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General Document 36

REPORT ON VISIT TO SWEDEN, UK AND INDONESIA, JUNE 1993

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by

J.M. GRANDAGE

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General Document 36

REPORT ON VISIT TO SWEDEN, UK AND INDONESIA, JUNE 1993

by

J.M. GRANDAGE

SUMMARY

This report describes the 1993 ICAF (International Committee on Aeronautical Fatigue) meetings in Stockholm. Associated visits were made to UK to discuss structural airworthiness requirements for military aircraft and to Indonesia to present a seminar on aircraft structural fatigue.

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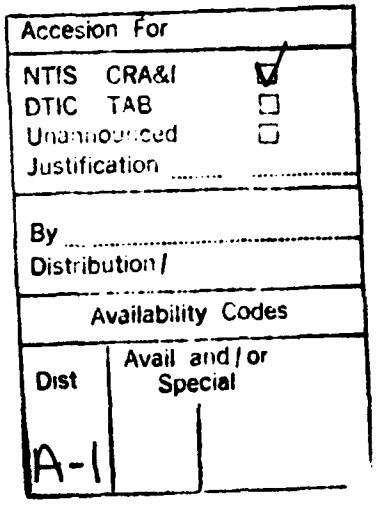

1. INTRODUCTION

The author visited Stockholm, Sweden in June 1993 with the objectives of

(a) attending, as Australian/New Zealand national delegate, the 23rd meetings of the International Committee on Aeronautical Fatigue, held in Stockholm from June 7 to June 11, 1993, and

(b) presenting a review of Australian and New Zealand investigations on aeronautical fatigue in the period April 1991 to May 1993.

Following the ICAF meetings I visited the UK Defence Research Agency (DRA) Aerospace Division, Farnborough to clarify various aspects of the UK structural airworthiness requirements for military aircraft.

While in transit to Europe I visited Industri Pesawat Terbang Nusantara (IPTN), Bandung, Indonesia to present a seminar on aircraft structural fatigue.

A summary of the itinerary and main contacts is given in Annex 1.

2. REPORT ON 1993 ICAF MEETINGS

The 1993 ICAF Meetings were held at the Scandic Crown Hotel, Stockholm from Monday June 7 to Friday June 11. The schedule for the meetings is given in Annex 2 and it is noted that Symposium paper numbers 1.1 and 1.2 were not presented. Australia and New Zealand were represented by the author as Delegate, Dr G.S. Jost and Mr S.A. Barter of ARL, Mr A.J. Emmerson of the Civil Aviation Authority, Prof. N. Mileshkin and Mr C.A. Patching of the Royal Melbourne Institute of Technology and Mr W.L. Price of the Defence Scientific Establishment (DSE), New Zealand. Total registration at the meetings was 197 of whom 32 were from Sweden. Representation by country is given in Annex 3.

2.1. Twenty third ICAF Conference, June 7-8, 1993.

As usual the Conference comprised the presentation of National Reviews of the ICAF member countries. The Reviews summarised aeronautical fatigue activities in each country over the last two years and were presented by the respective National Delegates. Annex 4 lists the contents pages from each Review and these indicate the research and investigation work in progress and completed in each country.

The Australian/New Zealand Review had been assembled by the author from material provided by contributors in Australia (mainly ARL) and the DSE, New Zealand. 200 copies were despatched to the meeting and to National Delegates and a further 125 copies were distributed to individuals within Australia and New Zealand.

The host country, Sweden, will publish the National Reviews in a single volume as the Minutes of the 23rd ICAF Conference, 1993. Individual copies of the Reviews are held at ARL.

2.2. Seventeenth ICAF Symposium, June 9-11, 1993.

In summary the Symposium comprised

(a) the invited Plantema Memorial Lecture*

(b) twenty nine formally presented papers, and

(c) twenty seven papers given as poster presentations.

This was the first ICAF Symposium at which poster papers were presented. Authors for both categories of paper were from Government agencies, academic institutions, manufacturing industry and the air-lines.

The Plantema Memorial Lecture was presented by Mr Ulf Goranson of the Boeing Commercial Airplane Group and was entitled "Damage Tolerance: Facts and Fiction". Mr Goranson's paper summarised the application of damage tolerance principles to commercial transport aircraft with an emphasis on damage detection and maintenance programs for aging aircraft.

The twenty nine presented papers included eleven papers (over one third) which related to practical experiences of fatigue life substantiation and testing. These included

(a) a British Aerospace paper on the B.Ae. Hawk,

(b) a Canadian paper on compressive induced fatigue cracking in the CF-5,

(c) Australian case studies involving the F/A-18 and Macchi MB326H,

(d) USAF experience of durability testing, and

(e) a Lockheed test program on the C-130.

Six papers dealt with life prediction methodology, more specifically the areas of cumulative damage, crack growth life, fretting fatigue and acoustic fatigue. Several papers presented probabilistic approaches to either life prediction or aircraft maintenance.

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* Dr Frederik J. Plantema, of the National Aeronautical Research Institute, Holland was largely responsible for the formation of ICAF and was its first Secretary General until his death in 1966. The first Plantema Memorial Lecture was presented at the 1967 ICAF Symposium, held in Melbourne.

Seven papers were in the general areas of stress analysis and fracture mechanics methods. A recurring theme of several of these was the problem of multi-site damage. Finally two papers were on the fatigue performance of composites.

Annex 5 lists the authors, titles and abstracts of the 30 papers (including the Plantema Lecture) which were presented to the Symposium.

Poster presentations were introduced at the Stockholm Symposium as a means of offering exposure to an increased range of authors and topics and expanding the level of participation. They were considered by the Symposium Organising Committee to be worthy of presentation but not of such a high calibre as to justify formal presentation to the Symposium. Annex 6 lists the authors and titles of the 27 poster papers.

All 57 papers (of both categories), including the Plantema Memorial Lecture, will be published late in 1993 under the title of "Durability and Structural Reliability of Airframes", Proceedings of the 17th Symposium of the International Committee on Aeronautical Fatigue, 9-11 June 1993, Stockholm, Sweden, edited by A.F. Blom. Copies of most of the Symposium papers are held by the author at ARL.

2.3 Commercial Exhibitions.

The following organisations presented exhibitions during the meetings:

- Boeing Commercial Airplane Company
- EMAS Scientific Publications
- Esprit Technology Inc.
- Fatigue Technology Inc.
- FFA (Aeronautical Research Institute of Sweden)
- FFV Aerotech AB
- Fokker Control Systems
- MTS Systems Corporation
- Spectrapot Inc.
- Springer Journals
- Volvo Flygmotor

2.4. Business meeting, June 11, 1993.

The formal minutes of the business meeting, and informal notes prepared by the author, are held by ARL. The next series of ICAF meetings will be held in Melbourne in May 1995 and the following issues relating to these meetings were discussed at the business meeting:

(a) date and venue,

(b) general theme and more specific topics,

(c) poster session,

(d) colloquium,

(e) publication of minutes and proceedings,

(f) attendance by authors,

(g) selection of Plantema Memorial lecturer, and

(h) involvement of non ICAF countries.

About 500 copies of a leaflet publicising the Melbourne meetings were distributed during the Conference and the Symposium in Stockholm.

The following additional issues were addressed during the business meeting:

(a) approval of the General Secretary's review for 1991-93,

(b) production and distribution of the Minutes and Proceedings of the Stockholm meetings,

(c) venues for the 1997 and 1999 meetings,

(d) distribution of ICAF documents,

(e) production of Plantema medals,

(f) future policy on admission of new member countries,

(g) election of Mr J.L. Rudd as the new US Delegate to ICAF.

3. UK FATIGUE REQUIREMENTS

3.1 Introduction and background

The Life Assessment Group of Airframes and Engines Division, ARL advises the RAAF on structural airworthiness aspects of their aircraft. In practice this involves estimating

safe fatigue lives and inspection intervals, planning and interpreting fatigue testing programs, advising on in-service monitoring procedures and investigating life extension techniques. Such advice can be needed at all stages of the fleet life-cycle from selection and procurement to life extension of aging aircraft. Frequently ARL needs to make reference to established sets of airworthiness requirements for military aircraft such as the UK Def. Stan. 00-970 and the USAF MIL-A-83444. The RAAF are at present reconsidering their airworthiness policy and are moving towards a policy which is largely based on Def. Stan. 00-970. For this reason ARL and RAAF need to improve their understanding of Def. Stan. 00-970.

The objective of the visit to DRA, Farnborough was to clarify various aspects of Def. Stan. 00-970 and its rationale and practical application.

The fatigue requirements in Def. Stan. 00-970 have been developed largely by Dr Arthur Cardrick, who heads the Requirements Group of the Airworthiness Division at DRA, Farnborough. The author had discussions with Dr Cardrick over the two days June 14-15, 1993. Prior to the discussions, contact had been established with Dr Cardrick by means of a letter which identified a large number of specific questions for discussion. Dr Cardrick had offered an initial written response to nearly all these questions and thus the first iteration of an interactive discussion had already taken place before any face-to-face discussion. At the completion of the meeting I agreed to prepare a draft summary of Def. Stan. 00-970 with comments on its rationale and practical application. Dr Cardrick agreed to review this and his comments have been incorporated into the final version (below).

There follows a summary of the airworthiness requirements in Def. Stan. 00-970, together with some explanatory comments and some examples of its application.

3.2 Def Stan. 00-970 and its rationale and application

3.2.1 Preamble

The relevant parts of Def. Stan. 00-970 are chapter 201, entitled "Fatigue damage tolerance", and the associated leaflets 201/1 to 201/8. (The term fatigue damage tolerance encompasses both the safe life and safety-by-inspection approaches to ensuring safety). The current version which was discussed by the author and Dr Cardrick is mostly dated October 1987 (Amendment 6) but leaflet 201/8 and several pages of chapter 201 are dated November 1990 (Amendment 10)*.

Some aspects of the UK structural airworthiness scene need to be recognised before discussing Def. Stan. 00-970 further. The Airworthiness Division at Farnborough has both a requirements role and a project support role in ensuring airworthiness of UK military aircraft. In the requirements role the Division takes initiatives to ensure that policies used for structural certification (a) keep pace with advances in knowledge and technology and (b) strike an appropriate balance between considerations of performance, safety and cost. These initiatives are agreed with the Ministry of Defence (MOD) and industry before

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* Other chapters have been amended as recently as December 1991 but chapter 201 and its leaflets were not changed at this time.

changes are incorporated into Def. Stan. 00-970. In the project support role the Airworthiness Division (a) advises the MOD (Project Director and Service Staff) and the Design Authority (industry) on the interpretation of the structural design requirements and (b) assesses the compliance that is submitted by the manufacturer. The two roles apply in a similar way to aircraft operated by the Air Force, Navy and Army. While the Airworthiness Division has overall responsibility for formulating the requirements and in ensuring compliance of particular projects, Dr Cardrick noted that in practice the three players (the MOD Project Director, the Design Authority and Farnborough) are generally in agreement on all issues.

Def. Stan. 00-970 has evolved from the earlier document Av.P.970 following a series of amendments and undoubtedly further developments will occur in future as technology changes and as experience is gained in applying safety-by-inspection procedures in practice. Farnborough are at present developing a simplified version of the Requirements which will be lacking in some of the background material contained in Def. Stan. 00-970.

During the discussion with Dr Cardrick it became apparent to the author that the UK has had relatively little experience of operating military aircraft fleets on a safety-by-inspection basis - less experience than the RAAF and the USAF for example. UK military aircraft are designed and operated essentially on a safe life basis. Those aspects of Def. Stan. 00-970 which concern safety-by-inspection were formulated from Farnborough's assessment of the problems which might be anticipated rather than from extensive experience of applying safety-by-inspection.

It is noted that ARL/RAAF policies on structural airworthiness have until recently been strongly influenced by the USAF damage tolerance procedures and also by USN airworthiness procedures. Some features of Def. Stan. 00-970 which contrast with these procedures are emphasised in the present report.

3.2.2 A summary of fatigue requirements in Def. Stan.00-970

The following summary should not be seen as a substitute for a reading of Def. Stan. 00-970. It attempts to summarise the requirements and procedures which would apply in typical cases, with some emphasis on the more novel features.

Para. 1.4 of chapter 201 states the three main aspects of the requirements:

(a) a good safe life,

(b) resistance to the growth of detectable damage; and

(c) tolerance to increases in spectrum severity ("such as those which arise from uncertainties in identifying and quantifying loads and in structural analysis, increases in all up weight and changes in the way in which aeroplanes are used").

The distinction between safe life and inspection-dependent structural details is made in paras. 2 and 3 of chapter 201. To qualify as safe life the following conditions (among others) shall be satisfied:

(a) the safe life shall be equal to at least the service life or a lower life if component replacement or modification is feasible, and

(b) the safe life with all the loads amplified by a factor of 1.2 shall be at least half the life from (a). (This quantifies the requirement for tolerance to spectrum severity increases).

Para. 3 states that "a detail ..... may be treated as inspection-dependent if the economic and operational consequences of inspection and subsequent modification or replacement are acceptable", and conditional on the following being established:

(a) the safe life under the anticipated and amplified loading,

(b) the likelihood of damage by (among other events) abnormalities in manufacture,

(c) the average crack growth curve to critical size, under both anticipated and amplified loading, from an appropriately established initial defect size,

(d) the reliability curve for the inspection technique in the range of defect sizes appropriate for (c), and

(e) the inspection interval under anticipated and amplified loading such that the probability of crack growth to critical size without detection is acceptably low.

Para. 5 states that "compliance with the requirements shall be demonstrated by calculations supported by relevant tests on details, sub-assemblies and the structure, component or system as a whole". Furthermore two full-scale fatigue tests are required:

"(i) A pre-production airframe shall be tested to at least three times the specified life under the design spectrum, in time for all significant design changes arising from the test to be incorporated in production aeroplanes with a minimum of retrospective modification.

(ii) A production airframe shall be tested preferably under the actual service loading as derived from a program of operational loads measurement, but an earlier test may be needed if major shortcomings are revealed by the pre-production test.

Both these tests shall be continued for five times the specified life (or an equivalent life) or until the specimen, through reasons of repair or production changes, is no longer representative". This does not imply that the test life factor (i.e. the ratio of test life to the validated safe life) must always be as high as five. After completion of the full-scale tests a residual strength test may be required and a teardown inspection of all major load carrying structure is necessary.

Para. 4 of chapter 201 deals with in-service monitoring. The requirements are firstly that a few aircraft in the fleet be fitted with comprehensive instrumentation to define the loads on

major structural components in each usage role, and secondly that each aircraft be fitted with instrumentation for estimating fatigue life consumption at significant structural locations throughout the life.

Para. 6 refers to compliance of aircraft not designed to Def. Stan. 00-970.

The requirements specified in chapter 201 are mandatory but leaflets 201/1 to 201/8 are advisory. The leaflets are now discussed. Leaflets 201/3 and 201/4 are emphasised since these cover the substantiation of safe lives and of inspection schedules.

Leaflet 201/1 outlines the scope of the remaining leaflets.

Leaflet 201/2 on material selection gives guidelines on the selection of metallic materials.

Leaflet 201/3 - Substantiation of fatigue life. This leaflet defines procedures for obtaining factors relating mean life to safe life and it also discusses loading spectra for tests and for calculations. The various factors allow for the following:

(a) statistical scatter due to fatigue life variability in similar structures tested to similar loading conditions and also due to the test life being based on only one (or at most several) specimens,

(b) uncertainty due either to calculated adjustments of test lives to allow for differences in loading between service and testing or to calculations being based on constant amplitude coupon test data alone, and

(c) uncertainty regarding service loading in the event that damaging loading actions are not monitored individually on each aircraft.

Allowance for item (a) is achieved by constructing the safe S-N curve (strictly a family of safe S-N curves assuming that mean stress effects are recognised). The safe S-N curve is derived from the mean S-N curve using a defined factor (typically 3.0) on life at the steep end of the curve and a defined factor (typically 1.5) on stress at the shallow end, and fairing the two resulting curves into eachother in the intermediate region. The factors on life and stress are in general functions of the number of items tested and the number of items fitted on each aircraft. The inherent scatter in fatigue life is assumed to be given by a $\log_{10}$ standard deviation of about 0.11 for most metallic materials, but higher scatter may be needed for some steel details. When interpreting the safe life from a representative full-scale test the mean S-N curve is first factored on a stress-wise basis to predict the test life, after which the safe S-N curve is derived from the factored mean curve and used for estimating safe service life.

Regarding items (b), differences between test and in-service loading create a need for further conservatism only in the event that the S-N curve shape is of doubtful validity for the structural detail in question. In such a situation a penalty of a factor of up to 3.0 on life is seen to be warranted. This would not often arise and in typical cases the S-N curve shape would be well-chosen and no further conservatism would be needed. Thus adjustments can be made for either the anticipated or the amplified spectrum being

different from the test load spectrum without more conservatism. The other aspect of item (b), namely the uncertainty due to calculations based on constant amplitude coupon test data alone, is generally covered by a further factor of 3.0 on life: however this is in part balanced by a reduced degree of conservatism to allow for item (a) in such cases.

To allow for uncertainty in service loading, additional factors of 1.5 and 1.2 (on life and on stress respectively) are introduced when deriving the safe S-N curve from the mean curve. This applies to structural details for which the damaging loads are unmonitored.

Tables 1 and 2 summarise the various factors for safe life derivation.

Some examples are given below of overall factors which would apply in illustrative cases. The factors can be seen to give either a safe life given an achieved test life or a test life needed to validate a required in-service safe life.

Finally leaflet 201/3 gives guide-lines on loading spectra for tests and calculations. Issues considered under the heading of test spectra include truncation of low load levels, "damage acceleration" to reduce test duration, number of cycles per flight and sequence duration.

Leaflet 201/4 - Substantiation of damage growth and associated inspection intervals. This leaflet defines procedures for deriving an average crack growth curve, allowing for statistical scatter and uncertainty, estimating residual strength and critical crack size, selecting an initial crack size and estimating the first inspection time and subsequent inspection intervals. In some respects the procedures are analogous to those in leaflet 201/3 which concern safe life.

The average crack growth curve is established over crack sizes ranging from the initial crack size to the critical size. It may be either calculated using standard fracture mechanics methods or derived from component or full-scale test data, if necessary allowing for a change in severity between the test and service loading.

In allowing for statistical scatter in crack growth lives, it is assumed that variability is the same as that for total lives used in leaflet 201/3.

When deriving a crack growth curve from test data, fracture mechanics calculations are used to relate the test loading to the service loading. A crack growth curve is predicted for the test loading: if this agrees acceptably with the empirical data the same method is used for predicting crack growth under the service loading, and no factors for uncertainty need be introduced. If the difference between the predicted crack growth life and the empirical life is more than 50%, additional factors may be required to allow for various uncertainties in the calculation method.

If the crack growth curve is based on calculations alone, the estimated crack growth life is reduced by a factor of at least 2 in the absence of stress measurement and a further 2 to allow for uncertainties in the stress intensity formulation. The resulting conservatism is in part balanced by a less conservative allowance for statistical scatter since it may be assumed that the average crack growth curve has a statistical confidence level consistent with that based on a large number of tests.

A further factor of 1.5 on crack growth life is required in the event of the damaging loads not being monitored on each aircraft individually.

Critical crack size is required such that the minimum residual strength of the damaged structure is 80% of the design ultimate load. It may be estimated using either standard elastic fracture mechanics or methods such as R-curve techniques. The critical stress intensity should recognise factors such as grain direction, material thickness, failure mode and operating temperature by using a conservative value based on relevant data. Data on residual strength from the structural test program may be used to validate the residual strength estimation procedure.

The rationale for selecting an initial crack size (i.e. the size used as a starting point on the average crack growth curve when estimating inspection intervals) is under review at Farnborough and the relevant sections will be clarified and probably modified in substance in future versions of Def. Stan. 00-970. The intent of the current version is to use the crack size which can be detected with 50% probability using the in-service NDI procedures, and to compensate for the implied lack of conservatism by introducing a factor of 2 on life when estimating inspection intervals. This is discussed further in Section 3.2.3.

When estimating the inspection times throughout the fleet life-cycle, Def. Stan. 00-970 does not in general require the time to the first inspection to be the same as the subsequent inspection interval. The first inspection (also referred to as the threshold time) will usually be at the safe life of the structural detail but consideration must be given to influences such as accidental damage, corrosion, manufacturing defects and material defects. For example poor material quality or excessive manufacturing defects might justify a requirement for significant initial cracking to be assumed on entry to service, in which case operation without inspection to the safe life would not be warranted. The inspection interval is estimated by reading the time from initial crack size to critical size off the average crack growth curve and dividing by the appropriate factors. (The current version of Def. Stan. 00-970 refers to a simulation model for deriving inspection intervals. Such an approach has not yet been applied in the UK).

Table 3 summarises the factors given in leaflet 201/4. Section 3.2.4 illustrates some typical values of overall factors which would be needed when obtaining inspection intervals from crack growth lives.

Leaflet 201/5 covers testing procedures needed to support the substantiation of the Def. Stan. 00-970 requirements. The main feature of the test program is the two full-scale fatigue tests as already noted above. The leaflet essentially amplifies the relevant paragraphs of chapter 201. It should be noted that the design development test program may need to include tests on coupons and components with the amplified (x 1.20) loading spectrum.

Leaflet 201/6 describes service monitoring procedures, both the fleetwide recording program and the more comprehensive operational loads measurement (OLM) program. The latter is an important part of the UK's overall structural integrity assurance procedure and it has been the subject of several recent DRA papers. The OLM is discussed below at Section 3.2.3.

Leaflet 201/7 deals with operation beyond the initially validated safe life and is particularly relevant to the operation of aging aircraft. Life extension is also discussed in a recent paper by Dr Cardrick (ref. 1). Leaflet 201/7 calls for a reassessment of the structure to identify locations which may become critical during the proposed life extension. If such locations cannot be operated using inspection-dependency the relevant components must be replaced. Inspection-dependent locations must be reassessed, paying particular attention to the possibility and implications of multi-site damage. Structural rework should be considered, as should the need for further testing of components or complete structures. The views in ref. 1 have not yet become DRA policy but may be incorporated into future versions of Def. Stan. 00-970. This states that "an airframe test must be done to demonstrate the extended life for the structure as a whole". Such a test would use the actual service loading and would provide data to substantiate not only the safe lives of locations which will continue to be operated on a safe life basis, but also a defined in-service inspection program for inspection-dependent items.

Leaflet 201/8 defines the procedure for clearance of aircraft designed to requirements other than Def. Stan. 00-970. An extensive review of all aspects governing in-service fatigue performance is required. This may lead to requirements for additional analysis, testing and load monitoring.

Various aspects of Def. Stan. 00-970 differ from the requirements which ARL/RAAF are most familiar with (namely the USAF and USN requirements). These are as follows:

(a) the requirement for a safe life reduction of 50% (or less) with the loading amplified by 1.20,

(b) the two full-scale fatigue tests,

(c) testing to five life-times with the second test under demonstrably realistic loading,

(d) the safe S-N curve concept,

(e) (when estimating inspection intervals) using initial crack size for a 50% probability of detection, together with a life factor of 2,

(f) the first inspection usually at the safe life,

(g) a reduced emphasis on the "rogue flaw" (i.e. approximately 1 mm) initial crack size requirement.

(h) the assumption that the variability in crack growth life is the same as for total life, and

(i) a systematic approach to the specification of factors to allow for various sources of variability and uncertainty.

3.2.3 Explanatory comments

As noted above in Section 3.1, the author prepared a list of questions and Dr Cardrick provided preliminary answers before the Farnborough visit took place. Further questions arose during the visit. This Section uses a question-and-answer format to clarify certain aspects of Def. Stan. 00-970 and other DRA publications, explain the rationale, and discuss the application. Many of the answers are slightly edited (or unchanged) versions of Dr Cardrick's responses.

(1) *To what extent are the requirements mandatory rather than advisory? To the extent that they are advisory, how is implementation assured?*

"The Chapters are *mandatory*, although in their application this Division may sometimes identify the need for certain clarifications/variations in policy - as may the Design Authority (manufacturer) - which will be recommended to the MOD Project Director. It is customary for the Project Director to take our advice on such matters and for any significant changes to be introduced as amendments to Def. Stan. 00-970. Formally, the Leaflets *advise* on acceptable methods of compliance, but in practice they have usually been written by the principal players in design and certification (for example, this Division took the lead in shaping the Fatigue Requirements, but worked closely with Industry) and for this reason it is unusual for other methods to be used. Implementation of the procedures in the Leaflets is assured through the day-to-day Project Support activities of the Division". (Ref. 2)

(2) *Are the requirements applicable to helicopters as well as to fixed-wing aircraft?*

"The general principles apply to both aeroplanes* and helicopters and with this in mind both sides of Industry were represented in drafting Chapter 201 of Volume 1 (Aeroplanes). However, before corresponding amendments were made to Volume 2 (Helicopters) it was decided to simplify the presentation of the Requirements and then to include the simplified version, with minimal differences, in both volumes. We are now nearing publication of this simplified version, which retains the same basic policies and introduces a few refinements". (Ref. 2)

(3) *Is there a defined overriding aim of the requirements - for example to keep the probability of survival above 0.999 for a random member of the fleet, and/or to keep the risk per hour below 10^{-6}, etc?*

"The *overriding aim* of the Requirements is to strike a proper balance between performance, cost and safety. We monitor the effectiveness of our procedures by reviewing accident/incident statistics (we believe the chance of losing a British military aircraft due to a structural failure (under normal operating conditions) is about right, since it is much the same as the risk of death due to a road accident when driving a car) and by collecting information on the costs of structural maintenance and repair (for combat aircraft (those of Western origin for which figures are available), it has been customary to spend about twice the purchase price on structural maintenance and repair - about ten times the figure for a car - but it must be said that usage involves a challenging spectrum of

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* The term "aeroplane" refers to fixed-wing aircraft.

repeated loads, adverse combinations of temperature and humidity, attack by corrosive fluids and the ever-present risk of maltreatment whether by accident or enemy action; moreover, it is not unusual for an aircraft to remain in service for 30 years or more). The current Fatigue Requirements seek to reduce these costs through the introduction of several novel features including the Safe S-N approach, a supplementary check for sensitivity to increases in spectrum severity (for aeroplanes, not less than half specified life must be demonstrated with loading spectrum severity (primarily amplitude) increased by 20%) and a preliminary major fatigue test to identify any major shortcomings in time for them to be eliminated from most production items. You observe that we refer to a probability of failure of about 1 in 1000 in the statistical basis that underpins our Fatigue Requirements, but I must point out that statistical methods are used largely as transfer functions in adjusting our factors to give a uniformity of approach for different numbers of specimens tested, different variabilities etc., and not in the sense of attempting to make a meaningful estimate of the probability of failure for the structure as a whole - you will agree that the outcome of any such estimate would be strongly influenced by the assumptions needed to perform the calculation - for example the likelihood of encountering a load of 20% above Limit Load (to which the 1 in 1000 figure is related), the number of potentially-critical features in each relevant loading action and the associated probability distributions. We strongly *advocate* the use of statistical methods *as a guide* in designing *particular parts* of a structure and we accept that they have a part to play in balancing the contribution to safety of different disciplines, such as structures, engines and avionics, but we strongly *resist* pressures to specify a numerical requirement as the *cornerstone* of airworthiness compliance". (Ref. 2)

(4) *The arguments for and against two airframe tests are fairly obvious. Why have DRA become convinced that two tests are justified?*

"As noted at (3) above, evidence relating to combat aircraft suggests that the cost of structural maintenance and repair amounts to about twice the purchase cost - to round figures, for British combat aircraft, the purchase cost has been estimated to be about 10% of the total life-cycle cost, which includes engines, fuel, avionics and development costs. By contrast, our estimates show that *two* major fatigue tests can be done for about the same cost of the structural maintenance and repair on just *one* airframe. In order words, for a fleet of 200 aircraft, the cost of two major fatigue tests would amount to only about 1% of the purchase cost or about 0.5% of the *customary* spend on structural maintenance and repair. It can be seen that a major return can be expected on this small investment in a preliminary major fatigue test. Let me emphasise that the main purpose of the preliminary test is to exercise an airframe of the correct geometry (the materials may sometimes be less representative - for example, urgency may dictate that machined plate must be used instead of forgings) under reasonably realistic loading (we would adhere to the limits of the anticipated operating envelope) in order to identify any major shortcomings before production is well established - that the *sequence* of failures might have been different in a conventional test in unimportant. In summary, we believe that performing two tests rather than one is justified on economic grounds". (Ref. 2)

(5) ***What is the rationale for the 80% ultimate load minimum strength requirement for both safe life and inspection intervals?***

"For our combat aeroplanes, loads in the region of Limit Load arise fairly frequently in normal operations (the design case is associated with normal manoeuvring rather than with a design gust) and for this reason we would not wish to see the strength fall below the Design Ultimate Load for more than a short period. The 80% residual strength requirement provides a margin of only 1.2 on Limit Load. In the safe-life context, only a few aircraft in the fleet should experience any perceptible reduction in strength and most of these only during the last few percent of the life. By contrast, some categories of 'inspection-dependent' operation might lead to large numbers of aircraft operating for substantial proportions of their life at strengths that are only just above that associated with the critical crack size (I have the 'crack-arrest' concept particularly in mind). For this reason we have retained the 80% requirement even though there is likely to be conservatism elsewhere in the 'inspection-dependent' approach. It is noteworthy that our 80% figure is in close agreement with the highest residual strength requirement that arises from the approach used in MIL-A-83444". (Ref. 2)

(6) *Presumably the 80% requirement applies to the empennage as well as to wing structure. Would this enforce higher safety levels for the empennage?*

The 80% requirement applies to the complete airframe. This could lead to higher safety levels for the empennage than for the wing if the margin of the design load over the typical fatigue loads is higher for the empennage. However the data base for fin structure is generally subject to a higher level of uncertainty than that for the wing.

(7) *The absence of an explicitly defined "rogue flaw" requirement is one of several differences between Def. Stan. 00-970 and MIL-A-83444 when considering inspection intervals. Have some case studies shown whether the inspection intervals from leaflet 201/4 are more or less conservative than those from MIL-A-83444?*

"We have not found a need to compare our procedures for deriving inspection intervals with those of 83444. We have serious doubts about the outcome of an 83444 analysis in which the declared aim is to assume the inspection will just miss a crack of such size that it can be shown to be detectable on 90% of occasions *with 95% confidence*; there would obviously be a serious penalty in establishing the 95% crack size from a small sample. Note that we would adopt an initial flaw size of 0.05 inch (as in MIL-A-83444) in circumstances where it is judged to be a reasonable representation of the real threat, for example due to manufacturing defects or accidental damage". (Ref. 2)

(8) *Having performed a full-scale test with the anticipated service loading, how is the requirement for half the safe life with the amplified loading substantiated? Is a further factor introduced to allow for uncertainty?*

The requirement is substantiated by calculation using the safe S-N curve. No further conservatism is needed since the simple factoring of the spectrum does not constitute a radical departure from the anticipated loading.

(9) *Could DRA provide some detail on the simulation model approach to inspection interval estimation (as discussed in leaflet 210/4)? What is the status of model development in the UK?*

The intention has been to develop a model which estimates the risk of a crack reaching critical size in service without being detected. Input parameters are average and variability in the crack growth curve, the time of crack initiation, and inspection reliability. Inspection intervals can be estimated for a given overall probability of detection. Such a model has not yet been developed in the UK and at present no applications are envisaged.

(10) *The "threshold " time and crack size can be seen as the initial condition for the purpose of inspection interval estimation. Leaflet 201/4 appears to define (possibly in different contexts) the threshold as the time and crack size for a probability of detection (PD) of zero or of 0.5. Could this be clarified?*

The term "threshold" is to be replaced by "initial"in future versions of the Requirements including the simplified version which is now under development. The initial crack size corresponding to PD=O is relevant only if using a simulation model. With the standard deterministic approach to estimating inspection intervals the current version of Def. Stan. 00-970 requires an initial condition for PD=0.5 (together with an additional factor on life of 2.0). Earlier versions have assumed a PD close to 1.0. The rationale for using PD=0.5 is that it is not practical to estimate the crack size for a PD of 0.9 (for example) with any degree of confidence unless very large samples of data are obtained, and that with smaller data samples an unreasonable penalty will be paid if one is required to demonstrate a PD of 0.9 with a high level of confidence. In current and past UK applications of safety-by-inspection the approach has been to use the "largest crack size that may escape detection", in other words a size having a PD close to 1.0. Dr Cardrick's current view, which is partly influenced by the arbitrariness of the life factor of 2.0, is that it is preferable to use a crack size having a PD appropriately close to 1.0, even if this is based on the judgement of suitably qualified experts in the absence of quantitative data, rather than use a PD of 0.5. He is in favour of changing Def. Stan. 00-970 accordingly.

(11) *Could the overall PD concept in leaflet 201/4 also be clarified?*

The requirement that overall PD be 0.999 (para. 9.2 of leaflet 201/4) would apply when using a simulation model.. The alternative (currently used) procedure does not lead to such an estimate of overall probability.

(12) *Is there a quantitative rationale for using the factor 2.0 to compensate for using PD=0.5 rather than a PD close to 1.0?*

The choice of 2.0 was arbitrary.

(13) *How long should a safety-inspection program be continued before safety levels at the location being inspected are compromised?*

"We currently use regular inspections to maintain safety in those instances where the safe-life policy cannot be applied (as for items subject to accidental damage) and in circumstances where clearance beyound the safe life (factor typically in the region of 3 1/3) has been justified by a demonstration of damage tolerance qualities (failure shown to be associated with damage that would be evident before there was any appreciable loss of structural integrity). In this latter case, it is customary to derive inspection intervals

assuming that a crack of the 'initial size' has just been missed, and so there is no need to reduce the inspection interval with increasing life in order to compensate for the increasing chance of cracking. In principle, therefore, the items concerned could be cleared indefinitely, but in practice their life would be limited by that of other parts of the structure. Exceptionally, we have to deal with the pressure cabins of large transport aircraft that have been cleared to the civil 'fail-safe/damage-tolerance' procedures and have not been exhaustively fatigue tested. In these circumstances we have been forced to accept that 'evident' failure modes have not been demonstrated and that MSD (multiple-site damage) could occur; in such cases we would endeavour to maintain a factor of at least 2 on the test life and/or to implement regular proving tests to 1.33 times the normal (Limit) pressure. In general, we do not perceive MSD to be a threat unless three conditions are satisfied simultaneously: the duration of testing must be below our customary standard (factor about 3 1/3), stress levels must be uniform in a region of the structure where cracks could occur at nominally identical features and clearance must depend upon regular inspections - fortunately we have few such instances on British military aircraft". (Ref. 3)

(14) *If a crack is detected when applying the inspection-dependent approach, would this necessarily result in aircraft retirement/component replacement or repair, or would there be circumstances in which known-to-be-cracked structures are permitted to continue flying?*

"The application of an inspection-dependent approach is governed by the acuity of the inspection technique together with the period of growth from the 'threshold size' (that at which a crack is unlikely to be missed) to the 'critical size' (that at which the strength of the structure is reduced to 80% DUL). Using the conventional approach (as distinct from a simulation model) the time between inspections is 1/3 of the time for the crack to grow from the threshold size to the critical size. Therefore, a period of continued use after a crack has been detected is permitted if the crack can be measured and the time between inspections is reduced to not more than 1/3 of the time for the crack to grow from the measured size to the critical size. In practice, provision for such an approach is often made by deriving a reduced inspection time from a higher threshold size that is sufficient to enable most items containing small cracks to remain in service. In setting this higher threshold size - beyond which no further operation would normally be permitted - consideration is given to the minimum practical interval between inspections (bearing in mind the difficulty of the inspection) and the effect of the crack on the difficulty of repair. Where appropriate, this approach is followed even for the compact, highly stressed items which characterise combat aeroplane structures". (Ref 4)

(15) *What procedures are used in UK for calibrating/validating crack growth models used for inspection interval estimation? What level of agreement between prediction and test data would be required?*

"We would look for test evidence covering the range of relevant crack lengths and obtained under a number of severities of realistic loading" (ref. 2). In the absence of such test data for validation purposes, crack growth lives would be factored as required by para 5.1 of leaflet 201/4. DRA would expect model predictions to be within 50% of the test data.

(16) *What is the rationale for implying that scatter in crack growth rates is the same as for total lives (para. 6 leaflet 201/4)?*

"Structural joints containing fasteners are commonly found to exhibit microcracks of tensile-opening form within the first few percent of their lives and so the scatter observed in the fatigue life of these features has also been used for crack propagation under corresponding loading conditions"(ref. 2).

(17) *Para. 8.2 of leaflet 201/4 refers to residual strength calculation. Does the residual strength test on the structure have a role here, either in providing directly usable data or for calibrating a residual strength calculation procedure?*

"Yes - for both purposes mentioned" (ref.2).

(18) *In para. 1.1 of leaflet 201/3, should "tests on elements, sub-assemblies and major components" also include full-scale structures? Similarly should para. 1.3, item (iv) (a) read "....component and full-scale test data...."*

"The term 'major component test' is taken to mean full-scale test" (ref.2).

(19) *Depending on whether a complete (tip-to-tip) wing or a half wing is defined as an "item", the most optimistic factor on life from table 1 of leaflet 201/3 could be either 3.0 or 2.8 after one representative full-scale test. Which value is appropriate?*

The value of 2.8 is appropriate. " In permitting an airframe to be treated as two nominally identical components which are subjected to nominally identical loading, it is possible to take advantage of the considerable benefit of testing two specimens rather than one (even if the test is discontinued after the first failure). This benefit is much greater than the associated penalty of increasing the factor to correspond to a probability of failure of 1 in 2000. It was never intended that more than two items per aircraft should be considered in aeroplane applications (the table was extended with helicopters in mind) and the 'simplified' version will be amended accordingly" (ref. 2).

(20) *Following completion of the two full-scale tests, the first test might be judged to be a fair representation of the in-service condition, from the viewpoints of test hardware and loading. In such circumstances would an average test life be based on the two tests and would table 1 of leaflet 201/4 be used with "2 items tested" ?*

This would be unlikely but in such a fortuitous set of circumstances the safe life would be based on the two tests as suggested.

(21) Various questions apply to para. 2.2 of leaflet 201/3 and figure 1:

(a) *Is the specified equation in item (1) of para. 2.2.4 too restrictive in some instances? Was consideration given to defining the S-N curves numerically, and interpolating between defining points?*

"Difficulties have been encountered because, understandably, some parts of Industry have

been reluctant to move away from the particular in-house S-N curves which are part of the body of experience upon which their 'nominal stress' methods are based. Since the form of the S-N curve is not central to the Safe S-N approach, it is anticipated that the recommendation that the Weibull form should be used will be deleted from the revised Requirements" (ref.2).

(b) *Ref. item (iii) of para 2.2.4. When locating the "static" value of S on the safe S-N curve given that for the mean curve, what factor on stress is used? (Figure 1 apparently does not use 1.5).*

"In practical terms, the variability at low endurance is academic and a safe value equivalent to the 'B' value of static strength may be used. A factor of 1.5 would be inappropriate here since the purpose is to constrain the safe S-N curve according to the material variability in the region" (Ref. 2).

(c) *The procedure for allowing for mean stress effects is not specified.*

"Recommendations on the methods that might be used to make adjustments for changes in mean stress were omitted in order to avoid unnecessary complication" (Ref. 2).

(22) *Para. 5.1.3. item (ii) of leaflet 2013 requires stress-wise adjustment of S-N curves to give* $\Sigma \; n/N = 1$. *There are reasons in favour of stress-wise adjustment and other reasons for life-wise adjustment. Why do DRA favour stress-wise adjustment?*

"When we considered this question, the customary view was that, simplistically, the stress-wise adjustment could be associated with a change in the fatigue limit. In hindsight, I can see no obvious reason why this argument should have carried the day, but equally I can see no reason to change. I think that what really matters is that we should be consistent in what we do" (Ref. 2).

(23) *Table 2 of leaflet 2013 indicates a factor up to about 3.0 to allow for S-N curve shape. What would a typical value be ?*

"Each case must be judged individually. In practice, I have found no difficulty in reaching agreement with my colleagues in Industry on what curves should be used and in these circumstances no additional factor has been required" (Ref. 2). Thus a value of 1.0 is used typically and the value of 3.0 quoted in table 2 provides an incentive to follow realistic procedures.

(24) *When deriving a safe life from the interpretation of a full-scale test having questionable relevance to the service loading conditions, judgements will have to be made about the extent of the difference between test and service loading that can be tolerated without requiring further conservatism. In the extreme a test might be seen as so unrepresentative that the safe life assessment can only be based on calculation. Could DRA comment on this?*

It is not possible to generalise. In making such judgements consideration would need to be given to the nature and extent of the difference between the test and service load spectra,

the relevance of sequence effects in the test loading, the confidence associated with the S-N curve shape, and the likelihood that the test may have failed to reveal the most critical location. Serious misgivings in these areas could result in a factor of up to 3.0 on life being needed to allow for the uncertainty. Table 2 of leaflet 201/3 is relevant here. If the factor was to be as high as 3.0, this would become equivalent to a safe life estimate by calculation and would provide a strong case for a further test.

(25) *Ref. paras. 3.2.2 and 3.2.5 of leaflet 201/3. Can the practice of accelerating the "just significant" cycles lead to a distortion of the load spectrum and hence an unrepresentative ranking of the failure locations?*

"When used in the manner described, the practice of representing the effect of very large numbers of low amplitude cycles by a much lesser number of slightly higher amplitude cycles should not affect the"ranking" of failure locations" (Ref. 2).

(26) *Paras. 3.3 and 4.2 of leaflet 201/5 address the question of completeness of the two fatigue test articles. Would manoeuvring devices and landing gear need to be tested in their own right?*

"Yes" (Ref. 2).

(27) *Ref. para. 4.5 of leaflet 201/5. The last two sentences seem to imply that artificial damage may need to be introduced before completion of the required life. Could this be clarified?*

"The meaning here is that artificial damage would need to be inserted only in those instances where the safe life approach had not been used - for example, in the case of components subject to accidental damage and for which crack-growth data would be needed to support the choice of inspection intervals" (Ref. 2).

(28) *Is a teardown inspection mandatory for both full-scale tests? What would be the likely crack size that can confidently be detected by the NDI procedures used during teardown? Would the same NDI procedures be used for compressive and tension loaded structure?*

"A teardown inspection is mandatory for both tests and the requirement would be waived only under exceptional circumstances. In making these inspections, our practice has been to focus on the major load paths and then on the particular features along these paths where incipient cracking may be present and would require attention if it were to develop. We have no hard-and-fast rules governing the depth of inspection, but all involved are aware of the unique opportunity that is provided to examine a simulation of the weakest airframe and thereby perhaps to gain some new insight into the damage growth that has occurred". (Ref. 2).

(29) *Does DRA advocate a teardown inspection of a high life in-service aircraft late in the fleet life-cycle?*

Yes, but such teardown inspections are implemented on an opportunity basis in the UK and are not mandatory.

(30) *Does the requirement for individual monitoring (para. 4.1 of chapter 201) apply to all components or only to the wing/centre section?*

"The intent is to ensure that each aircraft in the fleet carries some form of instrumentation to assist in estimating the damage accumulation in the principal features of concern. In order to qualify for the factor for monitored structure (traditionally 3 1/3, but now somewhat refined by the Safe S-N approach), it is necessary to provide a standard of monitoring that is at least as good as that provided for the centre-wing by the use of an RAE Fatigue Meter - using a weight-corrected formula which is checked at intervals by comparison with damage calculated using Operational Loads Measurement (OLM) on a sample of aircraft in the fleet" (Ref. 2).

(31) *Ref. para. 2.9 of leaflet 201/6. What are DRA's current views on the adequacy of the fatigue meter?*

"The Fatigue Meter continues to provide a low-tech method of collecting meaningful data for the central areas of the wing/fuselage of the airframe. Obviously, as one moves further away from these conditions the data become less meaningful and other methods of monitoring become more attractive. The Tornado, for example, still carries a Fatigue Meter, but the formula for each subject location is reviewed and up-dated as necessary using the results of OLM" (Ref. 2).

(32) *Is the removal of the 1.5 life factor for absence of monitoring conditional on both fleet-wide individual monitoring (for example by fatigue meter) and periodic load measurement in a few aircraft? Para. 4 of leaflet 201/3 seems to indicate that the first of these alone is sufficient to remove the 1.5 factor. However para. 4 of chapter 201 implies that both are mandatory.*

Both programs must be implemented to justify reducing the 1.5 factor to 1.0. An intermediate approach, which is applicable to components that are not amenable to continuous fleet-wide monitoring, is to use OLM in conjunction with "Sortie Pattern Code" (SPC) monitoring on all aircraft. (The SPC corresponds to the RAAF's type-of-flight coding system). Such an approach would justify a factor between 1.0 and 1.5. In the absence of both OLM and fleet-wide monitoring programs, the required factor would be greater than 1.5.

(33) *How does the OLM program fit into the overall process of fatigue substantiation?*

The OLM program is implemented periodically on each fleet and the general aim is to provide an on-going re-validation or adjustment of the procedure used to convert the output of the fleet-wide continuous monitoring program to consumed fatigue damage for each aircraft. The OLM involves strain gauge fitment to several aircraft (at least one for each role if the fleet is sub-divided into units flying different roles) and sufficient data are recorded to derive loading information on all major components. The rationale for the OLM is that loading severity can and does change in a manner which is not recognised by the fleet-wide instrumentation, which in the case of the fatigue meter records only normal acceleration counts. OLM programs have been applied to all types of UK military aircraft (for example fighters, transports, helicopters and fixed-wing) and are repeated at intervals

of three to five years even if there is no reason to suspect a change in usage. Data from OLM programs are also used for deriving test load spectra. As noted above, the absence of an OLM program results in extra conservatism being required when estimating safe lives. Further information on OLM programs is given in Ref. 5.

(34) *Para. 1.2 of leaflet 201/7 requires an assessment to reveal any additional locations which could become critical during life extension. How should this assessment be made?*

"An example of such a feature would be the pressure cabin of a combat aeroplane. Traditionally, although no longer, cabins have been designed with a static super factor to cover fatigue and have not been fatigue tested. We now judge this super factor to have been sufficient to cover a safe life of about 6000 hours and require supplementary substantiation thereafter. The preferred solution is to examine the cabin in the light of flight load measurements and known operating stresses and show by calculation that the required safe life can be obtained with a factor of at least 2 to allow for uncertainties. Only as a last resort would parts of the cabin be treated as inspection-dependent and in these circumstances some crack growth measurements might be done by introducing artificial cracks into the fatigue test specimen (the test would continue in the usual way to prove a life beyond 6000 hours for the rest of the structure)" (Ref. 2).

3.2.4 Some typical case studies.

Several hypothetical, but realistic, situations are now considered to illustrate the application of Def. Stan. 00-970 in deriving overall factors when estimating safe lives and inspection intervals from appropriate data.

Case 1. The safe life for the wing of a fighter-type aircraft is required. A test life is available from a representative full-scale structure tested under the service loading condition. Data are available from both an OLM and a fleet-wide load monitoring program. In this case no factors are needed except for the factor implied by the safe S-N curve. This would typically be about 3 1/3. In the event of the loading and/or the local stress response being particularly severe the factor could approach 3.0 or even 2.8 in the limiting case (see Table 1). An unusually low stress level or a mild load spectrum could give a factor closer to 4.0 as a result of the damaging load cycles being located further from the steep end of the S-N curve.

Case 2. A safe life for the fin of a fighter aircraft is required. Data are available for the fin from an OLM program but the fleet-wide monitoring program does not produce loading data for the fin. A representative full-scale test has achieved a failure in the fin. Here the factor from the safe S-N curve would be greater than 3 1/3 (possibly about 4.0) assuming that the damage is mostly located further down the S-N curve than for case 1. In the absence of fleet-wide monitoring a further factor of 1.5 is required, giving an overall factor of about 6.0. In the absence of the OLM data, further conservatism would be needed.

Case 3. Inspection intervals are required for a structural location in the wing of a fighter aircraft. An empirical estimate of the crack growth curve at this location has been obtained from a representative full-scale test. The initial crack size is associated with a probability of detection of 50%. Fleet-wide monitoring and OLM data are available. Firstly a factor of about 3 1/3 is needed for scatter in crack growth life. A further factor of 2.0 is needed since we have PD=50% for the initial crack size used. No further factors are required and the overall factor is about 6.6. If the initial crack size had been associated with PD approaching 1.0 the overall factor would have remained at 3 1/3.

Case 4. This is similar to case 3 except that an empirical crack growth curve is not available. The crack growth curve has been obtained from a fracture mechanics model which has been calibrated against data from a coupon tested with a different load spectrum from the aircraft in question, and not validated further. Since the crack growth curve is obtained from a calculation, a factor of 2.3 (based on 6 or more items tested) is required for scatter in crack growth life. A factor of 2.0 is required for uncertainties in stress intensity and possibly a further factor of 2.0 would be needed if stress levels were not measured. Together with the factor of 2.0 due to PD=50%, this gives an overall factor of either 9.2 or 18.4. The last-mentioned factor of 2.0 would be removed if the initial crack size was based on a PD close to 1.0.

4. VISIT TO IPTN, INDONESIA

IPTN is an aircraft design and manufacturing facility. Its main current activities are:

(a) (jointly with CASA, Spain) design and manufacture of the CN-235 commercial passenger transport aircraft,

(b) design and manufacture of the N-250 transport aircraft,

(c) manufacture of various helicopters,

(d) manufacture of components for F-16 and other aircraft.

I presented a seminar which gave an overview of aircraft structural fatigue and an outline of some of ARL's past and present activities in this area. The seminar was attended by about 50 people and, together with time for questions, lasted about 2½ hours.

5. CONCLUDING REMARKS

The objectives of the visits to the three countries were achieved. These objectives were:

(a) to attend the ICAF meetings and present the Australian National Review,

(b) to clarify various aspects of the UK structural airworthiness requirements for military aircraft, and

(c) to establish contact with IPTN and present a seminar on fatigue.

Attendance at the ICAF meetings is an essential aspect of Australia's involvement in ICAF, which is a useful information exchange medium providing significant benefit to Australia's aeronautical community.

The discussions with Dr Cardrick have clarified many issues which will be important in ARL's continuing commitment to advise the RAAF (and the ADF generally) on their structural integrity problems.

The visit to IPTN will have strengthened the Australian defence community's association with a neighbouring country, Indonesia.

6. ACKNOWLEDGEMENTS

I wish to thank Arthur Cardrick for his time and his willingness to engage in frank discussion on a wide range of structural integrity issues.

During the visit to IPTN, Dr Marjono and several other members of IPTN staff showed great hospitality to me and this is greatly appreciated.

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4. A. Cardrick. Personal communication 29 November 1993.

5. D.M. Holford. Operational loads measurement in military aircraft. Institute of Mechanical Engineers seminar "Structural Integrity in Civil and Military Aircraft", University of Bristol, 25 September 1990.

TABLE 1* TEST FACTORS FOR USE IN THE CONSTRUCTION OF SAFE S-N CURVES

Number of Items Tested	Factor on Log mean Endurance over Steep Part of S-N Curve	Factor on Arithmetic Mean strength at High Endurances
1 Item per Aeroplane		
1	3.0	1.50
2	2.6	1.45
3	2.5	1.43
4	2.4	1.42
6 or more	2.3	1.40
2 Nominally Identical Items per Aeroplane subject to the same loading		
1	3.2	1.54
2	2.8	1.49
3	2.6	1.47
4	2.5	1.45
6 or more	2.4	1.44
3 Items		
1	3.4	1.57
2	2.9	1.51
3	2.7	1.49
4	2.6	1.48
6 or more	2.5	1.47
4 or 5 Items		
1	3.5	1.60
2	3.0	1.54
3	2.8	1.52
4	2.7	1.50
6 or more	2.6	1.49

* TABLE 1 OF LEAFLET 201/3

TABLE 2* GUIDANCE ON THE FACTORS BY WHICH THE TEST ENDURANCE OF METAL DETAILS SHOULD BE DIVIDED TO ALLOW FOR UNCERTAINTIES IN ADJUSTING THE RESULT TO A NEW LOADING SEVERITY.

Source of Uncertainty			Factor Recommended
S - N Curve shape	Test severity too high	Conservatively steep curve	1.0
		Other curve	3.0**
	Test severity too low	Conservatively shallow curve	1.0
		Other curve	3.0**
Miner's Rule	Acceptable allowances made for residual stresses (elastic interaction accounted for by test)		1.0
	No allowance made for residual stresses	Test severeity too high	3.0**
		Test severity too low	1.0

* TABLE 2 OF LEAFLET 201/3

** These factors are for guidance, and supporting evidence must be presented if lower factors are to be used. (A separate allowance must be made for the effects of scatter)

TABLE 3* SUMMARY OF FACTORS TO BE USED WHEN DERIVING CRACK GROWTH CURVES AND ESTIMATING INSPECTION INTERVALS

1. DERIVATION OF CRACK GROWTH CURVES

No.	Leaflet Ref.	Reason for factor	Reduction factor
1	Para 5.1	Absence of stress measurement	2.0
2	Para 5.1	Use of approximate correction for stress intensity factor	2.0
3	Para 6.1	Scatter in crack propagation rate	Fatigue life factor from Table 201/3.1
4	Para 7.1	Uncertainties in Service Loading.	1.5

2. ESTIMATION OF INSPECTION INTERVALS

No.	Leaflet Ref.	Reason for factor	Reduction factor
1	Para 11.3	With simulation model	Use for full modelling incorporating uncertainties in threshold time, damage growth and reliability data. Interval as determined.
2	Para 11.4	Without simulation model	Divide growth period from PD=0.5 by 2.0 and then by factor to allow for scatter (para 6) to obtain inspection interval.

* TABLE 1 OF LEAFLET 201/4

ANNEX 1

ITINERARY AND MAIN CONTACTS

Thursday 3 June	IPTN, Bandung, Indonesia. Dr Marjono
Monday 7 June to Friday 11 June	23rd ICAF Conference and 17th ICAF Symposium, Stockholm
Monday 14 June and Tuesday 15 June	DRA, Farnborough, UK Dr A Cardrick

ANNEX 2

ICAF MEETING SCHEDULE

23rd ICAF CONFERENCE PROGRAM
NATIONAL REVIEWS

SUNDAY JUNE 6, 1993

17:00 - 20:00 Conference Registration
19:00 - 20:00 Welcome Reception

MONDAY JUNE 7, 1993

08:00	Registration	
09:00	Opening and welcome	A.F. Blom
	Chairman: R.M. Bader	
09:10	ICAF Secretary's Review	O. Buxbaum
09:20	The Netherlands	J.B De Jonge
10:00	Coffee break	
	Chairman: A. Kobayashi	
10:30	United Kingdom	P.Poole
11:30	Switzerland	A. Jordi
12:00	Lunch (included in registration fee)	
	Chairman: D.L. Simpson	
13:30	Italy	A. Salvetti
14:15	Sweden	A.F. Blom
15:00	Coffee break	
	Chairman: A. Berkovitz	
15:30	Belgium	M. Meurrens
15:50	Germany	O. Buxbaum
16:50	Closure	
18:30	Delegates' dinner	

TUESDAY JUNE 8, 1993.

	Chairman: M. Meurrens	
09:00	USA	R.M. Bader
10:00	Coffee break	
	Chairman: P.Poole	
10:30	Israel	A. Berkovitz
11:00	France	J. Rouchon
12:00	Lunch	
	Chairman: O. Buxbaum	
13;30	Canada	D.L. Simpson
14;15	Japan	A. Kobayashi
15:00	Coffee break	
	Chairman: A.F. Blom	
15:30	Australia	J.M. Grandage
16;15	Closure	

17th ICAF SYMPOSIUM PROGRAM

TUESDAY JUNE 8, 1993

17:00 - 20:00 Symposium registration
19:00 - 20:00 Welcome reception sponsored by Fatigue Technology Inc.

WEDNESDAY JUNE 9, 1993

08:00		Registration
08:45		Opening and welcome by A.F. Blom

FORMALLY PRESENTED PAPERS

09:00		14th Plantema Memorial Lecture: DAMAGE TOLERANCE - FACTS AND FICTION U.G. Goranson, Manager, Structural Damage Technology and Aging Fleet Programs, Boeing Commercial Airplane Group, Seattle, USA.
10:00		Coffee break
		Chairman: J.B. de Jonge
10:30	1.1	MODERN FIGHTER LOADS DERIVATION IN VIEW OF A FULL SCALE DURABILITY AND DAMAGE TOLERANCE TEST C. Perron, Bombardier/Canadair, Canada
11:00	1.2	A METHOD FOR FATIGUE LIFE EVALUATION OF STRUCTURES SUBJECTED TO VIBRATIONS M. Villarboito, T. Giacobbe, G. Cigna, Alenia, Italy
11:30	1.3	LIMITS OF REPRESENTATIVE ACOUSTIC FATIGUE PREDICTION - A NEGATIVE EXPERIENCE K. Konig, Deutshe Airbus GmbH, Germany
12:00		Lunch (included in registration fee)
		Chairman: A. Berkovitz
13:30	1.4	FATIGUE LOADS ON AERONAUTICAL COMPONENTS: VALIDATION OF LIFE PREDICTION FROM REPRESENTATIVE TESTING ON ALUMINIUM ALLOY SPECIMENS M. Chaudonneret, ONERA, France
14:00	1.5	CALCULATION METHOD FOR PREDICTING THE FATIGUE LIFE OF RIVETED JOINTS J.J. Homan, A.A. Jongebreur, Fokker, the Netherlands
14:30	1.6	FRETTING FATIGUE IN AIRCRAFT JOINTS D.W. Hoeppner, S. Adibnazari, University of Utah, USA
15:00		Coffee break

Chairman: A. Jordi

15:30 1.7 RESIDUAL STRESS DISTRIBUTION AROUND COLD EXPANDED HOLES
T. Ozdemir, Open University, R. Cook, Defence Research Agency, L. Edwards, Open University, UK.

16:00 1.8 DYNAMIC CRACK CURVING AT TEAR STRAP WITH SIMULATED MSD
A. Shimamoto, M. Kosai, A.S. Kobayashi, University of Washington, and P. Tan, FAA, USA

16:30 Poster session

18:00 Closure

19:30 - 21:30 Buffet in the City Hall with guided tour of the building, sponsored by the Stockholm City Council

THURSDAY JUNE 10, 1993

Chairman: J. Rouchon

08:30 2.1 PRACTICAL EVALUATION OF CRACK DETECTION CAPABILITY FOR VISUAL INSPECTION IN JAPAN
S. Endoh, Ministry of Transport, H. Tomita, ICAD, H. Asada and T. Sotozaki, National Aerospace Laboratory, Japan

09:00 2.2 INFLUENCE OF INITIAL DEFECT CONDITIONS ON STRUCTURAL FATIGUE IN RAAF AIRCRAFT
G. Clark, S.A. Barter, N.T. Goldsmith, DSTO, Aeronautical Research Laboratory, Australia

09:30 2.3 PREDICTING THE INFLUENCE OF INITIAL MATERIAL QUALITY ON FATIGUE LIFE
A.J. Hinkle, A.F. Grandt Jr., E.N. Forsyth, ALCOA Centre, USA

10;00 Coffee break

Chairman: A.F. Blom

10:30 2.4 PROBABILISTIC DURABILITY ANALYSIS METHODOLOGY FOR METALLIC AIRFRAMES
S.D. Manning, Lockheed/Fort Worth, J.N. Yang, University of California, USA

11:00 2.5 A MECHANISTICALLY BASED PROBABILITY APPROACH FOR PREDICTING CORROSION AND CORROSION FATIGUE LIFE
R.P. Wei, D.G. Harlow, Lehigh University, USA

11:30 2.6 CF-5 UPPER WING SKIN COMPRESSIVE INDUCED FATIGUE CRACKING: A CASE STUDY
M.D. Raizenne, D.L. Simpson, National Research Council, M. Zgela, G. Bateman, National Defence Headquarters, Canada

12:00 Lunch

Chairman: D.L. Simpson

13:30 2.7 RELIABLE STRESS AND FRACTURE MECHANICS ANALYSIS OF COMPLEX AIRCRAFT
B. Anderson, U. Falk, Aeronautical Research Institute of Sweden

14:00 2.8 FATIGUE GROWTH OF MULTIPLE CRACKS EMANATING FROM A ROW OF FASTENER HOLES AND ITS EFFECT ON RESIDUAL STRENGTH OF STIFFENED PANELS
J.H. Park, R. Singh, R. Jones, S.N. Atluri, Georgia Institute of Technology, P. Tan, FAA, USA

14:30 2.9 A FRACTURE CRITERION FOR WIDESPREAD CRACKING IN THIN SHEET MATERIALS
D.S. Dawicke, J.C. Newman Jnr., C.A. Bigelow, NASA Langley Research Centre, USA

15:00 Coffee break

Chairman: A. Salvetti

15:30 2.10 AN ANALYTICAL APPROACH TO MULT-SITE DAMAGE
A. Nathan, A. Brot, Israel Aircraft Industries, Israel

16:00 2.11 DAMAGE TOLERANCE OF FUSELAGE PANELS WITH WIDESPREAD FATIGUE DAMAGE
P. Tong, Hong Kong University of Science and Technology, R. Grief, US Department of Transportation, L. Chen, D.Y. Jeong, Tufts University, USA

16:30 Poster session

17:30 Closure

18:00 - 19:00 Visit to Vasa Museum

19:15 - 23:15 Dinner cruise in the Stockholm Archipelago

FRIDAY JUNE 11, 1993

Chairman: R.M. Bader

08:30 3.1 OPTIMUM AIRCRAFT STRUCTURAL DESIGN AND VERIFICATION FOR USERS
S. Kobayashi, Japan Airlines, Japan

09:00 3.2 THE EVOLUTION OF THE BAE HAWK DESIGN AND STRUCTURAL CLEARANCE
J. O'Hara, British Aerospace, UK

09:30 3.3 FULL-SCALE STRUCTURAL TESTING AS A TOOL IN MANAGING AGING AIRCRAFT
K.E. Brown, Lockheed Aeronautical Systems Co., USA

10:00 Coffee break

Chairman: O. Buxbaum

10:30 3.4 AEROSPATIALE PROBABILISTIC METHOD APPLIED TO AIRCRAFT MAINTENANCE
R. Boetsch, J-Y. Beaufils, Aerospatials, France

11:00	3.5	DESIGN PARAMETERS AND SCATTER IN THE AIRCRAFT SIZING PROCESS - ASPECTS OF DURABILITY AND DAMAGE TOLERANCE H. Ansell, T. Johansson, Saab Military Aircraft, Sweden
11:30	3.6	ASSESSMENT OF STRUCTURAL RELIABILITY DERIVED FROM DURABILITY TESTING J.W. Linclon, Wright - Patterson AFB, USA
12:00		Lunch
		Chirman: P. Poole
13:30	3.7	APPLICATION OF FULL SCALE TEST RESULTS TO STRUCTURAL MODIFICATION M. Gottier, GRD, Switzerland, A. Kuo, S. McCord, McDonnell Douglas Aircraft Co., USA
14:00	3.8	RESULTS OF FATIGUE AND DAMAGE TOLERANCE TEST ON A FOKKER 50 FORWARD FUSELAGE N.J. Fraterman, P.H. Kummel, Fokker, the Netherlands
14:30	3.9	CORRELATION OF T-37B SLEP DAMAGE TOLERANCE ANALYSES WITH FULL-SCALE AND COUPON FATIGUE TEST RESULTS J.W. Cardinal, D.H. Wieland, Southwest Research Institute, USA
15:00		Coffee break
		Chairman: J.M. Grandage
15:30	3.10	CORRELATION OF MATERIAL PROPERTIES AND DAMAGE TOLERANCE BEHAVIOUR OF COMPOSITE STRUCTURES J.J. Gerharz, A. Schopfel, H. Huth, LBF, Germany
16:00	3.11	FATIGUE TEST FOR CFRP HORIZONTAL STABILISER J. Takagi, Fuji Heavy Industries, Japan, J. Berner, Boeing Commercial Airplane Group, USA, M. Hirahara, Japan Aircraft Development Corporation, Japan

ANNEX 3

PARTICIPANTS AT ICAF, STOCKHOLM, 1993

Country	No. of participants
The Netherlands	11
United Kingdom	19
Sweden	32
Belgium	1
Switzerland	7
Germany	20
France	15
Italy	8
Australia	6
United States	36
Canada	7
Israel	6
Japan	9
Finland	8
Other	12
Total	**197**

ANNEX 4

23RD ICAF CONFERENCE: CONTENTS OF NATIONAL REVIEWS

Review number	Country
1	The Netherlands
2	United Kingdom
3	Sweden
4	Belgium
5	Switzerland
6	Germany
7	France
8	Italy
9	Australia
10	United States
11	Canada
12	Israel
13	Japan

NLR TECHNICAL PUBLICATION

TP 93109 L

REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE NETHERLANDS DURING THE PERIOD MARCH 1991 - MARCH 1993

by

J.B. de Jonge

This review will be presented at the 23rd ICAF Conference, Stockholm, June 7-8, 1993.

Division: Structures and Materials
Prepared: JBdJ/ 15/4
Approved: JBdJ/ 15/4

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REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN THE UNITED KINGDOM DURING THE PERIOD MAY 1991 TO APRIL 1993

compiled by

P.Poole

Materials and Structures Department
Defence Research Agency
Farnborough, Hampshire, UK.

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FLYGTEKNISKA FÖRSÖKSANSTALTEN
The Aeronautical Research
Institute of Sweden
Structures Department

A REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN SWEDEN DURING THE PERIOD MAY 1991 TO APRIL 1993

edited by

Anders F. Blom

SUMMARY

This document was prepared for presentation at the 23rd Conference of the International Committee on Aeronautical Fatigue, scheduled to be held in Stockholm, Sweden, on June 7-8, 1993.

A review is given of the work carried out in Sweden in the area of aeronautical fatigue during the period from May 1991 to April 1993. The review includes basic studies of fatigue development in metals and composites, stress analysis and fracture mechanics, studies of crack propagation and residual strength, testing of joints and full-scale structures, and fatigue life predictions.

A reference list of relevant papers issued during the period covered by the review is included.

Sponsoring Agency The Defence Materiel Administration (FMV)
Stockholm

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MINISTERIE VAN VERKEER
EN INFRASTRUCTUUR

BESTUUR VAN DE LUCHTVAART
TECHNISCHE DIRECTIE
RAKETSTRAAT 90 B 1130 BRUSSEL

REVIEW OF FATIGUE WORKS

PERFORMED IN BELGIUM DURING THE PERIOD

MAY 1991 AND MAY1993

REPORT MV 93 - 02

This paper has been prepared
for presentation at the
23th ICAF conference

STOCKHOLM, JUNE 7, 1993

Compiled by Mark Meurrens,
e.a. Ingenieur

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ADRESSES OF INSTITUTES AND COMPANIES HAVING CONTRIBUTED TO THE PRESENT REVIEW

1. SONACA : Parc Industriel
Route National Cinq
B. 6200 GOSSELIES - BELGIUM

2. TECHSPACE AERO : Route de Liers 121
B. 4041 MILMORT (HERSTAL) - BELGIUM

3. M.T.M. - K.U. LEUVEN : De Croylaan 2
B. 3001 LEUVEN - BELGIUM

4. P.M.A. - K.U. LEUVEN : Celestijnenlaan 300 B
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5. WTCM - KETELBOUW : Sint Pietersnieuwstraat 41
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Review of the Swiss Investigations on Aeronautical Fatigue

Period May 1991 to April 1993

by

A. Jordi

FWE-SIG TA 671

Dated 30 April 1993

SWISS FEDERAL AIRCRAFT FACTORY (F+W)
CH-6032 EMMEN

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LBF

Fraunhofer-Institut
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ICAF Conference 1993 – LBF Report S-199

Review of Investigations on Aeronautical Fatigue in the Federal Republic of Germany

by
O. Buxbaum and A. Schöpfel

Period of Review
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REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN FRANCE DURING THE PERIOD 1992 - 1993

Compiled by Jean ROUCHON
CENTRE D'ESSAIS AERONAUTIQUE DE TOULOUSE
31056 TOULOUSE - FRANCE

This review consists in contributions from :

- *ONERA*
- *SNECMA*
- *MESSIER-BUGATTI*
- *AEROSPATIALE*
- *DASSAULT AVIATION*
- *CEAT*

DGA

CENTRE D'ESSAIS AERONAUTIQUE DE TOULOUSE
DIRECTION DES CONSTRUCTIONS AERONAUTIQUES

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A.I.P.A. - ITALIAN ASSOCIATION FOR FATIGUE IN AERONAUTICS

DEPARTMENT OF AEROSPACE ENGINEERING - UNIVERSITY OF PISA

REVIEW OF INVESTIGATIONS ON AERONAUTICAL FATIGUE IN ITALY CARRIED OUT IN THE PERIOD APRIL 1991 - MARCH 1993

by
A. Salvetti and L. Lazzeri

This document was prepared for presentation at the 23rd Conference of the International Committee on Aeronautical Fatigue, scheduled to be held in Stockholm, 7-8 June 1993.

This Review is intended for the personal use of those whom it is addressed

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AR-008-351

DEPARTMENT OF DEFENCE
DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION
AERONAUTICAL RESEARCH LABORATORY

Technical Note 32

A REVIEW OF AUSTRALIAN AND NEW ZEALAND INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1991 TO MARCH 1993

Edited by

J.M. GRANDAGE

SUMMARY

This document was prepared for presentation to the 23rd Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Stockholm, Sweden on June 7 and 8, 1993.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The review summarises fatigue-related research programmes as well as fatigue investigations on specific military and civil aircraft.

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A REVIEW OF RESEARCH ON

AERONAUTICAL FATIGUE IN THE UNITED STATES

1991 - 1993

Compiled by

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FOR PRESENTATION AT THE MEETING

OF THE

INTERNATIONAL COMMITTEE ON AERONAUTICAL FATIGUE

7-11 JUNE 1993

STOCKHOLM, SWEDEN

10.01 INTRODUCTION

Leading government laboratories and aerospace manufacturers were invited to contribute summaries of recent aeronautical fatigue research activities. Their voluntary contributions are compiled here. Inquires should be addressed to the person whose name accompanies each section.

On behalf of the International Committee on Aeronautical Fatigue, the generous contribution of each organization is hereby gratefully acknowledged:

- NASA LANGLEY RESEARCH CENTER
- CHRYSLER TECHNOLOGIES
- NASA LEWIS RESEARCH CENTER
- PURDUE UNIVERSITY
- METAL IMPROVEMENT COMPANY, INC.
- LEHIGH UNIVERSITY
- MCDONNELL DOUGLAS AEROSPACE EAST
- PENNSYLVANIA STATE UNIVERSITY
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- GEORGIA INSTITUTE OF TECHNOLOGY
- SANDIA NATIONAL LABORATORIES
- UNIVERSITY OF WASHINGTON
- NAVAL AIR DEVELOPMENT CENTER
- FEDERAL AVIATION ADMINISTRATION TECHNICAL CENTER
- LOCKHEED AERONAUTICAL SYSTEMS COMPANY
- ESPRIT TECHNOLOGY INC.
- FATIGUE TECHNOLOGY INC.
- AIR FORCE OFFICE OF SCIENTIFIC RESEARCH
- BOEING COMMERCIAL AIRPLANE GROUP
- WRIGHT LABORATORY

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Reference Numbers are indicated as / /. For example, /1/ is Reference 1.

National Research Council Canada | Conseil national de recherches Canada
Institute for Aerospace Research | Institut de recherche aérospatiale
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NRC·CNRC

Review of Canadian Aeronautical Fatigue Work

1991-1993

LTR-ST-1932

D.L. Simpson

10 May 1993

Canada

REVIEW OF CANADIAN AERONAUTICAL FATIGUE WORK 1991-1993

Compiled by
D.L. Simpson

INSTITUTE FOR AEROSPACE RESEARCH
NATIONAL RESEARCH COUNCIL OF CANADA

SUMMARY

This paper provides a review of Canadian work associated with fatigue of aeronautical structures during the period 1991-1993. All aspects of structural technology are covered including full scale tests, loads monitoring, fracture mechanics, composite materials and engine fatigue.

Organization Abbreviations Used in Text

AETE	Aerospace Engineering Test Establishment/DND
BI	Bombardier Inc.
CDSD	Canadair, Defence Systems Division
CF	Canadian Forces
DND	Department of National Defence
DREP	Defence Research Establishment Pacific
IAR	Institute for Aerospace Research
NRC	National Research Council of Canada
PWC	Pratt & Whitney Canada
QETE	Quality Engineering Test Establishment
RMC	Royal Military College of Canada

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APRIL 1993

REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN ISRAEL
April 1991 - March 1993

Compiled by
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SUMMARY

The review summarizes fatigue and fracture mechanics work being pursued at Israel Aircraft Industries, Ltd. (I.A.I.), in the Israel Air Force, Israel Armament Development Authority (RAFAEL), at the Technion - Israel Institute of Technology, and at Tel Aviv University. At I.A.I. concerted effort is being applied in multi-site damage analysis, damage tolerance, and composite materials. The Defense Department Armament Development Authority is represented for the first time with FEM/fracture mechanics work. Work by IAF concerns probability of crack detection, and wing stiffener analysis. University research has, as in the past, been directed towards basic work in fatigue fracture mechanics problems.

Prepared for presentation at the 23rd ICAF Conference
Stockholm, 7 and 8 June, 1993.

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A REVIEW OF AERONAUTICAL FATIGUE INVESTIGATIONS IN JAPAN DURING 1991–1993

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FOR PRESENTATION AT THE 23RD CONFERENCE
OF
THE INTERNATIONAL COMMITTEE ON AERONAUTICAL FATIGUE

STOCKHOLM, SWEDEN, JUNE 8, 1993

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17th ICAF SYMPOSIUM: AUTHORS, TITLES AND ABSTRACTS OF PRESENTED PAPERS

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1.1	A.Berkovitz, D. Fang	An empirical design model for fatigue damage on the basis of acoustic emission measurements
1.2	A. Shimamoto, M. Kosai, A.S. Kobayashi, P. Tan	Dynamic crack curving at tear strap with simulated MSD
1.3	K. Koenig	Limits of representative acoustic fatigue prediction
1.4	C. Bleuzen, M. Chaudonneret, J. Flavenot, N. Ranganathan, M. Robert	Validation of life prediction models from representative testings on aluminium alloy specimens
1.5	J.J. Homan, A.A. Jongebreur	Calculation method for predicting the fatigue life of riveted joints
1.6	D.W. Hoeppner, S. Adibnazari	Fretting fatigue in aircraft joints
1.7	A.T. Ozdemir, R. Cook, L. Edwards	Residual stress distributions around cold expanded holes
2.1	S. Endoh, H. Tomita, H. Asada, T. Sotozaki	Practical evaluation of crack detection capability for visual inspection in Japan
2.2	S.A. Barter, G. Clark, N.T. Goldsmith	Influence of initial defect conditions on structural fatigue in RAAF aircraft
2.3	A.J. Hinkle, A.F. Grandt, Jnr	Predicting the influence of initial material quality of fatigue life

Paper No.	Author(s)	Title
2.4	S.D. Manning, J.N. Yang	Probabilistic durability analysis methodology for metallic airframes
2.5	R.P. Wei, D.G. Harlow	A. mechanistically based probability approach for predicting corrosion and corrosion fatigue life
2.6	M.D. Razienne, D.L. Simpson, M.B. Zgela, G. Bateman	CF116 upper wing skin compression induced fatigue cracking - a case study.
2.7	B. Anderson, U. Falk	Reliable stress and fracture mechanics analysis of complex aircraft components
2.8	J.H. Park, R. Singh, R. Jones, S.N. Atluri, P. Tan	Fatigue growth of multiple cracks emanating from a row of fastener holes and its effect on residual strength of stiffened panels
2.9	J.C. Newman, Jnr., D.S. Dawicke, M.A, Sutton, D.A. Bigelow	A fracture criterion for widespread cracking in thin sheet aluminium alloys
2.10	A. Nathan, B. Brot	An analytical approach to multi-site damage
2.11	L. Chen, P. Tong, R. Grief, S. Sampath	Damage tolerance of fuselage panels with widespread fatigue cracking
3.1	S. Kobayashi	Optimum aircraft structural design and verification for users
3.2	J. O'Hara	The evaluation of the B.Ae. Hawk and its structural clearance
3.3	K.E. Brown	Full-scale structural testing as a tool in managing aging aircraft
3.4	M. Tisseyre, J.Y. Plantec, J.Y. Beaufils, R. Boetsch	Aerospatiale probabilistic methods applied to aircraft maintenance

Paper No.	Author(s)	Title
3.5	H. Ansell, T. Johansson	Design parameters and variability in the aircraft sizing process-aspects of damage tolerance
3.6	J.W. Lincoln	Assessment of structural reliability derived from durability testing
3.7	M. Gottier, A. Kuo, S. McCord	Application of full-scale test results to structural modification
3.8	N.J. Fraterman, P.H. Kummel	Results of fatigue and damage tolerance test on Fokker 50 forward fuselage
3.9	J.W. Cardinal, D.H. Wieland	Correlation of T-37B SLEP damage tolerance analyses with full-scale and coupon fatigue test results
3.10	J.J. Gerharz, A. Schopfel, H. Huth	Correlation between material properties and damage tolerance behaviour of composite structures
3.11	J. Takaki, J. Kimura, J.K. Berner, M. Hirahara, A. Yahata	Fatigue test for CFRP horizontal stabiliser

14th Plantema Memorial Lecture
DAMAGE TOLERANCE - FACTS AND FICTION
U.G. Goranson, Boeing Commercial Airplane Group, USA

Design, analysis and verification of damage tolerant structures embraces both structural characterisation and damage detection assessments. Methods to determine fatigue performance, crack growth and residual strength of complex details have improved significantly since the introduction of commercial jet transports. Less technology development has occurred on integrating this capability in development of structural inspection program recommendations reflecting the value of normal operator maintenance activities. Damage detection considerations required to achieve a flexible maintenance program without compromising structural safety are addressed in this review.

Paper 1.1

AN EMPIRICAL DESIGN MODEL FOR FATIGUE DAMAGE ON THE BASIS OF ACOUSTIC EMISSION MEASUREMENTS
A. Berkovits, D. Fang, Israel Institute of Technology, Israel

Fatigue tests monitored for acoustic emission were carried out at room temperature on Incoloy 901 material specimens, over a stress-ratio range of - $1 \leq R \leq 2$. Valid AE data were obtained even when the load cycle passed through zero. The AE data permitted specific identification of the various phenomena occurring on the way to final failure. The AE findings were supported by microscopic examination. Based on the experimental data, a preliminary damage-prediction model was formulated. The model fits the data well, but further development is required before the model can be expressed specifically in terms of loading parameters and materials constants.

Paper 1.2

DYNAMIC CRACK CURVING AT TEAR STRAP WITH SIMULATED MSD
A. Shimamoto, M. Kosai, A.S. Kobayashi, University of Washington, USA
P. Tan, US Dept. of Transportation, USA

Full scale rupture tests of pressurised fuselages show that a rapidly propagating crack is arrested by crack curving at a tear strap. The mechanism of such crack arrest is attributed to the large crack flaps which generate a large axial stress ahead of the propagating crack. Such axial stress, which is larger than the hoop stress, will kink the crack path following known crack kinking and branching criteria. When the crack propagates along a lap joint, the unsymmetric flap of the upper skin generates an additional mode II stress intensity factor, K_{II} which enhances crack curving. In spite of the presence of K_{II} and the large axial stress ahead of the propagating crack, however, axial crack extension will continue in the axial direction if the skin is weakened by multiple site damages (MSD).

A hybrid experimental-numerical investigation was undertaken to establish a dynamic crack kinking criterion under mixed mode crack tip deformation. 2024-T3 aluminium alloy, biaxial fracture specimens, 0.76 mm thick, were loaded to failure with the ever-present hoop stress and the differential axial stress due to the unsymmetric flap. Axial weakening due to MSD was simulated by V-grooving the specimen to half of its thickness for a length of 25 mm ahead of the machine notch. Additional isolated regions of MSD were simulated by two V-grooves on the dowstream side of the tear strap. Biaxial loading was applied simultaneously within 10 milliseconds and the crack tip location with time was determined by an array of strain gauges along the predicted crack path.

These biaxial fracture tests were used to generate the crack curving and crack velocity data which was input to a dynamic finite element model of the fracture specimen. The numerical results were used to evaluate a postulated crack kinking criterion along the curved crack front. Excellent agreement was obtained between the predicted and measured crack kinking angle and crack arrest location.

Paper 1.3

LIMITS OF REPRESENTATIVE ACOUSTIC FATIGUE PREDICTION
Klaus Koenig, Deutsche Airbus, Germany

Acoustic fatigue prediction is usually based on computed random response strain together with relevant S-N curves from shaker tests. For new materials or designs PWT tests are preferred. The paper presents results of a systematic study allowing to evaluate the reliability and the limits of such methods of acoustic fatigue prediction for CFRP materials.

Paper 1.4

VALIDATION OF LIFE PREDICTION MODELS FROM REPRESENTATIVE TESTINGS ON ALUMINIUM ALLOY SPECIMENS
C. Bleuzen, CEAT, M. Chaudonneret, ONERA, J. Flavenot, SCETIM, N. Ranganathan, ENSMA, M. Robert, ONERA, France

This paper is devoted to fatigue testings and crack initiation prediction for smooth and notched specimens of 2024 and 7010 alloys, subject to various kinds of loadings (from tension compression up to aeronautical spectra). Three methods of fatigue life prediction were tested, and a comparison with the experimental results shows that a better prediction is obtained when a non-linear damage accumulation law is used.

Paper 1.5

CALCULATION METHOD FOR PREDICTING THE FATIGUE LIFE OF RIVETED JOINTS
J.J. Homan, A.A. Jongebreur, Fokker Aircraft, The Netherlands

This paper presents the calculation method for the fatigue strength of riveted joints, as developed by various engineers at Fokker Aircraft Company.

The method is based on the empirical law that fatigue is caused by varying stresses of constant or variable amplitude, and the similarity concept: equal stress history at the point of initiation gives equal fatigue life.

After a historical review of the development of the method, the method itself is described and compared with test evidence of coupon tests. This is followed by a discussion of the design parameters of a riveted joint, as applied in aircraft structures, and their significance for the fatigue strength. Further attention is paid to the validity of the method and its applicability for real aircraft structures. The paper closes with summarising statements and conclusions.

Paper 1.6

FRETTING FATIGUE IN AIRCRAFT JOINTS
D.W. Hoeppner, S. Adibnazari, University of Utah, USA

This paper presents an overview of the fretting fatigue challenge in the design of new aircraft components and the maintenance of structural integrity of "aging aircraft". It will first review the components of fixed wing and rotary wing aircraft that are known to experience fretting. Damage mechanisms then are reviewed and a discussion of fracture mechanics modeling of the phenomena will be presented. Subsequently, design practices that have proved effective will be presented. The role of the numerous parameters that are known to affect fretting fatigue will be discussed. Future research and development needs to assure the integrity of aircraft joints will be presented.

Paper 1.7

RESIDUAL STRESS DISTRIBUTIONS AROUND COLD EXPANDED HOLES
A.T. Ozdemir, L. Edwards, Open University, UK
R. Cook, Defence Research Agency, UK

This report describes a residual stress measurement programme and an experimental crack growth study of cold expanded holes in 7050-T76 aluminium alloy. Process variables including the degree of cold expansion, the expansion process and the nominal hole diameter, have been studied. Fatigue endurances and crack growth rates have been measured on specimens with holes cold expanded using different process variables and the results are discussed with reference to the residual stress fields present.

Paper 2.1

PRACTICAL EVALUATION OF CRACK DETECTION CAPABILITY FOR VISUAL INSPECTION IN JAPAN
S. Endoh, Ministry of Transport, Japan
H. Tomita, ICAO, Japan
H. Asada, T. Sotozaki, NAL, Japan

The role of visual inspection is important for maintaining and improving the aircraft structural integrity. The Civil Aviation Bureau in Japan organised the investigation team for visual inspection capability consisting of three major operators, four major

manufacturers and the National Aerospace Laboratory. This paper describes the collected field data of cracks deteced by visual inspections during maintenance activities for transport aircraft operated by Japanese airlines, the analysed results and the useful information for the safety and reliability of aircraft structures.

Paper 2.2

INFLUENCE OF INITIAL DEFECT CONDITIONS ON STRUCTURAL FATIGUE IN RAAF AIRCRAFT

S.A. Barter, G. Clark, N.T. Goldsmith, ARL, Australia

The influence of initial defect variability on structural fatigue of military aircraft has been highlighted by investigations into the crash of a training aircraft, and by the analysis of cracking which caused failure in a full-scale fatigue test of a fighter aircraft bulkhead. Both aircraft were designed using a safe-life approach to airworthiness.

The analyses described provide a clear demonstration of the sensitivity of the safe-life approach to initial defect size and condition, and highlight a particular concern that increasing design efficiency and improved fatigue-life enhancing technology could lead to greater sensitivity to metallurgical, machining and in-service damage.

Paper 2.3

PREDICTING THE INFLUENCE OF INITIAL MATERIAL QUALITY ON FATIGUE LIFE

A.J. Hinkle, Alcoa
A.F. Grandt, Jnr., Purdue University, USA

This paper describes a fracture mechanics analysis of the growth of fatigue cracks which initiate at microporosity or other natural inhomogeneities. The size, type, and spatial distribution of initial material inhomogeneities are input to the analysis, and cyclic growth, coalescence, and final fracture resulting from cracks which originate at these material locations is calculated. The goal is to employ statistical descriptions of initial material quality (i.e., micropore distributions) to compute the distribution of fatigue life. It is believed this analysis capability provides the design engineer with an effective tool for determining the influence of material quality on structural performance, and would also aid materials developers in assessing benefits gained by further improvements.

Paper 2.4

PROBABILISTIC DURABILITY ANALYSIS METHODOLOGY FOR METALLIC AIRFRAMES

S. Manning, Lockheed/Fort Worth
J.N. Yang, University of California, USA

Two different probabilistic approaches for the durability analysis of metallic structures are described, compared and demonstrated for a full-scale wing component of a fighter aircraft. One approach uses a single initial flaw size and stochastic crack growth. The second approach uses an equivalent initial flaw size distribution (EIFSD) and deterministic

crack growth. The initial fatigue quality of countersunk fastener holes and the baseline crack growth curve are based on fractographic results for replicate dog-bone specimens. Extent of damage predictions for both approaches are compared and correlated with wing tear-down inspection results. Reasonable correlations are obtained for a wide range of crack sizes. The sensitivity of durability analysis predictions to variations in key analysis variables is investigated.

Paper 2.5

A MECHANISTICALLY BASED PROBABILITY APPROACH FOR PREDICTING CORROSION AND CORROSION FATIGUE LIFE
Robert P. Wei, D.G. Harlow, Lehigh University, USA

A mechanistically based probability approach to life prediction is described. Differences between statistical and mechanistic modeling are addressed, and the need for this approach is demonstrated. The process is illustrated through simplified modeling of pitting and corrosion fatigue of aluminum alloys in aqueous environments, using assumed (but realistic) data and probability density functions (*pdfs*). The ability to provide predictions of life beyond the range of available data and assessments of the significance of each random variable is demonstrated. It is also shown that probability considerations must be integral to the entire process, and cannot be *ex post facto* appendages to mechanistic studies. Confidence levels for the predictions are not addressed and the numerical results herein are indication of trends only.

Paper 2.6

CF116 UPPER WING SKIN COMPRESSION INDUCED FATIGUE CRACKING - A CASE STUDY
M.D. Razienne, D.L. Simpson, IAR
M.B. Zgela, G. Bateman, NDH, Canada

State-of-the-art fracture mechanics codes and durability and damage tolerlance analysis techniques have provided the basis for engineering assessment of clasical metal fatigue problems for areas dominated by tension or fully reversed loading. As long as the cracking mechanism was well understood and the loading fully defined, these tools provided sufficient accuracy to enable proper assessment of the problem and possible course of action.

This is not the case for fatigue cracking resulting from tensile residual stresses due to compressive loading. Lack of detailed information about the strain distributions and the high strain gradients in stress concentrations do not allow confident analytical predictions of crack initiation and growth. It is also difficult to design simple coupon tests which replicate the problem area.

The Department of National Defence (DND) has experienced this typle of problem with its fleet of CF116 aircraft. Fatigue cracks were discovered to the upper wing skin golden triangle, an area heavily loaded in compression during high 'g' manoeuvring. Multi-site cracking was found in approximately 30 percent of the aircraft inspected.

DND's approach to the resolution of this problem has been to oversize the holes to remove the cracks and plastic zone and install interference fit steel bushings and fasteners. With this repair scheme, analysis has shown that the aircraft will be able to meet both static and fatigue requirements from normal operations. Critical to the metallic repair is the degree of interference in the fastener holes. If the interference is not achieved a new compression induced residual stress zone is more likely to occur since the net section has been reduced by material removal. To prevent this residual stress zone from forming, a local boron composite patch has been designed, installation procedures developed, and prototype installations performed on a full scale fatigue test and on an in-service aircraft. Since the structure is capable of meeting static and fatigue requirements with the metallic repair correctly applied, the patch is currently being considered only as a fatigue life 'enhancement' from a certification viewpoint.

This paper describes the CF116 upper wing skin cracking problems, solutions considered and repair design philosophy adopted. Issues such as the crack initiation mechanism and the repair certification process and discussed. Predicted stress reductions due to patching will be compared to actual reductions from strain survey data taken from a full scale test article. The development of an installation process specification will be described as well as results of verification testing. Certification issues and in-service inspection requirements will also be addressed.

Paper 2.7

RELIABLE STRESS AND FRACTURE MECHANICS ANALYSIS OF COMPELX AIRCRAFT COMPONENTS

B. Anderson, U. Falk, FFA, Sweden

A computational scheme for reliable linear elastic stress and fracture mechanics analysis of metal and composite structures is developed. By employing the h - p version of the finite element method and new and more effective numerical solution techniques, very complex 3D problems may be solved with a low and virtually guaranteed maximum error. More realistic large-scale structural analysis, interlaminar stress analysis in composites, analysis of multi-site damage, etc, seems possible using the methods described.

Paper 2.8

FATIGUE GROWTH OF MULTIPLE CRACKS EMANATING FROM A ROW OF FASTENER HOLES & ITS EFFECT ON RESIDUAL STRENGTH OF STIFFENED FUSELAGE PANELS

J.H. Park, R. Singh, R. Jones, S.N. Atluri, Georgia Institute of Technolgy, USA
P. Tan, FAA, USA

This paper deals first with the fatigue growth of multiple cracks, of arbitrary lengths, emanating from a row of fastener holes in a riveted lap joint in a pressurised fuselage. The effects of residual stresses due to rivet misfit, and of plastic deformation near the fastener hole, have been included. A Schwartz-Neumann alternating method which uses the analytical solution for a row of multiple collinear cracks in an infinite sheet (the crack faces being subjected to arbitrary tractions), is developed to analyse this MSD problem on a personal computer. It has been found that, for a range of crack lengths, a phenomenon wherein shorter cracks may grow faster than longer cracks, may exist.

The second topic dealt with in this paper is the effect on residual strength due to the interaction between the MSD (multiple cracks near a row of fastener holes) cracks and a single dominant crack. Various cases are considered: (1) single dominant cracks and MSD cracks in the upper row of fastener holes in a single-bay in a lap-joint, and (2) a two-bay dominant crack with a broken stiffener, with MSD ahead of it. Particular emphasis is placed on the role of plasticity and the quantification of residual strength estimates in the presence of MSD.

Paper 2.9

A FRACTURE CRITERION FOR WINDESPREAD CRACKING IN THIN-SHEET ALUMINIUM ALLOYS

J.C. Newan, Jnr., D.S. Dawicke, M.A. Sutton, C.A. Bigelow, NASA Langley, USA

An elastic-plastic finite-element analysis was used with a critical crack-tip-opening angle (CTOA) fracture criterion to model stable crack growth in thin-sheet 2024-T3 aluminium alloy panels with single and multiple-site damage (MSD) cracks. Comparisons were made between critical angles determined from the analyses and those measured with photographic methods. Calculated load against crack extension and load against crack-tip displacement on single crack specimens agreed well with test data even for large-scale plastic deformations. The analyses were also able to predict the stable tearing behaviour of large lead cracks in the presence of stably tearing MSD cracks. Small MSD cracks significantly reduced the residual strength for large lead cracks.

Paper 2.10

AN ANALYTICAL APPROACH TO MULTI-SITE DAMAGE
A. Nathan, A. Brot, Israel Aircraft Industries, Israel

Multiple-Site-Damage (MSD) has been defined as a group of small cracks that appear in an aircraft structure at about the same time, originating from similar design details located in a common area. After some period of slow crack growth, the MSD cracks may suddenly link-up and can result in a large crack that may threaten the safety of the aircraft. MSD has been implicated as contributing to the infamous Aloha Airlines, Boeing 737 accident of 1988. It contributed to the 1983 failure of a Beoing 747 aft pressure bulkhead, and to widespread cracking of USAF KC-135 and C-5A wings. Multi-Site Damage is one of the main areas of emphasis in the latest rise of interest regarding Aging Aircraft.

An aircraft has many areas which are prone to MSD, most notably the lap joints in pressurised fuselages. The crack growth life of any one crack at a specific hole may be considerably shorter than would be predicted by conventional damage-tolerance analysis which ignores the multiplicity of damage. There are two reasons for the shorter crack growth life. First, the crack growth rate is accelerated due to the influence of the adjacent hole and the adjoining damage. Second, the critical crack size may be significantly shorter due to the premature onset of net-section yield.

This paper presents an analytical model that deals with equal or unequal size cracks at a row of holes. The approach is based on the compounding method which is explained in more detail in the following sections. Loaded and unloaded holes are addressed as well as cracks at both sides of the holes and jst on one side. Coupon test results are presented and compared to the analytical models. P-version finite element analysis using PEGASHYS was performed by ESRD Inc. in St. Louis, Mo and the results were compared to our compounded stress intensity solution. The most dominant parameters which affect the MSD problem are studied with a parametric study presented at the end of this paper.

Paper 2.11

DAMAGE TOLERANCE OF FUSELAGE PANELS WITH WIDESPREAD FATIGUE CRACKING

L. Chan, Tufts University
P. Tong, R. Grief, S. Sampath, US Department of Transportation, USA

Widespread fatigue cracking (WSFC) is a type of multiple cracking which can adversely affect the damage tolerance of an airframe structure to the degree that the structure fails to meet its damage tolerance requirement. Not all multiple cracks can be considered as WSFC. WSFC is more likely to be found in aging airplanes.

In this paper we identify a number of WSFC configurations in stiffened and riveted panels and study the WSFC behaviour by examining the interaction of cracks. The focus is on longitudinal cracks in the fuselage with a large (or larger) main crack and a number of small (or smaller) adjacent cracks in a skin panel.

We use the hybrid finite element method, in conjunction with complex variable theory of elasticity, to provide accurate and efficient solutions to this problem. The anlaysis takes into account the flexibility of rivets and examines the effects of tear straps and frames on the crack arrest capability of fuselage panels with WSFC. Typical results include stress intensity factors at the crack tips, stress concentration factors in the tear straps and frames. The results also include the residual strength of the structure based on the linkup of cracks due to frature or net section yielding of ligaments. A method for calculating the threshold of WSFC will also be presented.

This study produces a better understanding of the interaction between multiple cracks, proposes a means to determine the threshold of WSFC, and provides insight for avoiding WSFC in future design.

Paper 3.1

OPTIMUM AIRCRAFT STRUCTURAL DESIGN AND VERIFICATION FOR USERS

Shinobu Kobayashi, Japan Airlines, Japan

When a manufacturer's structural engineer and an airline's structural engineer look at a same aircraft structure, they will have different feelings. A manufacturer's engineer will be proud of the excellence of his design in terms of competitiveness, weight saving and productivity. An airline's engineer will concern what location is prone to initiate a fatigue or stress corrosion crack, when they need to start inspection and how to repair it. This

paper introduces Japan Airlines in-service structural experiences and addresses the importance of structural design and verification to achieve a higher level of safety and reliability with the cooperation of manufacturers, authorities and airlines.

Paper 3.2

THE EVOLUTION OF THE BAe HAWK AND ITS STRUCTURAL CLEARANCE
J.O'Hara, British Aerospace, UK

Military aircraft are designed to meet fatigue requirements based on the utilisations and configurations envisaged at the design stage. Fatigue damage accruement is sensitive to perturbations in load level and to changes in load spectrum shape. Service usage does not always reflect the design utilisation assumptions, nor may the assumtions made in the structural modelling process always be reflected precisely in practice. Furthermore, designs may evolve over time, sometimes subtly, and great care must be exercised to ensure that structural integrity is maintained. The development of the fatigue life clearances of the BAe Hawk family is discussed, with particular emphasis on the confirmatory testing and Operational Load Measurement (OLM) activities necessary to ensure and maintain fleet aircraft fatigue life clearance.

Paper 3.3

FULL SCALE STRUCTURAL TESTING AS A TOOL IN MANAGING AGING AIRCRAFT
K.E. Brown, Lockheed, USA

A series of full scale structural tests were conducted on the C-130 airplane for use in managing the safe operation and maintenance of these aircraft. Data gathered from these tests are being used in the analytical procedures required for the scheduling of structural inspections and mofidications. A significant by-product of these test programs was the development of a number of structural repairs, a few of which are discussed in this paper.

Paper 3.4

AEROSPATIALE PROBABILISTIC METHODS APPLIED TO AIRCRAFT MAINTENANCE
M. Tisseyre, J.Y. Beaufils, R. Boetsch, Aerospatiale
J.Y. Plantec, Toulouse University, France

The aim of structural maintenance programs is to prevent fatigue damages. Maintenance requirements for fatigue are determined by Damage Tolerance principles. In most cases, the parameters involved to describe the growth of the fatigue damage are determined by using deterministic models. Unfortunately, a high variability exists for all these parameters. Deterministic results must be corrected by empirical scatter factors. Another approach is to use probabilistic models where the parameters are represented by statistical laws or probability distributions. This paper presents the three methods developed at Aerospatiale for the elaboration of maintenance programs.

Paper 3.5

DESIGN PARAMETERS AND VARIABILITY IN THE AIRCRAFT SIZING PROCESS - ASPECTS OF DAMAGE TOLERANCE

Hans Ansell, Thomas Johansson, Saab, Sweden

Several design and random parameters in the aircraft sizing process are expected to cause impact on the ability to achieve required structural life and safety. In a broad view, activities such as mission analysis, external loads anlaysis, internal loads analysis and stress anlaysis must be looked upon with as much attention as the fatigue and fracture mechanics analysis itself.

Deviations in the product model of the SAAB JAS39 Gripen aircraft are studied from a research point of view. Planned deviations from the base model originating from different disciplines of aircraft design are put into the ordinary sizing system producing response variables in terms crack growth lives. Parameters which are severe in terms of damage tolerance are identified, and the relation between design and random parameters are quantified.

Paper 3.6

ASSESSMENT OF STRUCTURAL RELIABILITY DERIVED FROM DURABILITY TESTING

J.W. Lincoln, WPAFB, USA

When the United States Air Force adopted the damage tolerance approach to ensure structural safety of their aircraft, they also adopted a two lifetime durability test to ensure the economic life of the aircraft exceeded the design life. This paper makes an assessment of the adequacy of this test duration based on crack growth from the distribution of initial structural defects observed in the development testing. A Weibull distribution is chosen to represent the uncertainty in the time to the end-of-test crack length. In addition, the usage variability of operational aircraft is represented by a log normal distribution. The change in aircraft mass with time is derived from historical data. From this basic information, the expected number of aircraft losses from an undetected potential failure location is determined.

Paper 3.7

APPLICATION OF FULL SCALE TEST RESULTS TO STRUCTURAL MODIFICATION

M. Gottier, A. Kuo, S. McCord, McDonnell Douglas, USA

Simplified analysis methods based on the crack initiation and growth data of full scale aircraft fatigue tests were developed to facilitate design analyses for structural modification. The methods circumvent the time-consuming step of detailed stress analysis which is a prerequisite for fatigue analysis. To demonstrate the simplified methods, examples are given of the structural modifications of a current production fighter aircraft for a substantially more severe future usage.

Paper 3.8

RESULTS OF FATIGUE AND DAMAGE TOLERANCE TEST ON FOKKER 50 FORWARD FUSELAGE

N.J. Fraterman, P.H. Kummel, Fokker, The Netherlands

The forward part of the fuselage of the Fo50, consisting of cockpit and forward door section has been fatigue tested for 270000 flights. The last 90000 flights of this test included artificial damage at a number of locations.

After the 270000 flights a number of residual strength tests have been performed on the test article with fatigue and/or artificial damage.

The fatigue-spectrum consisted of cabin pressure, fuselage shear, fuselage bending and nose under-carriage loads.

This paper will discuss the test article, test-setup, load spectrum and the results of the fatigue-tests, damage-tolerance tests and residual strength tests.

Paper 3.9

CORRELATION OF T-37B SLEP DAMAGE TOLERANCE ANALYSES WITH FULL-SCALE AND COUPON FATIGUE TEST RESULTS

J.W. Cardinal, D.H. Wieland, Southwest Research Institute, USA

The T-37B durability and damage tolerance analysis effort integrated test and anlaysis results to demonstrate that an 8,000 hour no inspection goal was achieved. Crack growth predictions were validated against coupon spectrum test results and crack growth measurements made during full-scale damage tolerance testing. Unique aspects of analysis correlations to test results are presented. These include the effects of coldworked holes, correlations and differences between full-scale and laboratory coupon test results, determination of continuing damage sites, and fracture mechanics modeling issues. Lessons learned regarding full-scale aircraft testing and the correlation of fracture mechanics analyses and testing are summarised.

Paper 3.10

CORRELATION BETWEEN MATEERIAL PROPERTIES AND DAMAGE TOLERANCE BEHAVIOUR OF COMPOSITE STRUCTURES

J.J. Gerharz, A. Schopfel, H. Huth, LBF, Germany

The improved performance of toughened composites was demonstrated by Interlaminar Fracture Energy (IFE)- and Compression After Impact (CAI)-tests with generic specimens (coupons), and the transfer of these improvements to the composite structure by CAI-, Fatigue After Impact (FAI)- and Delamination Growth (DG)-tests with specimens related to a horizontal tail structure. The work includes development of a special testing technique which was required for the derivation of D curves ($dA/dN \sim G_{total}$), and finite element calculations of strain energy release rate distributions along the delamination front.

Para 3.11

FATIGUE TEST FOR CFRP HORIZONTAL STABILISER

J. Takaki, J. Kimura, Fuji Heavy Industries, Japan
J.K. Berner, Boeing, USA
M. Hirahara, A. Yahata, JADC, Japan

Under a joint civil transport development program by Japan Aircraft Development Corporation (JADC) and the Boeing Commercial Airplane Group, a full scale CFRP test box (Test Box) representing the 7J7 (YXX) horizontal stabiliser was designed, fabricated, and tested. The Test Box was designed by Boeing and fabricated by Fuji Heavy Industries (FHI). The durability and damage tolerance capabilities of the structure were deomonstrated by static and fatigue testing conducted by JADC and FHI at National Aerospace Laboratory.

ANNEX 6

17th ICAF SYMPOSIUM: AUTHORS AND TITLES OF POSTER PAPERS

Item	Author(s)	Title
1	A. Loh, N.Mileshkin, C.A. Patching, L.A. Wood (Australia)	Fatigue behaviour of a glass fibre/epoxy glider wing
2	Y. Beauvais (Canada)	CF116 (CF5) full-scale durability and damage tolerance test (FSDADTT)
3	D. Lamanque, N. Barrett (Canada)	CF-18 fighter FS488 bulkhead post-ECP 45/320 flight strain survey
4	M. Ferahi, S.A. Meguid (Canada)	The effect of short-peening of the fatigue fracture behaviour of advanced airframe alloys
5	Zeng Mingzhong, Qiu Zhongyi (China)	A study on engineering small crack growth rule under both elastic and elasto-plastic conditions
6	X.Xin, E.R. de Los Rios, A. Navarro (UK)	Microstructural sensitive fatigue crack growth in Al-Li 8090
7	J.S. Boabaid, J.R. Yates (UK)	The effect of the grinding condition on short fatigue crack growth in Waspalloy at 19°C
8	J. Foulquier, C. Petiot (France)	Fretting-fatigue behaviour of major helicopter alloys and influence of surface protections.
9	Y. Martin-Siegfried, F. Cahierre (France)	Ground dynamic response of the aircraft
10	J. Frischbier, S. Sikorki, O. Peter (Germany)	Fatigue life and bird strike resistance of a composite fan blade for advanced ducted engines
11	K.Kussmaul, M. von Alberti, U. Eisele, T. Schneider (Germany)	Delamination of fibre-reinforced plastics under cyclic mode II loading: crack growth hours and their statistical evaluation

Item	Author(s)	Title
12	F. Walther, C. Mattheck (Germany)	Local stiffening of shell structures by SKO and CAO - a method to find fatigue resistant and lightweight designs in engineering
13	Li Jianchun, D. Taylor (Ireland)	A new method for modellinlg short crack closure
14	D. Fang, A. Berkovitz (Israel)	Mean stress models for low cycle fatigue of a nickel-base superalloy
15	V. Giavotto, C. Caprile, G.Sala (Italy)	Fatigue and damage mechanics of notched, impacted and jointed thermoplastic composites
16	G.Donzelli, F.Paracchini, P.Filipas (Italy)	Fatigue behaviour of electron beam welded joints for helicopter transmissions
17	J.Andersons, Y.Limonov (Latvia)	Effect of stress ratio on durability of fiber-reinforced composite
18	J.Lameris (The Netherlands) S.Kamil, D.Suryo, A.R.Sudrajat (Indonesia)	Description of the CN-235 fatigue load monitoring programme
19	P.H. de Haan, J.Hannelink, W.P. Hoogerworst, W.F. Pronker (The Netherlands)	The Fokker barrel test program
20	J.D. Yost (USA)	Aircraft fatigue and crack gwoth considering loads by structural component
21	A.D. Paulsson, D.W. Hoeppner (USA)	Corrosion fatigue crack growth rates in alunimium alloys subjected to variable ripple sequences
22	W.R. Elliot (USA)	A neural network application to a low-frequency eddy current nondestructive inspection.

Item	Author(s)	Title
23	L. Reid (USA)	Split sleeve cold expansion as a rework process for previously cold expanded holes.
24	A.F. Liu (USA)	Evaluation of current analytical methods for crack growth in a bolt
25	H.I. Syed, W.P. Winfree, K.E. Cramer (USA)	Thermographic NDE application for quantitative disbond and corrosion detection in aircraft skin
26	K.J. Ludas (USA)	Influence of aging aircraft programs on the MD-11 damage tolerance certification process
27	P.W. Tan, C.A. Bigelow, S.N. Atluri (USA)	Stress intensity factor solutions for surface and corner cracks at countersunk rivet holes.

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This report describes the 1993 ICAF (International Committee on Aeronautical Fatigue) meetings in Stockholm. Associated visits were made to UK to discuss structural airworthiness requirements for military aircraft and to Indonesia to present a seminar on aircraft structural fatigue.

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