET 208/4

Header	'May1988' changed to 'May 1988'

LEAFLET 208/6

1.5	'taxy' changed to 'taxi'
1.6	'off lying' changed to 'of flying'

APPENDIX 1

8.4	'205 para 3.4' changed to '205 para 3.4'

APPENDIX 2

line 3	‘RIGIDITY,RELIABILITY’ changed to ‘RIGIDITY, RELIABILITY’
LINE 5	‘RIGIDITY,DATA’ changed to ‘RIGIDITY, DATA’

VOLUME 1

PART 3 - CHANGES

CHAPTER 301

4.1	‘tyre,an’ changed to ‘tyre, all’

LEAFLET 301/4

Table 1, Hd 2	‘Occurances’ changed to ‘Occurrences’
5.3.3	‘taxy’ changed to ‘taxi’

LEAFLET 302/1

4/6	‘landing,particularly’ changed to ‘landing, particularly’
5.1	‘taxy’ changed to ‘taxi’

CHAPTER 303

6.1	‘taxy’ changed to ‘taxi’

LEAFLET 305/5

2.2	‘Taxyway’ changed to ‘’Taxiway’

CHAPTER 306

p5	‘position,undercarriage’ changed to ‘position, undercarriage’
P1end line 2	‘daring’ changed to ‘during’

CHAPTER 307

1.4	‘judgment’ changed to ‘judgement’
Table 5	‘Acceleraton’ changed to ‘Acceleration’
2.2.5	Brackets open, but not closed **-** NOT CHANGED
3.2.4	Does not make sense **-** NOT CHANGED

LEAFLET 307/3

1.1	remove ‘and’.

LEAFLET 308/1

2.3	‘cover an reasonable’ changed to ‘cover all reasonable’

CHAPTER 309

1.4	‘the fitting are’ changed to ‘the fittings are’

CHAPTER 310

1.3	‘fatigue lifes’ changed to ‘fatigue lives’
1.4.5	‘extend’ changed to ‘extent’
4.2.9	‘taxy’ changed to ‘taxi’

LEAFLET 310/1

6.3	‘modem’ changed to ‘modern’

LEAFLET 310/2

4.4.1(vi)	‘taxy’ changed to ‘taxi’

APPENDIX 1

6.2	‘wi11’ changed to ‘will’
7.2	‘wi11’ changed to ‘will’
10.8	‘713reference’ changed to ‘713 reference’
13.3	‘compliance chock’ changed to ‘compliance check’

APPENDIX 2

301-7	‘SILOCK ABSORBER’ changed to ‘SHOCK ABSORBER’
	‘LANDIC GEAR’ changed to ‘LANDING GEAR’
302	‘UNDERCARRIACES’ changed to ‘UNDERCARRIAGES’
	‘STERRINC’ changed to ‘STEERING’
	‘CENERAL’ changed to ‘GENERAL’
302.3	‘DESIGN,REQUIREMENTS changed to ‘DESIGN REQUIREMENTS’
302.4	‘STRENGHT’ changed to ‘STRENGTH’
303	‘LANDING CEAR’ changed to ‘LANDING GEAR’
303-1	‘REQIREMENTS’ changed to ‘REQUIREMENTS’

VOLUME 1

PART 4 - CHANGES

LEAFLET 400/2

	‘advoided’ (hard copy) ‘avoided’ (screen)
	‘others corrosive agents’ changed to ‘other etc.)

LEAFLET 400/3

2.2	‘surfaces.For’ changed to ‘surfaces. For’
3.2	PL1 changed to PLI

LEAFLET 401/0

38/60	‘tonaircraft steel S99’ changed to ‘ton aircraft steel S99’

LEAFLET 402/3

2.7	‘accoustic’ changed to ‘acoustic’

LEAFLET 402/4

3.1	‘analagous’ changed to ‘analogous’.

LEAFLET 402/7

3.5.1	‘viscocity’ changed to ‘viscosity’
3.7	‘Flamability’ changed to ‘Flammability’

LEAFLET 403/1

2.1	‘Metalic’ changed to ‘Metallic’ (correct on hard copy - not on screen)

CHAPTER 404

5.2.2	‘th’ changed to ‘the’ (correct on hard copy - not on screen)

LEAFLET 406/1

	‘Susceptitibility’ changed to ‘susceptibility’

LEAFLET 407/1

4	‘mportant’ changed to ‘important’ (correct on hard copy - not on screen)

CHAPTER 409

‘pretretment’ changed to ‘pretreatment’ (correct on hard copy - not on screen)

LEAFLET 409/4

FIRST PAGE MISSING FROM HARD COPY

LEAFLET 409/6

4.3	‘os’ changed to ‘of’

VOLUME 1

PART 5 - CHANGES

LEAFLET 500/1

3.2	‘Aeroplace’ changed to ‘Aeroplane’

LEAFLET 500/3

6.21	‘aerodynamic,forces’ changed to ‘aerodynamic forces’ (hard copy okay - screen changed)

LEAFLET 500/4

5.5 (iii)	‘controlsurface’ changed to ‘control surface’ (hard copy okay - screen changed)

Symbols as red highlight - okay on hard copy - not correct on screen

LEAFLET 501/1

2.3	‘demonstration wi11’ changed to ‘demonstration will’ (hard copy okay - screen changed)
2.3.3	‘particularlywhen’ changed to ‘particularly when’ (hard copy okay - screen changed)

LEAFLET 501/2

4.3.1	‘manoevre’ changed to ‘manoeuvre’.

LEAFLET 501/3

Annex A 2.5	‘Futhermore’ changed to ‘Furthermore’

LEAFLET 501/5

5.2	‘deflecing’ changed to ‘deflecting’)\

ANNEX A

P10 ‘f’ changed to $\hat{f}$’

723-4 ‘u’ changed to ‘μ‘

LEAFLET 501/6

‘Oc.t’ changed to ‘Oct’

VOLUME 1

PART 6 - CHANGES

LEAFLET 600/5

6	‘Atmostpheric’ changed to ‘Atmospheric’

LEAFLET 600/7

3.2.1	‘occurrence’ changed to ‘occurrence’

LEAFLET 600/8

1.1	see print-out (not yet changed)
4	‘OCCURENCES’ changed to ‘OCCURRENCES’

CHAPTER 601

1.1	‘diverge a periodically’ changed to ‘diverge aperiodically’

LEAFLET 601/1

2.1	‘about the trim’?? Should ‘about’ be changed to ‘above’.
Table 1	‘1bf’ changed to ‘lbf’.

CHAPTER 602

3.11	‘ASYMMETRICLOADING’ changed to ‘ASYMMETRIC LOADING’ (hard copy correct - screen changed)

LEAFLET 601/2

TABLE 4	’INECPTOR’ changed to ‘INCEPTOR’ (hard copy correct - screen changed)

CHAPTER 603

2.1	'Discernable' changed to 'discernible' (hard copy okay - screen changed)

LEAFLET 603/3

3.6.4	'longitundinal' changed to 'longitudinal'

CHAPTER 605

3.2.1 and 5.2.2	'permissable' changed to ' permissible'

LEAFLET 605/0

	'MEL' changed to 'MIL' (hard copy okay - screen changed)

LEAFLET 605/6

3.1	'permissable' changed to ' permissible'

CHAPTER 606

10.1	'permissable' changed to 'permissible'
TABLE	'REQUEREMENTS' changed to 'REQUIREMENTS'

PART 6 APP 1

5	2.1	'Reponse' changed to 'Response' (hard copy okay - screen changed)

PART 6 APP 2

600/	5	'FLYINC' changed to 'FLYING'
601/	4	'ATMOSPRERIC' changed to 'ATMOSPHERIC' (hard copy okay - screen changed)

VOLUME 1

PART 7 - CHANGES

CONTENTS

LEAFLET 727/0	REMOVED
CHAPTER 735	Leaflet 715/0changed to 735/0 Leaflet715/1 changed to 735/12
LEAFLET 723/4	ADDED

LEAFLET 700/1

1.2	'cooperation' changed to co-operation' (hard copy correct - screen changed)

CHAPTER 701

4.3.5	'However,consideration' changed to 'However, consideration' (hard copy correct - screen changed)
4.3.12	'means,shall' changed to 'means shall' (hard copy correct - screen changed)

LEAFLET 701/2

4.3.1	'Aeroplance' changed to 'Aeroplane' (hard copy correct - screen changed)

CHAPTER 702

2.10	'independant' changed to 'independent'

LEAFLET 702/1

5 Component	'performa' changed to ' perform a' (hard copy correct - screen changed)

LEAFLET 702/4

2.4.2	'intemal' changed to 'internal' (hard copy correct - screen changed)

LEAFLET 704/1

2.3 ‘formating’ changed to ‘formatting’

LEAFLET 707/2

‘1oads’ changed to ‘loads’

LEAFLET 708/1

4.4.3 ‘bandwith’ changed to ‘bandwidth’
(hard copy correct - screen changed)

LEAFLET 709/2

3 ‘siezures’ changed to ‘seizures’

LEAFLET 709/3

3 (iii) ‘substantia1’ changed to ‘substantial’

Tables 3 and 4 contain √ (hard copy correct - screen changed)
Symbol Wingdings changed to Symbol Set SWA

CHAPTER 710

8.5 ‘COOLIING’ changed to ‘COOLING’
(hard copy correct - screen changed)

LEAFLET 710/1

2.3 ‘REQIJIREMENTS’ changed to ‘REQUIREMENTS’
(hard copy okay - screen changed)

LEAFLET 710/4

MISSING??

CHAPTER 711

9.11.1 ‘areoplane’ changed to ‘aeroplane’
(hard copy correct - screen changed)

LEAFLET 711/2

3.11 ‘contruction’ changed to ‘construction’

LEAFLET 711/4

3.3 ‘manuoeuvres’ changed to ‘manoeuvres’

CHAPTER 712

2.1 (x) ‘hard-picked’ changed to ‘hard-packed’
(hard copy correct - screen changed)

LEAFLET 712/0

MechEng 62 ‘modem’ changed to ‘modern’

LEAFLET 712/2

3.3 ‘Celcius’ changed to ‘Celsius’
(hard copy correct - screen changed)

CHAPTER 713

7.4.(1) ‘tab inserted
(hard copy correct - screen changed)

LEAFLET 713/3

Header ‘Admt’ changed to ‘Amdt’
(hard copy correct - screen changed)

CHAPTER 715

1.3 changed to ‘a high order **of** accuracy’

CHAPTER 717

2.5 line 3 close brackets after ‘type’

CHAPTER 718

2.1(1) ‘otitic’ changed to ‘optic’
3.1.4 NOTE - brackets closed but not open

LEAFLET 718/0

Header	'Amdt7' changed to 'Amdt 7' (hard copy correct - screen changed)

LEAFLET 718/2

Header	'Amdt7' changed to 'Amdt 7' (hard copy correct - screen changed)

LEAFLET 718/6

1.1	'an altitude' chanted
2.3.3 (line 8)	should text in brackets read 'longer than say 30 minutes'??

CHAPTER 719

Header	'Chpter' changed to 'Chapter' (hard copy correct - screen changed)

LEAFLET 719/3

Header	'Leaflt' changed to 'Leaflet' (hard copy correct - screen changed)

CHAPTER 720

Table 2	'speciments' changed to 'specimens'
3.2	'Test No 1' underlined in Work (hard copy ok)

LEAFLET 718/0

Header	'Amdt7' changed to 'Amdt 7' (hard copy correct - screen changed)

LEAFLET 720/0

DTD925	'acylic' changed to 'acrylic'

LEAFLET 721/1

2.1.8	'accidently' changed to 'accidentally'

CHAPTER 722

3.1.1 ‘shalt’ changed to ‘shall’
(hard copy correct - screen changed)

LEAFLET 722/3

3.2 ‘Brackets closed but not opened
(hard copy correct - screen changed)

CHAPTER 723

Table 2 Ref 3f, col6 ‘(o.654)’ changed to ‘(0.65A)’
Table 3 Ref ‘1F’ changed to ‘1f’

LEAFLET 723/0

HTI-R-78-109 ‘Programms’ changed to ‘Programs’

LEAFLET 723/1

5.1 ‘absorbsion’ changed to ‘absorption’

LEAFLET 723/2

3.2.4(1) ‘suspectible’ changed to ‘’
3.1.1 (iii) ‘absorbtion’ changed to ‘absorption’
5.3 (vi) ‘polyprolylene’ changed to ‘polypropylene’
2.1.4 ‘operation; are’ changed to ‘operations are’

LEAFLET 723/4

1.3 (line 5) ‘electromognetic’ changed to ‘electromagnetic’
‘optical’ duplicated - one deleted.
3.2.2(b) (iii) para 4 ‘fule changed to ‘fuel’
References ‘G uidelines’ spaced removed

CHAPTER 726

2.3.2 ‘fight’ changed to ‘flight’
(hard copy correct - screen changed)

CHAPTER 729

4.6 (ii)	‘inadvertant’ changed to ‘inadvertent’
	‘inadvertantly’ changed to ‘inadvertently’
9-3	DOES NOT MAKE SENSE

LEAFLET 729/2

3	‘taxy’ changed to ‘taxi’
	(hard copy correct - screen changed)

LEAFLET 730/10

Header	‘DEFSTAN’ changed to ‘DEF STAN’

LEAFLET 731/2

Header	‘Amdt12’ changed to ‘Amdt 12’
	(hard copy correct - screen changed)

CHAPTER 736

6.5.1	‘nicely’ changed to ‘likely’
App 1 8.1 (line 3)	‘in-fight’ changed to ‘in-flight’

VOLUME 1

PART 8 - CHANGES

CHAPTER 800

2.2.3 (v)	‘pars’ changed to ‘para’ (hard copy correct - screen changed)

CHAPTER 801

2.1 (I) (ii)	etc corrected

VOLUME 1

PART 9 - CHANGES

LEAFLET 900/2

Table 2 p6	‘CONDITONS’ changed to ‘CONDITIONS’ (hard copy correct - screen changed)

LEAFLET 900/3

2.2	‘judgments’ changed to ‘judgements’
2.4	‘judgments’ changed to ‘judgements’

CHAPTER 901

7.1	‘taxy’ changed to ‘taxi’

LEAFLET 901/2

4.1.2 (VII)	‘taxy’ changed to ‘taxi’

CHAPTER 902

7.9	‘REDUCTION’ changed to ‘REDUCED’

CHAPTER 903

Header	‘Andt’ changed to ‘Amdt’

CHAPTER 910

4.1	‘assimulated’ changed to ‘assimilated’
6.3	‘througout’ changed to ‘throughout’

CHAPTER 913

1.1 (line 5)	‘occurrance’ changed to ‘occurrence’

CHAPTER 917

7.4.5 (line 6)	‘progessively’ changed to ‘progressively’
7.5.1 (line 3)	‘concemed’ changed to ‘concerned’

LEAFLET 918/2

Header changed to Nov 1990 (Amdt 12)

LEAFLET 918/4

1 (b)	‘cloud is’ changed to ‘clouds’

APPENDIX 2

900-6	‘CONDITTIONS’ changed to ‘CONDITIONS’
909	‘AIRPI.ANES’ changed to ‘AIRPLANES’
911	‘MANOUEVRES’ changed to ‘MANOEUVRES’
916	‘FLYINC’ changed to ‘FLYING’

VOLUME 1

PART 10 - CHANGES

CHAPTER 1000

Header	left-hand. Chapter no and amdt date - wrong way round

CHAPTER 1004

1.1 (I)	‘actuactors’ changed to ‘actuators’ (hard copy correct - screen changed)

CHAPTER 1005

6.3 1	‘guage’ changed to ‘gauge’
6.3.2	‘guaging’ changed to ‘gauging’
7.3.1. (v)	‘loding’ changed to ‘lodging’ (hard copy correct - screen changed)

CHAPTER 1007

6.1 (ii) (c)	‘Taxy’ changed to ‘Taxi’ (hard copy wrong)

CHAPTER 1008

6.2 (iii)	‘Taxy’ changed to ‘Taxi’
5.1	‘tile’ changed to ‘the’

CHAPTER 1013

4.4 Note	‘flourescin’ changed to ‘cluorescein’ (hard copy correct - screen changed)

APPENDIX 2

1011 Part 2	‘SUCEPTIBILITY’ changed to ‘SUSCEPTIBILITY’

VOLUME 2

PART 1 - CHANGES

LEAFLET 102/2

6.2 ‘below’ changed to ‘be low’

LEAFLET 105/1

Tabbing for ‘FEATHER BEACON, ENGAGE’ corrected

CHAPTER 110

Fig 1 (page 5) - figure reduced to allow page number to show in PDF

CHAPTER 111

Headers and footers added to page 3/4

VOLUME 2

PART 2 - CHANGES

CHAPTER 200

Table 2	‘+’ symbol corrected to ‘t’ as per hard copy

CHAPTER 207

P11	Last line on page ‘now’ changed to ‘not’

VOLUME 2

PART 3 - CHANGES

LEAFLET 301/3

Super/subscript for MTL corrected

CHAPTER 306

Headers and footers added to page 3/4

VOLUME 2

PART 4 - CHANGES

LEAFLET 402/2

‘Leaflet 400/2’ missing from page 2

CHAPTER 401

Page ‘1 of 1’ changed to ‘1/2’

VOLUME 2

PART 5 - CHANGES

LEAFLET 501/2

Blank pages left in to maintain page numbering between pages 25 and 27 (start of Annex A) and 33 and 35 (Annex A and Annex B)
Fig 6 - reduced in size so that it, plus title, fits on page
Annex B 4.1.1 iv(b) and v +/- symbol missing

LEAFLET 501/3

Blank pages left in to maintain page numbering between pages 15 and 17 (start of Annex b) and 19 and 21 (Annex C and Annex D)

VOLUME 2

PART 7 - CHANGES

LEAFLET 702-6

‘bode’ changed to ‘above’

LEAFLET 704-0

‘Andt’ changed to ‘Amdt’

LEAFLET 704-1

Page 3/4 no header

CHAPTER 705

Pages 11, 12 - wrong header

LEAFLET 706-0

‘sevo’ changed to ‘servo’
‘contacters’ changed to ‘contactors’

LEAFLET 706-2

Pages 2 and 3 - text alignment
‘1oads’ changed to ‘loads’

LEAFLET 707-1

Page 15/16 incomplete header

LEAFLET 708-0

‘toincidents’ changed to ‘to incidents’

CHAPTER 712

Pages 3, 5, 7, 9, 11, 13, 15 incorrect header

LEAFLET 713-0

Text alignment

CHAPTER 715

‘cateogry’ changed to ‘category’
‘accummulation’ changed to ‘accumulation’
‘REQUIRMENTS’ changed to ‘REQUIREMENTS’
‘EMMISSIONS’ CHANGED TO ‘EMISSIONS’
‘equences’ changed to ‘sequences’

ANNEX A Page 4 - line spacing (bottom of page)

LEAFLET 715-3

‘INTERGRATING’ changed to ‘INTEGRATING’

CHAPTER 717

‘OCCURENCE’ changed to ‘OCCURRENCE’
‘PARAMATER’ changed to ‘PARAMETER’

LEAFLET 717-0

‘Programms’ changed to ‘Programs’
‘Lealflt’ changed to ‘Leaflet’

LEAFLET 717-1

‘nulcei’ changed to ‘nuclei’

LEAFLET 717-2

‘emmission’ changed to ‘emission’
‘jeopardising’ changed to ‘jeopardizing’

LEAFLET 717-3

‘absortion’ changed to ‘absorption’
‘focussing’ changed to ‘focusing’

CHAPTER 721

‘necelle’ changed to ‘nacelle’

LEAFLET 721-1

‘cognizance’ changed to ‘cognisance’

CHAPTER 727

‘judgment’ changed to ‘judgement’

CHAPTER 731

‘seater’ changed to ‘greater’

LEAFLET 731-0

‘military NOTE’ formatted

VOLUME 2

PART 8 - CHANGES

LEAFLET 800-1

Page 3/4 incorrect header

CHAPTER 806

‘horizonal’ changed to ‘horizontal’
‘Chapater’ changed to ‘Chapter’

VOLUME 2

PART 9 - CHANGES

LEAFLET 900-1

Page 5 page number

LEAFLET 900/3

Page 2 figure partly obscuring page number

CHAPTER 903

Page 7 page number

CHAPTER 908

Page 7 Fig title incomplete

CHAPTER 909

Page 7 page number

VOLUME 2

PART 10 - CHANGES

LEAFLET 1000/1

Odd page headers incorrect

CHAPTER 1003

‘Page6’ changed to ‘Page 6’

CHAPTER 1004

‘despressurisation’ changed to ‘depressurisation’

CHAPTER 1005

Page 5 heading done without text (bottom of page)

CHAPTER 1007

‘Part10’ changed to ‘Part 10’

CHAPTER 10014

Odd pages headers incorrect