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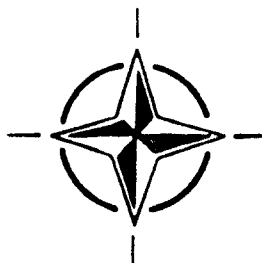
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AGARD CONFERENCE PROCEEDINGS 506

Fatigue Management

(La Gestion de la Fatigue)

*Papers presented at the 72nd Meeting of the AGARD Structures and Materials Panel,
held in Bath, United Kingdom 29th April to 3rd of May 1991.*



NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION STATEMENT A

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Distribution Unlimited

Published December 1991

Distribution and Availability on Back Cover

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North Atlantic Treaty Organization
Organisation du Traité de l'Atlantique Nord

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- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
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Published December 1991

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ISBN 92-835-0642-1



Printed by Specialised Printing Services Limited
40 Chigwell Lane, Loughson, Essex IG10 3TZ

Preface

With the ever-increasing trend towards the retention of aircraft in service much longer than originally planned, coupled with the facts that airframe structures are much more precisely optimised and advanced active control systems are common, it is essential that state-of-the-art fatigue monitoring procedures are used. Information from such systems is vitally important for efficient cost effective fleet management. Such data are also important to procurement authorities when trying to plan aircraft replacements.

The Specialists' Meeting on Fatigue Management was very well supported, both in terms of the number of papers presented and in terms of the very large attendance. Topics addressed during the presentations covered design philosophies, testing, aircraft tracking, control of fatigue consumption rates and education. Both deterministic and probabilistic procedures were discussed.

The meeting concluded with a well-attended discussion period. A summary of the issues and recommendations is provided at the end of the Proceedings.

A.P. Ward
Chairman
Sub-Committee on Fatigue Management

Préface

Etant donné la tendance de plus en plus marquée vers le maintien en service des aéronefs au-delà des dates limites initialement prévues, en plus du fait qu'aujourd'hui les structures des cellules sont optimisées de façon plus précise et que les systèmes à commandes actives se sont banalisés, la mise en oeuvre des dernières procédures de contrôle de la fatigue est désormais indispensable.

Les informations issues de tels systèmes sont d'une importance capitale pour la gestion efficace et rentable de la flotte aérienne. De telles données sont également très utiles aux responsables des approvisionnements lors de la planification de la flotte.

La réunion de spécialistes sur le contrôle de la fatigue a été bien soutenue, tant du point de vue du nombre de communications présentées que celui de l'assistance. Les présentations ont porté sur les philosophies de conception, les essais, le suivi des avions, le contrôle des taux de consommation en fatigue et la formation. Les discussions ont porté aussi bien sur les procédures déterministes que probabilistes.

La réunion a conclu par une table ronde qui a rassemblé un nombre important de participants. Un résumé des questions débattues et des recommandations qui ont été formalisées est donné en annexe au compte-rendu de la réunion.

A.P. Ward
Président du Sous-Comité
sur la Gestion de la Fatigue

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KEYNOTE ADDRESS

STRUCTURAL INTEGRITY IN THE ROYAL AIR FORCE

by

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Military aircraft are now more likely to be retired when their fatigue lives expire, rather than when they become obsolete. As aircraft are kept in service much longer than originally envisaged it becomes more difficult to maintain airworthiness. This presentation gives a brief overview of the RAF's structural integrity organisation, with particular emphasis on fatigue analysis and fatigue management.

Ladies and Gentlemen, Good Morning. I am Air Commodore Davies, one of the six Directors of Support Management in the Royal Air Force. I was particularly pleased to be asked to speak to you today and to be able to add my own welcome, especially to those delegates who have travelled from overseas. Looking at your programme for the week ahead, I think you will find your visit to Bath both interesting and rewarding. In these times of stringent defence budgets it is essential that we NATO allies tailor our research and development in the most cost effective manner. You in AGARD have a vital role to play in achieving this. The subject of your Specialists' Meeting — Aircraft Fatigue — is a typical area of concern which affects us all and it is most certainly an issue in which the RAF is deeply involved. Indeed I feel that it is very appropriate for me to present today's Keynote Address. Within my organisation sits the RAF's aircraft structural integrity policy branch and its head, Wing Commander Cunningham, is a member of the SMP. In addition, I am also responsible for the in-service support of Tornado, an aircraft with which I have been associated for many years. These two aspects of my work have been brought together in presentation which you will hear later on during the Specialists' Meeting. The presentation, entitled 'Fatigue Management in the RAF', will use the Tornado as an example.

Over the next 15 minutes or so I would like to describe how the RAF maintains aircraft structural integrity. There is no doubt that this is a support area which requires very close attention. This is particularly so in Britain at the moment, where the Ministry of Defence is making strenuous efforts to combine authority and accountability under its New Management Strategy; in short we are looking for better value for money. I will not dwell on the routine aircraft maintenance issues, most of you will be familiar with those. Instead I will look at our policies, in particular those for fatigue analysis and fatigue management. With the cost of new combat aircraft inexorably rising it is perhaps not surprising that we tend to keep aircraft in service much longer than we originally intended. These days it is often more cost effective to update an aircraft's avionics rather than to replace the whole weapons platform. Usually therefore, the fleet is withdrawn from service only when the fatigue life expires. The Buccaneer, is a typical example.

Specified and built for the Royal Navy it subsequently entered service with the RAF about 20 years ago. When we finally retire it in a few years time the aircraft will have served for more than 6 times its originally specified fatigue life. It is perhaps ironical that the Buccaneer is due to be withdrawn from service shortly after it flew in combat for the first time with the RAF in the Gulf War recently.

Now about the time that the Buccaneer was entering service, although the timing was pure coincidence, the RAF suffered a spate of fatigue-related accidents to a variety of aircraft types. A study into aircraft structural integrity was therefore commissioned. The study concluded that the day-to-day airworthiness issues were being quite properly managed by the appropriate Engineering Authorities — in fact we still have a similar support system today. However, the study identified the need for a separate organisation to look at the broader aspects of structural integrity. A new branch was subsequently formed in the Ministry of Defence and it currently consists of 6 specialist officers, 3 of whom have an MSc in Aircraft Design from Cranfield Institute of Technology. In broad terms this branch is charged with defining and implementing the RAF's structural integrity policy, but as you can imagine much detailed work lies behind that all-embracing remit. During the week I know Wing Commander Cunningham would be delighted to elaborate on the work of his branch if you are interested. Some may question the need to lay down a formal policy. After all the RAF's accident rate compares favourably with that of its NATO allies and aircraft losses due to fatigue have historically accounted for only 2% of major accidents. I suppose that doesn't sound too bad a record. However, if I put it another way and said that in the last 17 years we have lost 5 aircraft due to structural failure the picture is not as rosy. Apart from the unacceptable risk to life, this represents a significant capital loss. Add to this the costs of corrosion rectification and fatigue remedial packages throughout the life of an aircraft fleet and you can see that maintaining structural integrity is an expensive business. Our current policy, then, has 3 aims:

- | | |
|-----------|--|
| Firstly: | To maintain Flight Safety. |
| Secondly: | To maintain operational capability and maximise aircraft availability. |

Thirdly: To minimise life cycle costs.

So how do we meet those aims in practice? Clearly there are many threats to structural integrity — corrosion, stress corrosion, the environment and normal operating hazards are typical examples. However, it is mainly fatigue that we are concerned with here today. I would venture to suggest that fatigue remains the most significant threat to aircraft structural integrity and will continue to be so for the foreseeable future. In addition, structural maintenance — much of it fatigue-related — represents a large proportion of an aircraft's life cycle costs. I would therefore like to describe how, over the past 20 years or so, our fatigue monitoring policy has evolved in response to the threat.

It was back in the 1950s, in the wake of the Comet disasters that the Royal Aerospace Establishment, Farnborough introduced a system of fatigue monitoring that eventually became known as the safe life philosophy. Most of our combat aircraft today are still designed to this principle and of course it is combat aircraft that give us the most fatigue problems. The fatigue life of the aircraft is demonstrated by the factored results of a full-scale fatigue test specimen. In service, fatigue consumption is derived from a fatigue meter and we have a manual recording system based on the Flying Log and Fatigue Data Sheet, or Form 725. The F725 records sortie details such as start up and shutdown masses, fuel received and stores expended. After the sortie the fatigue meter readings are entered on the same form. The final step is to turn this data into a Fatigue Index, or FI, of damage by means of the fatigue formula. With this rudimentary monitoring system we thought we had pretty well taken care of fatigue. Needless to say, there is no room for complacency in aviation and it was not long before we were caught out with another series of fatigue-related accidents in the early 1970s. Hindsight is a wonderful thing and it was readily apparent that these accidents had a common thread. Aircraft were being used in service in different ways to those assumed by the designer. For example, we lost a Harrier because a flap drive failed and the resulting asymmetry put the aircraft into an uncontrollable roll. After the Harrier entered service its pilots found that combat manoeuvrability could be improved dramatically by using the flaps. There was nothing to stop them doing this as limiting speeds were not being exceeded. However, no-one had told the designer, who quite naturally assumed that flaps would be used only for take-off and landing. The flap drive system was therefore not designed for the arduous loading cycle it was experiencing and the result was a classic fatigue failure.

Obviously designers have to make certain assumptions about how an aircraft will be used in service, but the fatigue meter — which is after all only a counting accelerometer — was not providing sufficient feedback on actual usage. The solution adopted was the production of this, the Statement of Operating Intent, or SOI. This document, which is written by my structural integrity branch, is now a formal Air Publication and one is produced for every aircraft type. It provides the Design Authority with detailed information on how the aircraft is being used, including all the representative sortie profiles. Every year the document is subject to a formal review and naturally this involves us in a close dialogue with our aircrew colleagues.

So 15 years ago, with the SOI and details from the F725 providing a measure of fatigue damage on a sortie-by-sortie basis, we had every confidence that fatigue was nicely under control. However, in 1980 we were caught out once again when in the space of 7 months we lost 2 Buccaneers through

unrelated wing structural failures. You might reasonably ask why the full-scale fatigue test did not predict these failures. Well again with the benefit of hindsight there was a clear link between the 2 accidents. The fatigue test load spectrum included only symmetric loads on the structure. This was realistic for the Buccaneer in maritime use. However, in RAF service the aircraft was being used in a terrain-hugging overland role which involved aggressive flying, including combined rolling and pitching manoeuvres. Following the second Buccaneer crash the Ministry of Defence was criticised for failing to exploit available technology in fatigue monitoring systems. In retrospect, the criticism was well founded. In both accidents the fatigue failure occurred because the test loads applied were not representative of in-service usage.

I'm pleased to say that we have made significant progress over the past 10 years. Having established that test loads were unrepresentative, it was a logical step to measure actual loads on in-service aircraft and this we have been doing for some time through our Operational Loads Measurement, or OLM, programmes. Data from this sort of equipment can be used in 3 ways. Firstly, it can be used to confirm the assumptions made by the designer. Secondly, when substantial discrepancies are apparent we can feed it back into the full-scale fatigue test so that we apply actual loads that have been measured in flight. Thirdly, we can use the data to modify the aircraft's fatigue formula in the light of actual usage. Of course such programmes are not cheap, costing many £ millions to set up and run. Nevertheless, at the moment OLM programmes seem to offer the most cost-effective method of improving our overall fatigue monitoring and hence our confidence in structural integrity. The RAF is firmly committed to OLM and it is now policy to have such a programme on all new types entering service — including gliders. Incidentally, my structural integrity branch writes the detailed Staff Requirements for OLM and also acts as sponsor for the programmes.

So having given you a quick resume of how we arrived at where we are today, how does our system work in practice? For example, what do we do with all the fatigue data that we gather on the F725s? For most of our aircraft, the actual computation of Fatigue Index is carried out at the Maintenance Analysis and Computing Division, or MACD, at RAF Swanton Morley in Norfolk. My policy branch works very closely with MACD in this area and you will hear of this in more detail during the Specialists' Meeting. Over the past few years we have made excellent progress in making fatigue data more easily digestible. MACD now produces a range of colour graphical outputs which are much more user friendly than the pages of tabulated data they replaced. Not only do we present historical data, but we try to produce realistic projections to enable fleet managers to anticipate future problems.

However, in the field of aircraft structural integrity other specialist agencies are involved. As I mentioned earlier, the day-to-day airworthiness issues are handled by the relevant aircraft Engineering Authority, now re-termed Support Authority under our New Management Strategy. However, the broader policy aspects must involve the aircraft Design Authorities, the Royal Aerospace Establishment at Farnborough, the aircraft Project Offices in the Ministry of Defence (Procurement Executive) and the Air Staff (who need to be kept aware of any operational implications) as well as MACD and my own policy branch. Quite clearly these agencies need to be drawn together and this we do through the medium of the Structural Integrity Working

Group, or SIWG; there is one for each aircraft type. The SIWGs meet at least twice per year (more frequently for the troublesome types) and adopt a common agenda, which covers in detail all aspects of aircraft structural integrity. Topics considered include corrosion, stress corrosion and the effects of any recent modifications on fatigue life or static strength. Furthermore, the SIWGs ensure that the SOIs are kept up to date and they monitor closely the results of OLM programmes. SIWGs also task the Design Authorities to consider the effect of any change in aircraft usage on the fatigue formula. Each SIWG produces an annual report, which summarises the main areas of structural integrity activity on its aircraft. These reports, which are widely circulated, provide our senior officers with an assurance that aircraft structural integrity is being properly managed.

These days, FI consumption is a major topic for SIWGs to consider. Over the years we have come to realise that fatigue is a financial resource and, just like any other resource, it must be budgeted to ensure its most cost-effective use. It is not something that can be left until the aircraft approaches the end of its life, by then it is far too late. We must start as soon as the aircraft enters service and monitor closely fatigue consumption throughout its life. Of course the problem becomes even more acute when our Air Staff colleagues decide to run on an aircraft fleet beyond its originally specified out-of-service date. The modern MACD outputs that I described a few moments ago are proving to be excellent aids for the SIWGs. Using this sort of presentation we can quickly assess which are the most fatigue-damaging sorties and how often they are flown. We can then offer our aircrew colleagues some soundly based advice. Indeed we place much emphasis on aircrew education and with some success. It is worth remembering that fatigue is also an aircrew resource. It is the aircrew that use fatigue, govern its rate of usage and hence determine how long an aircraft will remain in service. Frequent dialogue between engineers and operators is therefore vital; this is an area that we ignore at our peril.

I would now like to turn my thoughts to the future. There is no doubt that in recent years the major advances in combat aircraft have been driven by the dramatic improvement in on-board computation. For example, the Tornado computer's power has increased by a factor of 8 since the aircraft entered service. The way ahead for fatigue monitoring systems is clearly to make use of these ADP facilities. Our first such system is already taking shape — the Fatigue Monitoring and Computing System, or FMCS — for the Harrier GR7 and we hope EFA will have a comparable system. FMCS will give us an accurate, almost real-time output of FI consumption on a sortie-by-sorte basis using its on-board computer and hard-wired strain gauge system. Using electronic data transfer between operating units and MACD we will be able to present users with processed fatigue information much more quickly than hitherto. Although the in-service date for FMCS is still some way off we are already working hard to formulate plans for its introduction. With the mass of data that can be extracted from such fatigue monitoring systems it is important to lay down a firm policy for its control and dissemination. Too much information in the wrong hands could be counter productive.

My second point about the future is one which I hope will give you food for thought: access to the structure. For combat aircraft, access to systems is addressed, with varying degrees of success, during the design phase; however, structural access is often difficult and frequently expensive. For example, after the second Buccaneer crash we scrapped one third of the fleet because the repair scheme was so difficult and expensive; in addition, we spent around £100 million repairing the remaining aircraft. With structural maintenance accounting for about 20% of an aircraft's overall life cycle costs I believe this is an area which merits further study. There are potentially some very large savings. In theory it is only damage tolerant aircraft which require inspection to assure structural integrity. However, experience has shown that it is naive to expect a fail safe design to last its service life with no significant fatigue defects. Tornado had a whole series of fatigue remedial packages which began very early in its service life and each of which required massive stripdown to reach the relevant items of structure. I am no more confident about future generations of aircraft. Despite closer attention to R & M aspects in weapons system specifications, basic aircraft mass is still the performance driver and performance is paramount. Even with the advent of advanced materials and CAD techniques the pressure is still on to keep the structural mass to a minimum. Unpredicted, in-service fatigue failures are therefore here to stay and undoubtedly our successors will still be in the business of unplanned replacement of structural components. My question to you is this: we spend £ millions on advanced fatigue monitoring systems to aid cost-effective fatigue management but could we be equally or more cost effective if, during the aircraft design phase, we considered structural access to potentially fatigue-critical areas?

In just a few minutes it is very difficult to examine any subject in depth, particularly when it has as many facets as aircraft structural integrity. Even using fatigue as my theme I have been able only to skate over the surface. No doubt over the next two days you will be covering the ground in much greater detail and I very much look forward to hearing some of the presentations. Nevertheless, I hope I have been able to give you a flavour of how the RAF considers structural integrity, in particular from a fatigue management point of view. The subject is just as important to us as it is to the fare-playing airline passenger, although perhaps somewhat less emotive! Over the years our organisation has evolved to meet the fatigue threat. It would be true to say that we have made some mistakes and that in the past our system only reacted to problems. However, with a specialist policy branch dedicated to aircraft structural integrity now firmly embedded in the organisation, I believe we are well placed to meet the future with confidence.

Ladies and Gentlemen, I very much appreciate being given this opportunity to speak to you this morning. Once again, on behalf of all the British panel members, may I welcome you to England and Bath in particular, a delightful city to visit in the spring. I hope the British weather is kind to you this week and that you enjoy your stay. Most importantly, I hope you find the SMP a professional and stimulating forum in which to participate. I am sure you will. Ladies and Gentlemen, thank you.

"THE DEVELOPMENT OF FATIGUE MANAGEMENT REQUIREMENTS AND TECHNIQUES"

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SUMMARY

Fatigue management requirements and techniques have evolved over a period of 40 years or more. This paper provides an overview of these developments.

An historical summary is presented covering the introduction of the different monitoring techniques ranging from the simple V-g recorder through to the multi-channel systems with on-board processing that now exist.

The paper concludes with a summary of the main requirements for modern systems and then identifies a number of key issues that should be addressed during the course of the Specialists' Meeting.

1. INTRODUCTION

Fatigue management requirements and techniques are fundamental to the efficient utilisation of modern fleets of military aircraft and to the maintenance of structural integrity. There have been significant advances made in the techniques available for fatigue monitoring and a number of these are discussed by other authors at this Specialists' Meeting.

This paper provides a brief historical review of developments in fatigue monitoring and then identifies a number of matters that should be addressed when designing new systems.

2. PRE-1960

Over 40 years ago there were no requirements for the collection of data for fatigue monitoring and management purposes. In the UK studies were underway, using V-g recorders, to collect data for design purposes for both military and civil aircraft (refs. 1 and

2). This information was of some assistance in making fatigue predictions but there was also a recognition that better information was necessary. Based on the knowledge that the wing root tended to be the critical area and for those subsonic aircraft with simple wing planforms, that wing loading and cg acceleration were readily related, the counting accelerometer was developed (ref. 3). This device was used in conjunction with height, speed, time and weight data in a major programme commencing in 1951 to gather turbulence intensity data on a world wide basis (ref. 4).

The widespread use of the counting accelerometer, or "fatigue meter", followed. From 1954 onwards most UK military aircraft have been fitted with the device as have many UK civil aircraft.

In the USA parallel activities were underway. The Service Loads Recording Programme (SLRP), which commenced in 1936 with the objective of collecting design information, had its bias changed in 1958 towards the collection of fatigue usage data (ref. 4). Also, in 1954, Skopinski et al published their report on the calibration of strain gauge installations for use in load measurement programmes (ref. 5), these techniques in general still being relevant.

In this same period special studies were initiated to gather fatigue relevant information for different types of flying (ref. 6, for example) for those parts of the structure for which cg acceleration had little relevance. Such data provided damage rates which could be used for monitoring purposes if the mixture of types of flying was known.

An alternative approach was to use Role Severity Factors. These were theoretical damage rates for different roles (missions) relative to the design role. The damage rates were estimated for each type of sortie to be flown by the aircraft using assumed sortie profiles and atmospheric turbulence data, possibly supplemented by assumptions of typical manoeuvres that would occur in the sorties.

3. 1960 - 1970

The simplifying assumptions associated with the use of the fatigue meter, and the fact that loads on many parts of the structure could not be related in any way to cg acceleration, led to the need to develop improved methods of fatigue monitoring. To record more information on a fleetwide basis was a major task because of airborne recording and subsequent ground processing demands. The USAF, for example, did record more data in the SLRP on a limited number of development aircraft. These studies enabled bi-variant tables of fatigue relevant parameters to be produced, from which fatigue damage rates could be predicted. By 1970 the Life History Recording Programme was introduced (ref. 4) with the objective of using statistical techniques to establish whether changes were occurring relative to the load spectra generated from the SLRP.

In the same period (1960 - 1970) special fatigue monitoring devices were being devised and evaluated (refs. 7, 8 and 9 for example). Such devices were intended to be installed at the fatigue relevant locations on the airframe to measure directly local strain histories which could then be converted to fatigue damage.

In 1968 the NATO Military Committee suggested that "Fatigue load monitoring of tactical aircraft" was a problem that required studying. A task was placed on the SMP who subsequently presented agreed conclusions and recommendations (ref. 10). The recommendations were that there should be efforts

- (i) to establish statistical relationships between movement parameters and structural loads
- (ii) to develop a simple strain recording system
- (iii) to establish techniques for fatigue life monitoring in those NATO countries where established procedures did not exist

4. POST 1970

A considerable explosion of activity has taken place during the last twenty years. In addition to the use of manoeuvre data to produce bi-variate tables of fatigue relevant parameters similar data are being used to develop parametric methods of fatigue monitoring. In this case aircraft response data are used to determine predictions of loads from parametric equations that have been derived using regression techniques (refs. 11, 12 and 13 for example).

In the UK, for each aircraft type, there has been a commitment to the use of Operational Load Measurement programmes on a limited number of aircraft in the fleet to supplement data obtained from simple fleetwide monitoring systems such as fatigue meters. The Tornado SUMS programme is such an example, this being reported at the last AGARD meeting where fatigue monitoring was discussed (Sienna, Spring 1984) (ref. 14).

At the same meeting it was reported that the USAF would use multi-channel parametric recording on some of the F-16 fleet, supplemented by 100% fit of a Leigh Mechanical Strain Recorder (ref. 15). On the other hand the USN planned to use parametric data in conjunction with seven strain gauge channels for 100% of the fleet of F-18 aircraft and parametric data with no strain sensors for 100% of the F-14 fleet (ref. 16).

In Australia there has been widespread use made of the Aircraft Fatigue Data Analysis System (AFDAS) which is a multi-channel strain gauge driven system with on-board processing supported by

subsequent ground analysis. In the UK the Fatigue Monitoring and Computing System (FMCS), which is also strain gauge driven has been designed to perform all computations on-board.

5. REQUIREMENTS

For current and future aircraft there must be more effort to produce on-board monitoring systems that derive fatigue damage at a number of locations on each aircraft in the fleet. The primary contenders seem to be:-

- o direct strain monitoring using uncalibrated strain gauges and multi-channel recorders and computing systems
- o load monitoring using load calibrated strain gauges on a limited number of aircraft to supplement and "tune" less complex systems on the remainder of the fleet
- o parametric systems using signals already available on each aircraft from flight control, avionics, navigation, fuel management and stores management systems

The main requirements for a fatigue monitoring and management system are:-

- o to ensure that aircraft do not exceed their design life
- o to identify the optimum time for the embodiment of modifications that are fatigue dependent
- o to control life used in different sorties and to identify high damage sorties
- o to identify high damage manoeuvres
- o to establish correct inspection intervals for "damage tolerant" or "fail safe" structure,

During the Specialists' Meeting a number of speakers will discuss such systems. There are many questions to be addressed, the main ones being summarised below.

- o With an uncalibrated strain gauge system what sort of repeatability might be expected from one aircraft to the next?
- o Differences in strain gauge response per unit load can exist - are these important from a fatigue monitoring viewpoint?
- o What long term reliability (20 to 30 years) can be expected from a strain gauge system?
- o How easy can new monitoring locations be accommodated?
- o With a parametric system what accuracy might be expected?
- o What verification procedure is necessary for any system?
- o If new monitoring stations are necessary what effort is involved in setting-up new parametric equations?
- o A more fundamental question relates to the main priority of a fatigue monitoring and fleet management system. For a fleetwide fit should it be to identify changes of severity of one sortie relative to another with a high degree of reliability even though the absolute accuracy in fatigue damage terms is slightly degraded? If so, to what extent should there be special studies on a limited number of aircraft and what would these be?

It is hoped that, at the conclusion of this Specialists' Meeting, guidance can be provided with regard to all these points.

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LE COEFFICIENT DE SECURITE EN FATIGUE: CALCUL DU NIVEAU DE SECURITE ASSOCIE

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1. INTRODUCTION

La tenue en fatigue d'une structure d'avion ne se laisse que très imparfaitement appréhender par les calculs du concepteur, qu'il faut sans cesse recalculer et valider par des essais.

C'est la raison d'être des grands essais de fatigue réalisés alors que l'avion concerné est déjà construit en série et souvent même déjà opérationnel.

L'objectif principal d'un tel essai est de connaître par avance les défaillances susceptibles de se produire, et de définir des actions préventives ou curatives pour les avions en service. Les informations fournies par l'essai portent d'une part sur la zone concernée et la nature du dommage, et d'autre part sur le nombre de cycles ou d'heures de vol à l'apparition du dommage.

L'exploitation de cette seconde information est délicate du fait de son caractère fortement aléatoire.

C'est à ce problème que nous nous intéressons ici, en essayant de répondre à des questions du type: "quel risque y a-t-il à laisser voler tel avion x heures?" ou: "jusqu'à quand laisser voler cet avion avec un risque d'apparition de tel dommage inférieur à 10^{-x} ?".

Cette étude concerne donc en premier lieu les avions justifiés selon le concept de "durée de vie sûre" (safe-life), ce qui est le cas de la grande majorité des avions de combat actuels. La méthode présentée peut cependant être utilisée en dehors de ce concept; elle permet en particulier de déterminer une probabilité d'amorçage d'une fissure, une probabilité de propagation sur une longueur fixée, ... en fonction des dispersions connues ou estimées et des résultats d'essai; elle peut donc être utilisée dans le cadre d'une justification en "tolérance aux dommages".

2. CADRE DE L'ETUDE

Considérons un type d'avion dans une utilisation donnée (par exemple MIRAGE 2000 monoplace en défense aérienne) dont un exemplaire en définition série a subi un essai de fatigue.

Un dommage D est apparu au cours de l'essai de fatigue, après N_E heures de vol simulées. D peut être une rupture, l'initiation d'une fissure, une fissure de taille donnée, ... tout type de dommage de fatigue.

N_E est appelé *durée de vie* de la cellule d'essai (pour le dommage D considéré).

De même pour un avion A en utilisation opérationnelle, nous appelons *durée de vie* de l'avion A (notée Q_A) le nombre d'heures de vol de l'avion à l'apparition du dommage D considéré (en supposant qu'il soit maintenu en service assez longtemps).

Le coefficient de sécurité (noté K) a pour rôle d'empêcher l'apparition de ce dommage; cet objectif sera atteint si Q_A est supérieur à N_E/K . Dans le cas contraire ($Q_A < N_E/K$), il y aura échec.

Le niveau de sécurité peut être quantifié par la probabilité d'échec, qui est la probabilité de l'événement $Q_A < N_E/K$:

$$P_{\text{échec}} = P(Q_A < N_E/K) \quad (1)$$

Pour calculer cette probabilité, il nous faut déterminer les lois de probabilité des variables aléatoires N_E et Q_A , et

avant cela examiner les paramètres qui conduisent au caractère aléatoire de la tenue en fatigue.

3. PARAMETRES INFLUANT SUR LA TENUE EN FATIGUE

De très nombreux paramètres peuvent influencer sur la tenue en fatigue d'un élément de structure. Il est possible de les regrouper dans deux ensembles distincts: d'une part ceux liés au matériau et à la technologie (tout ce qui se passe avant la livraison de l'avion aux utilisateurs), et d'autre part ceux liés à l'utilisation opérationnelle (charges subies).

3.1 Matériau et technologie

Les phénomènes de cette rubrique font qu'un élément de structure testé en laboratoire avec des conditions de chargement parfaitement définies a une durée de vie aléatoire.

Pour un avion A donné nous définissons la durée de vie N_A : nombre d'heures de vol à l'apparition du dommage D, en supposant que cet avion ait été chargé rigoureusement comme la cellule d'essai de fatigue.

Nous supposons que la variable aléatoire N_A suit une loi log-Normale, c'est-à-dire que la variable $X = \log N$ suit une loi Normale (ou loi de GAUSS). Cette hypothèse, généralement admise, peut être justifiée de diverses manières non détaillées ici.

3.2 Utilisation opérationnelle

La sévérité des vols est très variable: elle dépend de la mission effectuée, du pilote, de la configuration d'emport, ...

Pour quantifier ce phénomène, une autre durée de vie (purement théorique) d'un avion A est définie: P_A est le nombre d'heures de vol à l'apparition du dommage D, en supposant que la seule source d'incertitude dans la tenue en fatigue soit liée à l'utilisation opérationnelle. C'est-à-dire en supposant que, si les charges subies étaient parfaitement connues, la tenue en fatigue n'aurait pas de caractère aléatoire.

Ici encore, nous supposons que la variable aléatoire P_A suit une loi log-Normale ($\log P_A$ suit une loi de GAUSS). Des justifications peuvent être tentées; elles ne sont pas présentées ici.

3.3 Bilan

Dans la réalité les deux sources d'incertitude ci-dessus se combinent, et la durée de vie réelle d'un avion (apparition du dommage D) n'est ni N_A ni P_A ; elle est notée Q_A .

Il est aisé de montrer que, si N_A et P_A suivent des lois de probabilité log-Normales, il en est de même pour Q_A .

Les notations des moyennes et écart-types sont définies ainsi:

Variable aléatoire	Moyenne	Ecart-type
$\log N$	m	σ
$\log P$	m'	σ'
$\log Q$	M	Σ

Les deux premières lois sont indépendantes entre elles.

La troisième est très proche de la seconde: elle résulte de la superposition à celle-ci des phénomènes aléatoires liés au matériau et aux technologies de fabrication.

Nous pouvons ainsi nous convaincre que les moyennes m' et M sont égales, et que l'écart-type Σ se déduit de σ et σ' par la relation: $\Sigma^2 = \sigma^2 + \sigma'^2$.

Ces résultats se démontrent sans difficulté.

4. CALCUL DU NIVEAU DE SECURITE

Nous sommes maintenant à même de calculer la probabilité d'échec.

$$P_{\text{échec}} = P(Q_A < N_E/K) \\ = p(\log Q_A - \log N_E < -\log K)$$

La variable aléatoire $(\log Q_A - \log N_E)$, somme de deux variables aléatoires indépendantes, suit une loi de probabilité Normale, de moyenne $(M-m)$ et d'écart-type $(\Sigma^2 + \sigma'^2)^{1/2}$ (additivité des variances).

Nous en déduisons

$$P_{\text{échec}} = F \left[\frac{-\log K - M + m}{\sqrt{\Sigma^2 + \sigma'^2}} \right]$$

où F est la fonction de répartition de la loi Normale Réduite:

$$F(x) = \int_{-\infty}^x e^{-t^2/2} / \sqrt{2\pi} dt$$

En utilisant les relations $M=m'$ et $\Sigma^2 = \sigma^2 + \sigma'^2$, nous pouvons écrire:

$$P_{\text{échec}} = F \left[\frac{m-m'-\log K}{\sqrt{2\sigma^2 + \sigma'^2}} \right] \quad (2)$$

* K est le coefficient de sécurité,

* $m-m'$ caractérise l'écart entre l'utilisation simulée en essai et l'utilisation opérationnelle réelle moyenne,

* σ caractérise la dispersion due à l'élaboration de l'élément considéré (matériau, technologie, ...),

* σ' caractérise la dispersion de l'utilisation opérationnelle.

Dans le cas général m , m' , σ et σ' sont inconnus. La relation (2) peut cependant fournir de précieuses informations dans les différents cas de suivi d'une flotte.

5. CAS DU SUIVI INDIVIDUEL PAR CALCUL D'ENDOMMAGEMENT

Nous nous plaçons ici dans le cas où l'avion A est équipé d'un enregistreur de bord permettant de calculer l'endommagement dans la zone étudiée de chacune de ses heures de vol (voir remarque ci-dessous).

Dans ce cas l'avion est géré en endommagement: les "durées de vie" ne sont pas exprimées en heures de vol mais en endommagement cumulé, et comparées à l'endommagement de la cellule d'essai de fatigue. La formule (2) ne s'applique donc pas directement.

Le calcul de la probabilité d'échec à partir des endommagements conduit au résultat suivant:

$$P_{\text{échec}} = F \left[-\log K / \sigma \sqrt{2} \right] \quad (3)$$

En comparant ce résultat avec la formule (2), nous constatons que les termes en $(m-m')$ et σ' ont disparus: cela s'explique par le fait qu'il n'y a plus d'incertitude sur

l'utilisation opérationnelle du fait du suivi individuel qui permet de connaître les charges réelles subies par chaque avion.

La formule (3) nous permet de faire une première application numérique: la littérature donne généralement pour σ des valeurs comprises entre 0,1 et 0,15.

Le niveau de sécurité correspondant, pour le coefficient de sécurité $K=3$ classiquement retenu dans le cas du suivi individuel, est $P_{\text{échec}}=4.10^{-4}$ à $1,2.10^{-2}$.

En l'absence d'élément permettant d'apprécier la valeur de σ , il convient de prendre la valeur conservatrice de 0,15. Nous retiendrons donc que le suivi individuel par calcul d'endommagement conduit à un niveau de sécurité de 10^{-2} (probabilité que le dommage à éviter se produise sur un avion donné avant la limite de vie fixée).

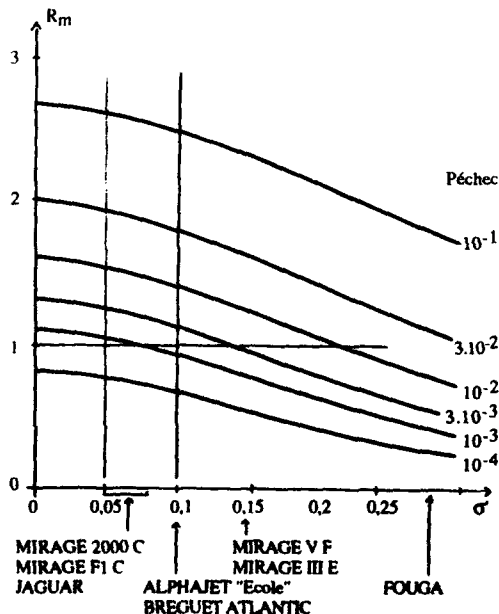
Remarque sur la notion d'endommagement calculé

L'"endommagement" réel est la grandeur croissante égale à 1 pour la durée de vie moyenne sous le chargement considéré. Aucune méthode de calcul ne permet d'accéder sûrement à cette grandeur. Nous faisons l'hypothèse que l'endommagement calculé donne l'endommagement réel à une constante multiplicative près, et que lorsque des informations font défaut pour le calcul celui-ci est mené de façon conservatrice.

6. CAS DE L'ABSENCE DE SUIVI

Lorsqu'une structure est justifiée "Safe-Life" et que la flotte ne fait pas l'objet d'un suivi individuel par calcul d'endommagement, le coefficient de sécurité adopté est classiquement de 5.

Le graphique ci-dessous donne le niveau de sécurité correspondant ($P_{\text{échec}}$), en fonction des paramètres $R_m=10^{m-m'}$ et σ' , pour $\sigma = 0,15$.



Afin de situer la zone du graphique susceptible d'être concernée, quelques exemples tirés des flottes françaises ayant fait ou faisant l'objet d'un suivi individuel sont donnés.

Les valeurs calculées pour les flottes "FOUGA", "MIRAGE III E" et "MIRAGE V F" sont peu significatives car calculées sur un faible nombre d'avions et sur une année seulement.

Les valeurs les plus significatives sont celles des flottes de Défense Aérienne "MIRAGE F1" et "MIRAGE 2000", et la flotte de la Force Aérienne Tactique "JAGUAR": σ' est compris dans la plage 0,05-0,08.

Pour les avions "ALPHAJET" des Ecoles la valeur de σ' est 0,1, de même que pour les "BREGUET ATLANTIC" de la Marine.

La zone "active" du graphique est donc délimitée en abscisse (valeur de σ') par l'intervalle [0,05-0,1].

L'ordonnée R_m mesure la représentativité de l'essai de fatigue: par exemple $R_m=2$ signifie que le chargement en essai est trop peu sévère par rapport à l'utilisation moyenne de la flotte, d'un facteur 2 sur les durées de vie (soit d'environ 15% sur les charges appliquées).

Nous pouvons ainsi conclure que le facteur 5 conduit dans le cas général à un niveau de sécurité (Péchec) équivalent à celui obtenu dans le cas du suivi individuel (10^{-2}), à la condition que le chargement appliqué en essai soit équivalent avec une incertitude de 10% aux charges réelles subies en moyenne par la flotte.

7. CAS DU SUIVI STATISTIQUE

Il y a suivi statistique lorsqu'une partie de la flotte (n avions) est équipée d'un système de suivi permettant un calcul d'endommagement (voir au §5 les hypothèses faites sur l'endommagement calculé).

La majeure partie de la flotte n'étant pas équipée, le suivi se fait en heures de vol.

Il est alors possible dans certains cas de calculer un coefficient de sécurité K moins pénalisant que 5 et conduisant au même niveau de sécurité (10^{-2}) que dans le cas du suivi individuel (cas du §5).

En effet, les enregistrements effectués sur les n avions équipés permettent de calculer un écart-type σ' et une moyenne $m-m_n$ à intégrer dans la formule (2), après avoir pris une marge de sécurité du fait du nombre limité d'avions équipés.

L'enchaînement des calculs n'est pas donné ici; il fait appel aux lois de STUDENT-FISHER (pour la moyenne) et de PEARSON (pour l'écart-type), qui quantifient les marges de sécurité à prendre en fonction du nombre n d'avions équipés et du "degré de confiance" recherché.

Voici le résultat obtenu:

$$K = K_1 \times K_2 \text{ avec } \log K_1 = m \cdot m_n \text{ et}$$

$$\log K_2 = F^{-1}[P_{\text{échec}}] \cdot \sqrt{2\sigma^2 + n \sigma_n^2 / x^2} + \frac{t}{\sqrt{n-1}} \sigma_n$$

. F^{-1} est la fonction inverse de la fonction F du §4 (fonction de répartition de la loi normale réduite),

. $P_{\text{échec}}$ est le niveau de sécurité recherché,

. n est le nombre d'avions équipés,

. σ_n est l'écart-type calculé sur ces n avions,

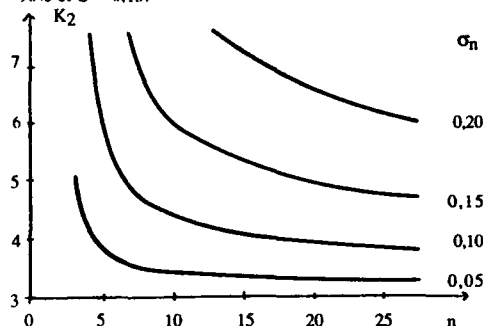
. $m \cdot m_n$ est la moyenne calculée sur ces n avions,

. t est le coefficient de STUDENT-FISHER,

. x^2 est le coefficient de PEARSON.

La détermination de K_2 peut se faire de façon simple après tracé d'un abaque. Voici celui obtenu pour un niveau

de sécurité recherché de 10^{-2} , un degré de confiance de 90% et $\sigma = 0,15$:



Nous voyons apparaître l'intérêt de cette méthode: pour 10 à 20 avions suivis et un écart-type compris entre 0,05 et 0,1 (cas le plus fréquent, voir §6), le coefficient K_2 est compris entre 3,2 et 4,3. Ces valeurs sont à comparer au coefficient 3 du suivi individuel.

8. SI PLUSIEURS ESSAIS SONT REALISES

Un seul essai de fatigue est en général réalisé, et un seul résultat est disponible. C'est ce qui a été supposé dans tout ce qui précède.

Il peut cependant arriver que plusieurs résultats soient disponibles: si plusieurs essais identiques ont été réalisés, ou si l'éprouvette et le chargement présentent une symétrie, ce qui est fréquent (par exemple demi-voilures droite et gauche sur un essai d'ensemble).

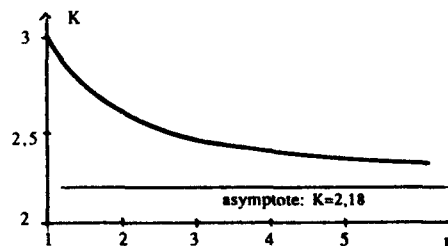
L'incertitude concernant le résultat de l'essai est alors réduite, ce qui doit permettre de réduire le coefficient de sécurité tout en maintenant le même niveau de sécurité.

Si n résultats d'essai sont disponibles, nous prendront pour N_E (durée de vie en essai) la moyenne logarithmique de ces n valeurs; l'écart-type de la loi de probabilité Normale suivie par $\log N_E$ n'est plus σ mais $\sigma/n^{1/2}$, et la formule (2) devient:

$$P_{\text{échec}} = F \left[\frac{m \cdot m' \cdot \log K}{\sqrt{(1+1/n)\sigma^2 + \sigma'^2}} \right]$$

Pour assurer un même niveau de sécurité ($P_{\text{échec}}$ constant), le coefficient de sécurité K sera d'autant plus faible que n sera grand.

Dans le cas du suivi individuel, avec $P_{\text{échec}}=10^{-2}$ et $\sigma=0,15$, K est donné par la courbe suivante:



Pour $n=2$ (deux résultats d'essai disponibles), $K=2,6$ soit un gain de potentiel pour les avions en service de 15% (par rapport à $K=3$).
Pour $n=3$, le gain est de 22%.

9. QUEL NIVEAU DE SECURITE EST ACCEPTABLE

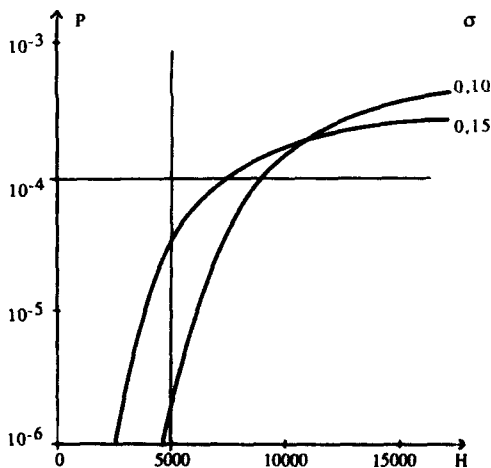
Nous nous limitons aux avions de combat, et évidemment au "temps de paix".

Plaçons nous dans le cas le plus critique, celui où le dommage D considéré est une rupture soudaine et "catastrophique" (perte de l'avion). Quelle probabilité peut-on accepter pour cet événement?

Dans la réalité, les causes de perte d'avions de combat sont essentiellement le facteur humain, parfois la panne moteur, et de temps à autre un autre problème technique.

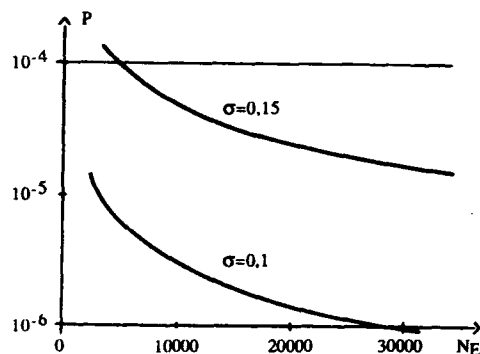
Les pertes liées au vieillissement étant marginales, le risque dépend peu de l'âge de l'avion et de ce fait se mesure en 10^{-x} par heure de vol; le taux le plus souvent reconnu pour les avions de combat est 10^{-4} par heure de vol (un avion perdu pour 10 000 heures de vol). Cette valeur est bien sûr à prendre avec précaution car très variable d'une flotte à l'autre, d'un utilisateur à l'autre, ...

En ce qui concerne notre dommage D , sa probabilité d'occurrence par heure de vol n'est pas constante, mais augmente tout au long de la vie de l'avion. Voici l'évolution obtenue dans le cas du suivi individuel, pour $\sigma=0,1$ et $0,15$ et $N_E=15\ 000$ heures ("durée de vie en essai"):



Dans le cas $\sigma=0,15$ l'application du coefficient 3 conduit à un risque au cours de la dernière heure de vol (risque maximum) de $3 \cdot 10^{-5}$ (point de la courbe correspondant à $N_E / 3 = 5000$ H). L'application d'un coefficient 2 conduit à un risque de $1,7 \cdot 10^{-4}$.

Ces résultats dépendent de N_E et de σ ; le graphique ci-dessous donne le risque au cours de la dernière heure de vol en fonction de N_E , dans le cas du suivi individuel, avec coefficient 3 et pour $\sigma=0,1$ et $0,15$:



Nous constatons que le risque est d'autant plus grand que le dommage est apparu tôt en essai (l'heure de vol est une quantité plus "significative").

Dans tous les cas, le risque au cours de la dernière heure de vol peut être considéré comme acceptable en comparaison du taux de 10^{-4} cité plus haut. Rappelons que nous nous sommes placé dans le cas extrême où le dommage D conduit à coup sûr à la perte de l'avion.

10. FLOTTES FRANCAISES

Les principales flottes françaises sont suivies en fatigue par calcul d'endommagement individuel, pour tous les avions et toutes les heures de vol: MIRAGE 2000, MIRAGE F1, JAGUAR, ALFAJET, MIRAGE IV, ...

Les autres flottes ont fait l'objet pour la plupart de mesures en service pendant plusieurs années, et il a été estimé qu'un suivi individuel n'était pas nécessaire.

Les avions embarqués sur porte-avion font exception; la raison essentielle est qu'un calcul d'endommagement fiable nécessite pour ces flottes l'enregistrement des charges subies au catapultage et à l'appontage, ce qui complique le système à mettre en œuvre. Des études de vieillissement de ces flottes sont en cours, avec campagnes de mesure en utilisation opérationnelle, essais de fatigue, ...

Pour les flottes "terrestres" au contraire, l'enregistrement du facteur de charge et des informations sur la durée du vol, le carburant et les charges emportées, la mission effectuée, ... permettent de faire un calcul d'endommagement acceptable dans les zones les plus critiques (proches de la liaison voilure-fuselage).

Un système de suivi plus complexe que l'accélérocompteur a été récemment développé (le MICRO-SPEES), et équipera les avions MIRAGE 2000-D. Ce système permet l'enregistrement de 8 paramètres analogiques (et 8 discrets) et possède une logique de concentration de données qui lui donne une autonomie conséquente pour un encombrement équivalent à celui d'un accélérocompteur. Il est également utilisé ponctuellement pour des campagnes de mesure.

Pour l'avion de combat futur RAFALE, la fonction de calcul d'endommagement sera probablement intégrée au calculateur de bord. Chaque avion calculera ainsi en temps réel ou légèrement différé son endommagement en différents points de la structure, en fonction des paramètres du vol (altitude, vitesses, accélérations, masse instantanée, incidence, braquages de gouverne, ...).

11. CONCLUSION

Une estimation du "niveau de sécurité" correspondant à un coefficient de sécurité en fatigue est possible de façon simple.

La méthode proposée, en permettant de calculer le coefficient de sécurité correspondant à un objectif donné en fonction de la méthode de suivi, montre en particulier l'intérêt que peut présenter le suivi statistique d'une flotte.

Aucune méthode de calcul ne doit évidemment être considérée comme une règle absolue pour fixer les marges de sécurité en fatigue; le calcul ne fait qu'apporter un élément de décision. D'autres éléments essentiels interviennent dans la décision, et en premier lieu les contraintes économiques et opérationnelles.

De plus certains paramètres ne sont pas pris en compte par la méthode exposée. Par exemple, si l'essai de fatigue est réalisé avec un avion retiré du service après de nombreuses heures de vol, le gain en représentativité de l'essai est considérable par rapport au cas où l'essai est réalisé avec une cellule neuve (prise en compte des aspects corrosion, dommages de maintenance, ..., vieillissement réel des structures en opération). La méthode ne prend pas non plus en compte les recalages possibles lorsque des dommages identiques apparaissent en essai d'une part et en service sur les avions les plus âgés d'autre part (mesure de la représentativité du chargement appliqué en essai d'après les dommages et leur date d'apparition, estimation d'une dispersion sur les avions en service, ...).

Le Service Officiel chargé de fixer les marges de sécurité en fatigue a, cas par cas, à faire un compromis en fonction des contraintes et des informations dont il dispose. Parmi ces informations il est indispensable que figure une estimation, même imparfaite, du "niveau de sécurité" correspondant à un coefficient.

Référence:

G.HERNIAUX: Cours de Statistique.

**Probabilistic Design and Fatigue Management Based on
Probabilistic Fatigue Models with Reliability Updating.**

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ABSTRACT

The fatigue limit state is the governing limit state for an aging air-frame. The trend of operating air-crafts longer than their originally planned life, calls for extensive testing and inspection. Current practice is to base decisions with respect to inspections and repair on the durability and damage tolerance methodologies. These procedures make little use of probabilistic methods. The paper addresses the same basic problems based on similar fundamental models. The formulations are, however, casted into a probabilistic format. In particular the possibilities of incorporating new in-service information based on e.g. inspection results or load measurements in updating the reliability is discussed and demonstrated. The advantage of this formulation is illustrated by some examples. Some comments are made on necessary future research that would be necessary to fully utilize the capabilities of probabilistic methods.

INTRODUCTION

The development of the theory of structural reliability has a history of some 50 to 60 years. e.g. Mayer(1926) proposed design based on mean and variance of random variables. Most of the early work in structural reliability appears in retrospect to have been largely ignored by later research until Freudenthal(1947) presented the fundamental problems of structural reliability of a member under random variable load; his contribution was the first to evoke acceptance among structural engineers. Cornell(1967) suggested the use of second-moment format, e.g. mean values, standard deviations and a measure of dependency. Lind(1973) showed that the Cornell's safety index requirement could be used to derive a set of safety factors on load and resistances. This work therefore represents the practical link of structural reliability to practical acceptable methods of design. Today this is taken to the level of code optimization (Ravindra and Lind, 1973). Most design codes developed today have been through some formal code calibration or code optimization process, see e.g. Tvedt et al (1991).

The development of dedicated software for solving structural reliability problems seems to have started in 1975, see e.g. Helder (1979). The procedures used were documented in guideline reports (CEB, 1976, CIRIA, 1977, CSA 1981). Today several general purpose structural reliability software products are available, e.g. Tvedt (1988). A modern general purpose structural reliability software includes a number of dedicated analytical and simulation methods to effectively calculate the small probabilities of structural failure, together with distribution methods to calculate the full distribution of e.g. response quantities. Quite general methods for updating the reliability based on in-service observations of the structural performance like inspections are also available.

In the marine industries probabilistic methods are used directly both for design purposes and inspection planning, and guidelines for use of the methods are developed, see Torhaug et al.(1991) and Ditlevsen and Madsen (1989). In particular the methods have gained acceptance during the last few years, e.g. in the North Sea most offshore Jacket (frame) structures have their inspection plans developed on direct use of probabilistic models including description of the uncertainties in the environment, the global response analysis, the local stress analysis, the fatigue strength analysis and the uncertainty in the inspection results, see e.g. Skjong and Torhaug(1991), Madsen et al (1989) and Madsen et al. (1987). The interest in improved inspection

programs and methods for evaluating structural reliability in order to possibly extend the in-service life has to some extent been initiated by the need in the industry to use existing structures longer than their anticipated design life.

Similarly, the trend over the past decades has been to operate air-crafts longer than their originally planned life. Continuing service requires extensive analysis and testing to establish airframe inspection and modification action required to ensure the desired new design life.

The present durability and damage tolerance design requirements are based on fatigue crack growth prediction laws subjected to predefined load spectra extracted from real measured load spectra. The durability and damage tolerance requirements assume the existence of a fixed deterministic crack size in the structural details after fabrication. The in-service inspection requirements are then established from further assumptions on deterministic crack growth rates using assumed loads spectra. These spectra depend to some extent on the utilization of the aircraft to reflect the difference in use of the individual aircrafts. The critical crack size is established deterministically from the design limit stress.

As an alternative to the above approaches the paper will address the problem of inspection planning (and possible life extension) from a probabilistic viewpoint, see e.g. Yang and Manning (1980) (1989). The initial crack sizes and the crack growth laws parameters will be modeled as random, an attempt to illustrate the effect of uncertainties in the load spectrum are made. Inspection methods will also be modeled with its associated uncertainties in detection probabilities (for no-finds) or sizing uncertainties (when cracks are found). It can thus be demonstrated that probabilistic methods represent a methodology where all aspects of the deterioration process are modeled, and that in-service inspection results can be accounted for. Trade-off studies can therefore be made between all parameters influencing the fatigue reliability of airframes.

RELIABILITY METHODS

Uncertainties are always the main motivation for inspections. Uncertainties governing the fatigue limit state stems from uncertainties in the load predictions, the response analysis/models, the detail stress analysis and the material parameters. To treat all uncertainties in a consistent way structural reliability models can be applied, see e.g. Madsen et al.(1986), Thoft-Christensen and Baker (1982). In a structural reliability formulation the fundamental notion is the limit state function $g(x)$, the failure criterion, which divides the set of the random variables x into the failure set $g(x) \leq 0$ and the safe set $g(x) > 0$, see Figure 1 for illustration. The safety is assured by requiring that the probability content of the failure set P_f is less than a some given number. A corresponding reliability measure is the reliability index $\beta_p = -\Phi(P_f)$. Here $\Phi()$ is the standard normal distribution. Fast computer programs for calculating this probability, together with a rich variety of importance, see Madsen(1987), and sensitivity measure are available, e.g. Tvedt (1988). The parametric sensitivity parameters includes such measures as $d\beta/dx$, where x can be any distribution parameter of the random variables x (e.g. mean values, standard deviations, correlations) or a fixed/deterministic variable. This is an important property of the First Order Reliability Methods (FORM) which are the methods of particular importance in resource allocation models and trade-off studies.

Updating

In the same way as the limit state represent the failure criterion it is possible to represent observations as events, represented by, $h(x) < 0$ or $h(x) = 0$. Such observation may represent inspections where cracks are not found, a crack is observed or load and response measurements. The updated reliability can then be calculated by use of Bayes theorem, see Figures 2 and 3. In the case of inequality events (e.g. no cracks are detected):

$$P(g(x) < 0 | h(x) < 0) = \frac{P(g(x) < 0 \cap h(x) < 0)}{P(h(x) < 0)} \quad (1)$$

That is the intersection of the "failure" and the "observation" divided by the probability of the "observation". The numerator are thus calculated as an intersection or parallel system.

And in the case of equality events (e.g. given crack size is observed):

$$P(g(x) = 0 | h(x) = 0) = \frac{P(g(x) = 0 \cap h(x) = 0)}{P(h(x) = 0)} \quad (2)$$

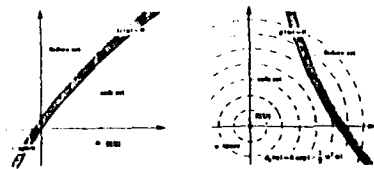


Fig 1a: Mapping from basic formulation space to standard normal space.

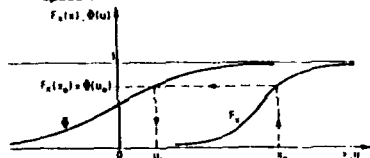


Fig 1b: Transformation (Rosenblatt) in one dimension

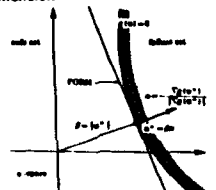


Fig 1c: Linearization at design point (FORM)

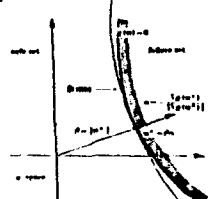


Fig 1d: Second order approximation (SORM)

Figure 1: First and Second Order Reliability (FORM/SORM) methods transforms the basic stochastic variables into a space of standard normal variables. No approximation is made in the transformation. For FORM the transformed limit state function is approximated in the design point by a hyperplane and in SORM by a Second Order surface. The design point is the point of highest failure probability density in the failure set.

where the partial derivatives are derived at $\Theta = 0$. Such events may represent cracks found and measured at certain times or loads measured during a time interval. These formulas are easily generalized to updating conditioned on a number of observations of $\leq$ type and a number of observations of equality type. The equality constraints are calculated from the sensitivity factors for intersections, see Hohenbichler and Rackwitz (1988), Tvedt (1989).

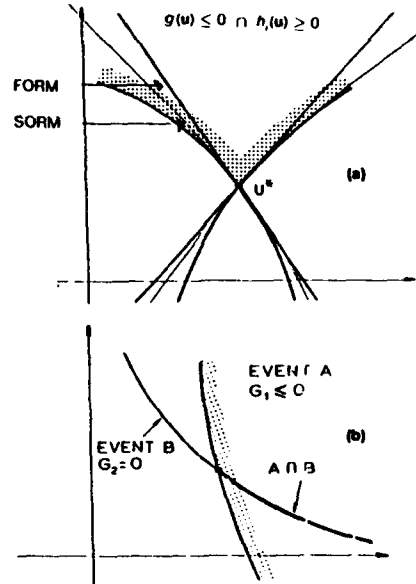


Figure 2: First and Second Order Reliability (FORM/SORM) methods used for reliability updating. The problem reduces to calculating reliabilities of intersections and in the case of equality constraints the problem reduces to the problem of calculating sensitivities on intersections.

Probabilistic models for durability analysis.

Durability may be considered as a quantitative measure of the structures resistance to fatigue cracking under specific service conditions. The durability of a structure is usually concerned with sub-critical cracks which may cause functional impairment (e.g. fuel leakage, linament breakage, loss of cabin pressure, etc.), and adversely affect the operational readiness and maintenance repair costs of airplanes. These cracks are generally smaller than can be found with Non-Destructive Inspection (NDI) techniques (e.g. cracks smaller than 1.27 mm/0.05 in.); thus the durability life is often considered to be equivalent to the crack initiation life.

For aluminum fastener hole specimens, subject to a fighter plane load spectra, extensive fractographic results indicate that in the small crack region, the crack growth equation can be expressed as

$$\frac{da(t)}{dt} = Qa^b(t) \quad (3)$$

where $a(t)$ is the crack size at t flight hours, and Q and b are crack growth parameters, and e.g. Yang and Donath(1983), see Figure 4a for illustration. Taking the logarithms on both sides one obtains

$$Y = bU + q \quad (4)$$

where

$$Y = \log \frac{da(t)}{dt}, \quad U = \log(a(t)), \quad q = \log Q \quad (5)$$

Straight lines are fitted by Maximum likelihood estimates to the individual experiments, see Figure 4b. This gives Q and b and then obtaining $a_0 = a(0)$ from Equation 3, using backward integration. Joint statistics for the three parameters can thus be established, see example below.

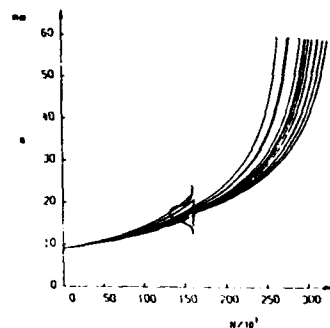


Figure 3: Illustration of reliability updating. By the updating unlikely outcomes are excluded by inclusion of new information.

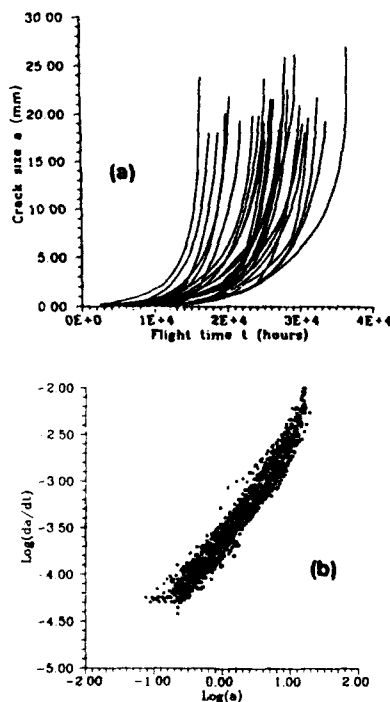


Figure 4: Data from crack growth of fastener hole specimens.

The limit state function for the durability model can now be established as

$$g_{\text{dur}}(x) = g(t, a_e, a_0, Q, b) = a_e - a(t) \quad (6)$$

where a_e is the economic repair limit and $a(t)$ can be found by solving Equation 3

$$a(t) = (tQ(1-b) + a_0^{1-b})^{1/(1-b)} \quad \text{for } b \neq 1 \quad (7)$$

$$a(t) = \exp(tQ + \ln(a_0)) \quad \text{for } b = 1 \quad (8)$$

Example

The failure is defined to be when the crack size reaches the economic repair limit $a_e = 1.27$ mm. The initial crack size distribution and the distribution of the crack growth parameters Q and b are established from the crack growth data from the "Fastener Hole Quality" program, Noronha et al. (1978). From this reference, the data termed "wfl" and "wflp", shown in Figure 4a, have been used to represent the critical points analyzed. Standard tests gave the best fit by a Weibull distribution for the initial crack size a_0 and Normal distribution for the crack growth parameters Q and b , see Figure 5.

$$W(a, t, \cdot) = a \cdot I(x, \cdot)^{1-b} \exp(-I(x, \cdot)^b) \quad (9)$$

with

$$\mu = a_e^{-1/(1+b)} \left((1+1/\eta) + \cdot \right); \quad \sigma = a_e^{-1/(1+b)} \left((1+2/\eta) - (1+1/\eta)^2 \right)^{1/2}$$

with the parameters $a_0 \in W(\mu = 0.03239 \text{ mm}, \sigma = 0.03244 \text{ mm}, 0)$, see Figure 5. The other two random variables are defined through the data by

$$N(\mu, \sigma) = \frac{1}{\sqrt{2\pi}\sigma} e^{-1/2(x-\mu)^2/\sigma^2} \quad (10)$$

$$Q \in N(2.66 \cdot 10^{-4}, 5.56 \cdot 10^{-5}) \quad (11)$$

$$b \in N(1.03, 0.0864) \quad (12)$$

$$\rho[Q, b] = 0.03 \quad (13)$$

These random variables can be used as input to the durability limit state giving the reliability index as a function of time as shown in Figure 6. The parametric sensitivity factors can be used as follows:

$$I_{\text{new}} = I_{\text{calculated}} + \Delta I / \Delta t \quad (14)$$

As shown in Figure 6, for $t = 4100$ hours, the reliability is $I = 4.0$. The parametric sensitivity factor for the standard deviation of the initial crack size a_0 at $t = 4100$ hours, is $\partial I / \partial a_0 = 104.25$. Assuming that new observations result in the increase of a_0 with 10%, the new reliability at time $t = 4100$ hours, is obtained as

$$I_{\text{new}} = 4.0 - 104.25(0.1 \cdot 0.03244) = 3.66$$

This shows that all changes within reasonable limits may be calculated by hand. This example shows that an increased a_0 reduces the reliability, which further leads to an earlier required inspection.

Probabilistic damage tolerance analysis.

Damage tolerance is concerned with the safety and reliability of the structure rather than the economics of its operation. Damage tolerance life can be thought of as life of the structure from crack initiation until final fracture. In airframe design, crack initiation is defined to be the point when the crack reaches an inspectable crack length, and failure is defined to be when the crack reaches the critical crack size. Since the cracks of concern in the damage tolerance life are by definition of inspectable size, inspections during the service life may be used to ensure that the damage tolerance requirements are met.

The advantage of being able to quantify the fatigue damage accumulation in airframes using fracture mechanics methodology have resulted in USAF's military specifications defining durability and damage tolerance requirements in terms of crack propagation characteristics, see i.e. MIL-A-83444 (1974), MIL-STD-15030A (1975), MIL-A-8866B(1975), MIL-A-8867B(1975), MIL-A-87221(1985). Current fatigue design specification requires that adverse cracking should not occur within the two lifetimes of the airframe for specified usage and environment. The initial flaw size assumptions are given in specifications for a deterministic analysis. The inherent reliability resulting from these requirements are not known.

Major uncertainties can be found in the initial fatigue quality estimates of the structural material, load spectrum variations, geometric effects, inaccurate prediction of crack growth, critical stress intensity factors and inspection quality. These uncertainties can be considered in a probabilistic analysis.

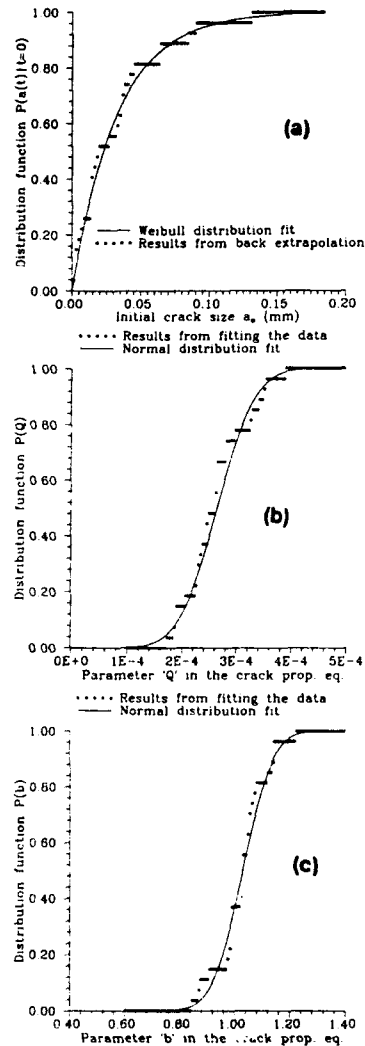


Figure 5: Fitted distributions for the durability models; initial crack size, a_0 , and crack growth parameters Q and b .

Numerous fatigue crack growth models have been proposed in the literature to describe the relationship between the crack growth increment in a stress cycle da/dN and the stress-intensity range, ΔK . In the present analyses we have chosen the model proposed by Johnson(1981), since this successfully models crack growth load interaction effects due to spectral loading. This model, the so-called Multi-Parameter Yield Zone model, accounts for crack growth retardation, acceleration and under-load effects. The load interactions are attributed to residual stress intensity due to the plastic deformation at the crack tip. The model is based on a slightly modification of the Forman crack propagation model, see Forman et.al(1967) and modifications of the Willenborg retardation model, see Willenborg et.al (1971). The crack size increment in one stress cycle is

$$\Delta a_i = C \frac{\Delta K_i^{m_1}}{(1 - R_i^{m_2})^{m_2} K_{IC} - \Delta K_i} \quad (15)$$

where $m_1 = 1$ at $R_i^{m_2} \geq 1$, and $m_1 = 2$ at $R_i^{m_2} < 0$.

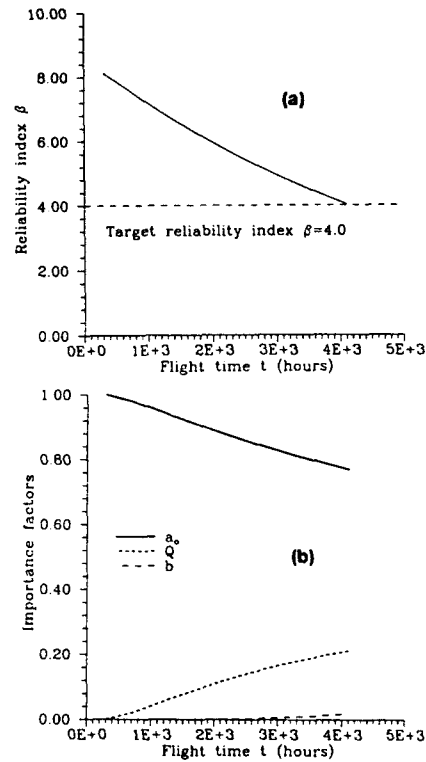


Figure 6: Results from the durability limit state. Included are also the importance factors as given by a FORM.

For details of the model, reference is given to Johnson(1981). The limit state function is now expressed as

$$g(x) = a_c - a(N) \quad (16)$$

a_c is the critical crack size and $a(N)$ is the crack size after N cycles. $a(N)$ can be estimated as

$$a(N) = a_0 + \sum_{i=1}^N \Delta a_i \quad (17)$$

The index i represents the i 'th stress cycle. a_0 is the initial crack size as established/described under the durability analysis, except that it is as found by the regression analyses at the time t of the first inspection, i.e. $t = 4100$ hours, conditioned on the POD in the case of no crack was found. Further,

$$\Delta K_i = \Delta \sigma \sqrt{\pi a_i} \sqrt{\sec(\pi a_i / 2w)} \quad (18)$$

where $\Delta \sigma$ is the stress range and w is the half specimen width. $R_i^{m_2}$ is a function of the previous stress and crack propagation history. The summation over stress cycles thus starts at the time of the first inspection, where a_0 has been updated conditioned on no crack found at the first inspection. At the present stage we have not developed a random model for the load spectra. The loads are randomized for illustrative purposes by scaling the deterministic spectra by a random factor x_i . Thus

$$\Delta K_{i, actual} = \Delta K_{i, deterministic} x_i \quad (19)$$

x_i represents the uncertainty in transferring the wing bending moment to stress at the fastener hole, and $\Delta K_{i, actual}$ is used in Equation 15.

The random variables in the model has thus been limited to: a_0 , C , n , x_i and x_i .

The limit state is therefore written as

$$g(x) = g(N, a_C, a_T, C, n, x_T, x_L) = a_C - a(N, a_T, C, n, x_T, x_L) \quad (20)$$

In the case a crack is not found in an inspection at time N_i with a measured load spectra the event margin representing this no-find can be written as

$$h_i(x) = g(N_i, a_{POD}, a_T, C, n, x_T, x_L = x_i), h_i(x) \geq 0 \quad (21)$$

where a_{POD} is the POD distribution or the distribution of the smallest detectable crack size, (see e.g. Yang and Chen(1984)) and $x_L = x_i$ represents a measured spectrum. The other variables are the same as in the failure event represented by $g(x)$, except that the time is the time of inspection N_i . The conditioned reliability can be calculated by Equation 1.

When cracks are found, they are usually measured down to a certain tolerance. This event can be formulated as

$$h_j(x) = g(N_j, a_D, a_T, C, n, x_T, x_L = x_j), h_j(x) = 0 \quad (22)$$

a_D is the measured crack size, which can be modeled as a stochastic variable because of the uncertainty in sizing the crack. The reliability updating can be performed using Equation 2.

Example

The load spectrum used in this example is a deterministic fighter load spectrum with a peak stress of 234 MPa (34 ksi) (365 different missions, total 400 flight hours and approximately 33330 stress cycles), i.e. x_T and x_L are assumed to be deterministic. The stochastic variables are assumed being represented by

$$\begin{aligned} a &: W(\mu = 8.56 \cdot 10^{-5}, \sigma = 7.51 \cdot 10^{-5}, 0) \\ C &: N(4.626 \cdot 10^{-9}, C.O.V. = 0.1) \\ n &: N(3.171, C.O.V. = 0.1) \\ [C, n] &: 0 \end{aligned}$$

where all units are conditioned on crack increments in meters.

The POD is chosen as represented by a Weibull distribution with parameters $a_{POD} \in W(\mu=0.7 \text{ mm}, \sigma=0.2 \text{ mm}, 0)$, see Figure 7. The critical stress-intensity factor is taken as $K_{IC} = 88 \text{ MPa}\sqrt{\text{m}}$ and the critical crack size as $a_c = 5.08 \text{ mm}$ (0.2 in.). Crack growth curves, for different values of the stochastic variables are shown in Figure 8.

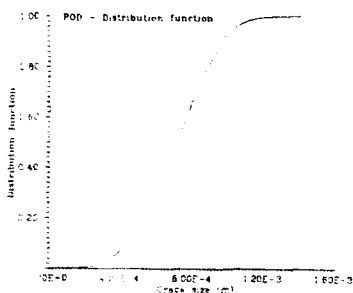


Figure 7: POD - Distribution.

Assuming that no crack is detected in the 1. inspection, $t = 4100$ hours (results from the durability analysis) the failure probability can then be calculated from equation (1) (number of flight hours t is now used instead of number of stress cycles)

$$P_f = P(g(x) \leq 0 | h_1(x) \leq 0) \quad (23)$$

where

$$g(x) = a_C - a(t, a_T, C, n) \quad (24)$$

$$h_1 = a_T - a_{POD} \quad (25)$$

Still using target reliability index $\beta = 4.0$, the 2. inspection will be required at time $t = 7350$ flight hours. The reliability index as a function of the time is shown in Figure 9.

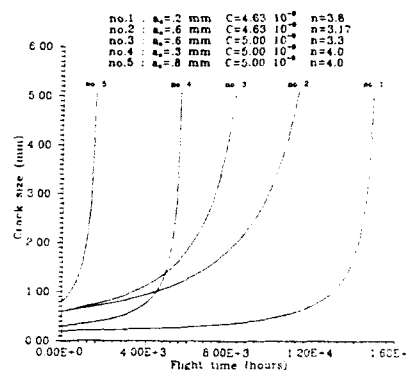


Figure 8: Crack growth for different values of the initial crack size and the crack growth parameters.

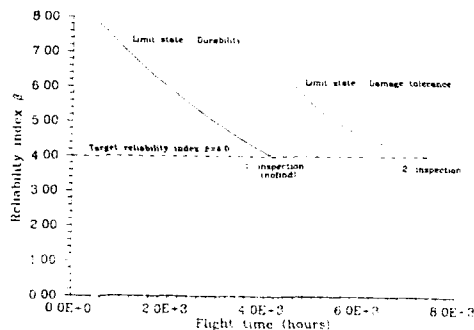


Figure 9: Results from the durability analysis ($t < 4100$ flight hours) and the damage tolerance analysis, including reliability updating based on no-find in 1. inspection.

Two different outcomes of the 2. inspection are considered in the following:

1) No cracks detected in 2. inspection:

The failure probability for $t \geq 7350$ flight hours can be calculated as:

$$P_f = P(g(x) \leq 0 | h_1(x) \leq 0, h_2(x) \leq 0) \quad (26)$$

where $g(x)$ and h_1 are given in equations 24 and 25 and

$$h_2 = a(t=7350, a_T, C, n) - a_{POD} \quad (27)$$

The reliability index as a function of the time is shown in Figure 10. The 3. inspection is at time $t = 8900$ flight hours.

2) Crack is detected in 2. inspection, $a_D = 0.8 \text{ mm}$:

The failure probability for $t \geq 7350$ flight hours can be calculated as:

$$P_f = P(g(x) \leq 0 | h_1(x) \leq 0, h_2(x) = 0) \quad (28)$$

where $g(x)$ and h_1 are given in equations 24 and 25 and

$$h_2 = a(t=7350, a_T, C, n) - a_D \quad (29)$$

The reliability index as a function of the time is shown in Figure 11. The 3. inspection is at time $t = 7700$ flight hours.

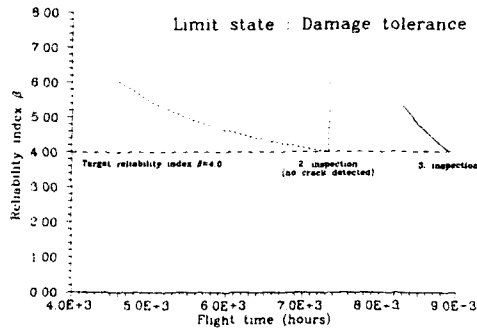


Figure 10: Results from the damage tolerance analysis ($t \geq 4100$ flight hours), including reliability updating based on no-find in 1. and 2. inspection.

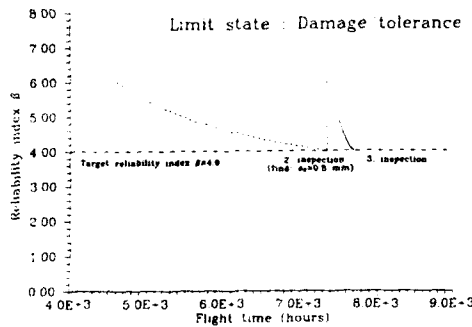


Figure 11: Results from the damage tolerance analysis ($t \geq 4100$ flight hours), including reliability updating based on no-find in 1. inspection and crack is detected in 2. inspection, $a_D = a(t=7350) = 0.8$ mm.

ECONOMIC DESIGN LIFE

As can be seen from the preceding models the methodology described so far can be used for inspection optimization in the sense that the optimal time to next inspection which maintains the reliability level can be derived as well as how long the life can be extended with the same reliability level. This is thus a highly relevant model for inspection planning when information of the airframe integrity is collected through inspections and load surveys.

This model is however, of less interest at the design stage and when considering economic in-service life extension. The model for such decisions has to be quite different. At the design stage the trade-off study has to be made between design parameters like plate and stiffeners thicknesses and the cost of inspections in service. At the design stage no in-service inspection results are available and only expectations on inspection results can be calculated. The same considerations are valid when life cycle costs are considered. All possible results of future inspections can, however, be modeled as previously described, e.g. Madani (1988), Shiong et al (1989).

For example, the basic safety margins for the fatigue sensitive detail is written as, (Equations 8 and 20)

$$M = g_{\Delta}(x(z)) = g(t, a_0, a_r, Q, b) \quad (30)$$

or

$$M = g_{\Delta}(x(z)) = g(N, a_0, a_r, C, n, x_1, x_L) \quad (31)$$

for the damage tolerance and durability life, respectively. x is a function of the design variables z .

If inspections are performed at times T_1 and T_2 with no repair at time T_1 and repair at T_2 , and T_1 and T_2 fall within the damage tolerance life, the safety margin for failure time $t > T_2$ is

$$M_{01} = g(N_2 - N_1, a_0, a_r, C, n, x_1, x_L, x_{L, obs}(0, T_2)) \quad (32)$$

where a_r is the crack size after repair, modeled as a random variable and $x_{L, obs}(t_1, t_2)$ is a measured load spectrum from time t_1 to t_2 .

The event margin corresponding to the event that a crack is found and repair is performed at the first inspection T_1 can be formulated as

$$R = g(N_1, a_0, a_r, C, n, x_1, x_L, x_{L, obs}(0, T_1)) \quad (33)$$

where a_r is the smallest detectable crack size.

Similarly, the event margin corresponding to the event that repair is performed at the 3rd inspection (time T_3) given repair at T_1 and no repair at T_2 is written as

$$R_{10} = g(N_3 - N_1, a_0, a_r, C, n, x_1, x_L, x_{L, obs}(T_2, T_3)) \quad (34)$$

Assuming that I inspections are performed at times $T_i, i \in [1, I]$ the reliability index β for failure before t is

$$\beta(t) = -\Phi^{-1}(P_F(t)) \quad (35)$$

where, for $0 < t \leq T_1$,

$$P_F(t) = P(M(t) \leq 0) \quad (36)$$

and for $T_1 < t \leq T_2$

$$P_F(t) = P_F(T_1) + P(M(T_1) > 0 \cap R > 0 \cap R_0(t) \leq 0) + \quad (37)$$

$$P(M(T_1) > 0 \cap R \leq 0 \cap R_1(t) \leq 0)$$

etc. for the paths in Figure 12 of repair realizations. From this the expected number of repairs can be calculated $E[R]$.

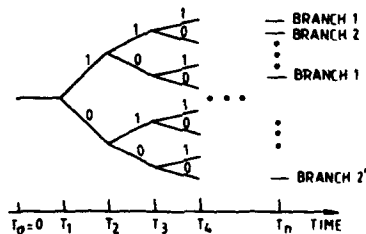


Figure 12: Repair realizations for single elements. 0 denotes no repair while 1 denotes repair.

The resource allocation model is now formulated as

$$\min C(t, q, z) = C_0(z) + \sum_{i=1}^I C_{R_i}(q_i) \quad (38)$$

$$+ C_R E[R_i | (1 - \Delta P_F(T_{i-1}, T_i)) / (1+r)^{T_i}]$$

$$+ \sum_{i=1}^I C_F(T_i) \Delta P_F(T_{i-1}, T_i) / (1+r)^{T_i}$$

subjected to the reliability constraint $\beta(t) \geq \beta_{min}$, the minimum and maximum time between inspections $t_{min} \leq t \leq t_{max}$, the limitations on inspection quality $q_{min} \leq q_i \leq q_{max}$ and the limitations on the design variable $z_{min} \leq z \leq z_{max}$.

r is here the real rate of return and the cost functions are initial cost $C_0(z) = C_{i0} + C_{r0}(z - z_0)$ (a function of the design variable z), the inspection cost $C_{R_i}(q_i)$ which could be a function of the inspection quality q_i , the repair cost C_R and the cost of failure C_F .

The control variables in the optimization formulation for a given design (z) are the inspection times $t = (t_1, t_2, \dots, t_I)$ and the inspection qualities $q = (q_1, q_2, \dots, q_I)$. By performing the optimization for different life times the optimum life extension can be found. Examples of these optimization studies

remains to be performed for airframes, examples from offshore applications can be found in Skjong et al. (1989). The conclusions from such studies can easily be transformed into optimum design life requirements, since the total life cycle costs are included in the model.

CONCLUSIONS

- A probabilistic fatigue crack growth model has been formulated both for the durability and the damage tolerance limit states.
- Methods for reliability updating during in-service has been demonstrated, giving the possibilities of optimization of inspection plans.
- Reliability methods provide a rationale for in service considerations of an aging airframe.
- Better utilization of using probabilistic inspection planning will require a random load model based on load measurements.
- The developed models can be used in design to specify the design life requirement that leads to minimum life cycle costs.
- Reliability analysis combined with a resource allocation technique provides data which are important for reduced life cycle costs of an air-frame considering design, in-service inspection/repair and consequence costs of a possible failure.

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AGING AIRCRAFT STRUCTURAL DAMAGE ANALYSIS

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SUMMARY

The structural problems experienced by both military and commercial aging aircraft are described. The programs that are in place in the US Air Force, the research that is being performed and the facilities that may be used to address the problem of aging aircraft are identified. The use of one research product, the computer program PROF, describes how current technology can be used to predict damage in aging aircraft structures and the reliability of those structures. Areas where additional research is required are identified and conclusions are drawn.

LIST OF SYMBOLS

a	crack size
$a_p(T_R)$	pth percentile of the crack size distribution at time T_R
da/dN	crack growth per cycle
$f_{after}(a)$	crack size distribution inspection and repair
$f_{before}(a)$	crack size distribution maintenance
$f_r(a)$	equivalent crack repair size distribution
$f_T(a)$	crack size distribution at any time T
$g(K_{IC})$	fracture toughness distribution
K_{IC}	fracture toughness
$P[x]$	probability of "x" occurring
POD(a)	probability of detecting a crack of size "a"
POF	probability of fracture
$POF_A(T)$	single flight POF from any element in a single airframe at T hours
$POF_F(T)$	single element POF during flight at T hours
$POF_r(T)$	single flight POF from any element in any airframe in a fleet at T hours
T_R	reference time
T	number of spectrum hours
β	stress intensity geometric correction factor
ΔK	stress intensity factor range
ΔT	change in time (spectrum hours)
σ_{cr}	critical stress
σ_{max}	maximum stress

1. INTRODUCTION

Aging aircraft is a growing problem in both military and commercial aviation. With the economics of keeping current

military and civilian aircraft in service, and the growing demand for air travel worldwide, the problems of aging aircraft will continue to increase in importance. The purpose of this paper is to report on recent and ongoing USAF research that is applicable to aging aircraft, to demonstrate how the technology can be used to aid in the management of an aging aircraft fleet, and to identify areas where more research is required.

The problem of aircraft aging can be defined in a number of different ways. When addressed from a flight hours (or equivalent flight hours) perspective, an aging aircraft is an aircraft that is approaching its designed service life in terms of flight hours. From the standpoint of flight cycles, an aging aircraft is an aircraft that has sustained a certain percentage of its designed ground-air-ground cycles (take-offs and landings). From a structural damage standpoint, an aging aircraft would be defined as an aircraft that contains multiple site damage (MSD) characterized by the link-up of fatigue cracks. Looking at calendar age alone, an aging aircraft could be defined as an aircraft that has been in service for a certain number of years, independent of its actual usage. In any event, the criteria that is used to define aircraft aging is relatively unimportant. The important issue is that the structural strength properties of the aircraft are decreasing with time and usage and will continue to decrease if no action is taken.

In a general sense, aging aircraft are characterized by the deteriorating strength properties of the structure, and the related problems and increasing costs of maintenance that result. Some of these problems are time related, such as corrosion, which also depends heavily on the usage environment. Others are usage dependent, such as in fatigue cracking, which is naturally caused by the mechanical loads that are introduced into the structure. Most often, the damage state of an aircraft is the result of both time and usage. To maintain structural reliability, steps must be taken toward the prevention, detection, repair and prediction of the initiation and growth of aircraft structural damage.

Three factors that have an effect on the problems associated with aircraft aging are the usage of the aircraft, the environment the aircraft is subjected to, and the inspection and maintenance practices of the maintenance personnel. Depending

on how an aircraft is used, the aircraft may have an expended life significantly different from what is predicted for that aircraft at that time. The simple fact is that aircraft are often not used the way they were intended to be used when they were designed and commissioned into the fleet. The effect is that an aircraft that is used harder than expected will have a higher cumulative equivalent flight hours than expected. As a result, that aircraft may have a higher damage state than is predicted, and may have a corresponding higher probability of failure.

Depending on where an aircraft is used, an aircraft will have corrosion problems indicative of the environment. This is most prevalent for military and small airline aircraft. Military aircraft that are stationed near saltwater environments experience a higher degree of corrosion than other military aircraft. However, these military aircraft are probably the most carefully monitored and cared for aircraft as far as corrosion is concerned.

Through the requirements of the US Air Force Aircraft Structural Integrity Program (ASIP)⁽¹⁾, the inspection of military aircraft for structural damage and subsequent repair are routinely performed at Air Force Air Logistics Centers at prescribed usage intervals. These intervals occur when an aircraft experiences one half of the number of cycles that it takes a crack of a given size assumed to be present in the structure to grow to its critical length. After the aircraft is inspected, the damage that is detected is repaired. The damage state that is assumed to be present in the structure is then readjusted to correspond with the capability of the inspection technique that was used.

The prevention of structural damage probably has the greatest payoff in the area of corrosion. Generally, all surfaces of an aircraft that will come in contact with the environment are either painted or otherwise coated with corrosion inhibiting materials. Materials that react corrosively when they contact each other are kept from making a good contact. Although these practices greatly decrease the areas subjected to corrosion damage, corrosion still occurs due to the wearing or cracking of coatings exposing the bare metal to the environment.

2. USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)

The US Air Force established the Aircraft Structural Integrity Program (ASIP) in 1959. ASIP provides for⁽¹⁾:

- 1) establishing, evaluating and substantiating the structural integrity (strength, rigidity, damage tolerance, and durability) of airframes, 2) the acquisition, evaluation, and utilization of operational aircraft usage data for the continual assessment of in-service integrity of individual aircraft, 3) provides a basis for determining logistics and force planning requirements, and 4) provides a basis to improve structural criteria and methods of design, evaluation,

tion, and substantiation for future aircraft. The performance concept of the ASIP is a time-phased set of required actions performed during the life of an aircraft to ensure that the aircraft operational life is at least equal to its design life. The ASIP identifies and schedules critical structural tests that are required during full scale development of a new aircraft. Thus the need for structural retrofit of large numbers of aircraft later in operational commands is minimized. An orderly schedule of inspections, replacements, or repair actions is established to maintain the structural integrity of the airframe. The scheduling of these actions is determined by damage tolerance and durability analyses using the flight loads and inspection data associated with force management.

2.1 Damage Tolerance Analysis

For every fracture critical location in an airframe, analyses have been performed for deterministically predicting the growth of initial fatigue cracks assumed to exist at these locations. These analyses require: 1) representative stress spectra from which flight by flight stress sequences can be derived, 2) stress intensity factor coefficients to account for differences in structural detail geometries, and 3) crack growth rates as a function of stress intensity factor. These data items can be used to predict crack growth versus time (spectrum hours) relationships in the usage environment. These relationships, when coupled with the probability of detection (POD) curve for quantifying inspection capability in terms of a reliably detected crack size, a_{ms} , produce the curves from which the Air Force damage tolerance approach to establishing inspection intervals has been formulated. Inspection is to be made at no later than half the time it will take a crack to grow to its critical size. This process is illustrated in Figure 1.

2.2 Durability Analysis

Durability is normally defined in terms of the number of flight hours until cracking in the airframe impairs the function of the structure. In the past, durability has been deterministically demonstrated by showing that a crack representative of the initial quality of the structure (e.g., 0.127 or 0.254 mm (0.005 or 0.010 inch)) will not grow to a functional impairment size (e.g., through the skin thickness or to an uneconomical repair size) in two design lifetimes. A schematic of the deterministic durability approach is shown in Figure 2.

A stochastic approach to durability has recently been developed⁽²⁾. In the stochastic approach, the initial quality of a structural element is characterized in terms of an Equivalent Initial Flaw Size Distribution (EIFSD). The EIFSD is obtained by conducting tests, collecting data and determining the distribution of times for a crack to reach a given reference size in a structural element subjected to a specified loading history.

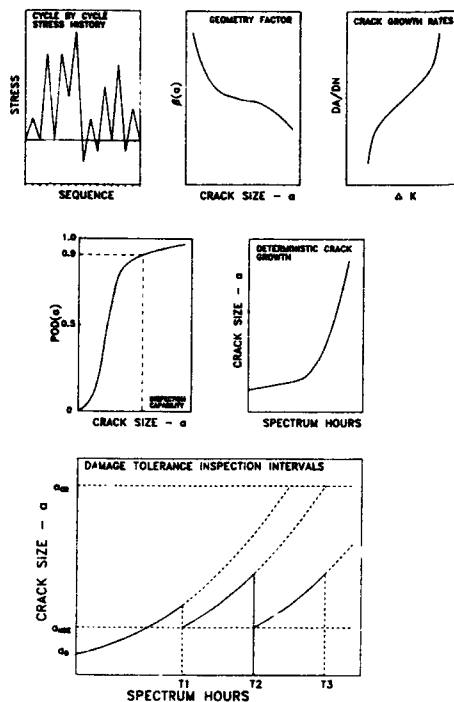


Figure 1. Schematic of Deterministic Damage Tolerance Analysis

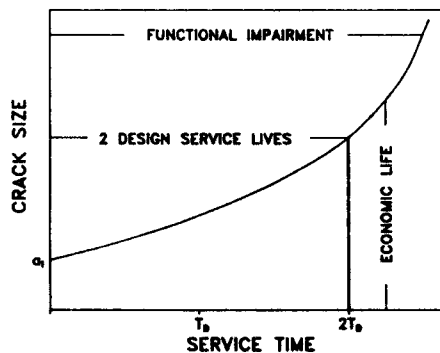


Figure 2. Deterministic Durability Approach

This distribution of times is known as the Time To Crack Initiation (TTCI) distribution. The TTCI distribution is then stochastically modeled backward in time to obtain the flaw size distribution at time zero (EIFSD). The EIFSD is then "grown" forward in time (using the same stochastic model) to determine the flaw

size distribution at any service time. Durability is then based on the number of spectrum hours for low percentiles of the flaw size distribution to reach the functional impairment size. A schematic of the stochastic durability approach is shown in Figure 3.

Further research is currently being performed on the stochastic durability approach by the Flight Dynamics Directorate of the Wright Laboratory in conjunction with the Aluminum Company of America (Alcoa). In this research, two different qualities of 7050 aluminum plate are being investigated to determine the role material quality plays in the initial quality (EIFSD) and thus the durability of a structural element. At the same time, the stochastic durability approach is being evaluated and any needed improvements will be identified.

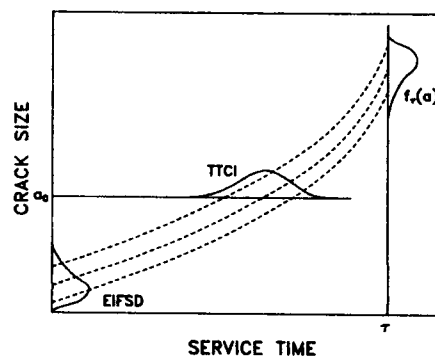


Figure 3. Stochastic Durability Approach

2.3 Force Management

The Air Force collects two types of operational usage data within the ASIP requirements for Force Management. These are Individual Aircraft Tracking (IAT) and Loads/Environment Spectra Survey (L/ESS) data. The IAT data are potentially useful in risk analyses because they provide a method for correlating the results of inspection feedback between aircraft of a fleet which may have been used differently. The L/ESS data can be used to verify and update the data of the magnitude of the stresses that the aircraft is expected to encounter. The IAT and L/ESS data are collected and assembled under the Aircraft Structural Integrity Management Information System (ASIMIS). ASIMIS is responsible for collecting and editing aircraft in-flight usage parameters from all US Air Force fleets world-wide. The information resulting from this activity is the basis for ASIP events. Approximately 15 percent of the aircraft in each fleet (fighter, bomber, cargo) have a multi-channel flight recorder installed on-board. In addition, all aircraft in some fleets have a normal load factor counting

accelerometer. The multi-channel flight recorder provides time-histories consisting of 15 to 24 flight parameters (depending on the aircraft). These parameters include airspeed, altitude, angle of attack, three axes' acceleration and rates, control surface deflection, outside air temperature, sink speed, and fuel flow. The data is categorized by mission type and segment.

The existence and continuance of ASIP and in particular ASIMIS has resulted in a large source of information detailing how the US Air Force uses and maintains their aircraft. This information is of great value to the US Air Force and could be of help to the commercial carriers in solving their aging aircraft problems.

3. US AIR FORCE RESOURCES FOR ADDRESSING AGING AIRCRAFT PROBLEMS

This section gives an overview of the facilities that the Wright Laboratory has to perform testing to address aging aircraft problems, the applicable research that has been and is being performed, and the analytical tools that are available to address aging aircraft issues.

3.1 Analysis Capability Specifically for Aging Aircraft

Recent research has been performed to specifically investigate the problems that aging aircraft experience. Among this research, a computer program named PROF (PRObability Of Fracture)⁽³⁾ was developed by the University of Dayton Research Institute under contract to the US Air Force. PROF is a VAX-based computer program written in FORTRAN. PROF is capable of determining the probability of fracture of a single aircraft or of any aircraft in a fleet that is subjected to a given usage. Inspection capabilities and intervals can be manipulated to yield the optimum inspection schedule for the specific aircraft or the fleet. PROF accounts for cracking in a metallic aircraft structure, and accounts for a distribution of crack sizes that may exist in a structural detail. PROF does not account for any corrosion or multiple site damage (interaction) effects.

Another FORTRAN based computer program⁽⁴⁾ has been developed which predicts the corrosion damage at various locations of interest on a USAF C-5A aircraft. This program makes use of environmental factors and the amount of time the aircraft spent at different Air Force Bases. The program produces recommended times for inspection and maintenance based on the time it takes for a crack to grow to half its critical length, the length of time it will take for an alloy to corrode to a depth of 0.0762 mm (.003 inch), and the optimum time for the scheduling of the next paint renewal.

3.2 Other Research Areas Applicable to Aging Aircraft

This section gives an overview of other areas of research that may be applicable to aging aircraft. The main benefits

from this research is targeted for other technical areas, however.

3.2.1 Smart Structures

The concept of a smart aerospace structure is an aircraft that is capable of the real-time detection and analysis of damage that is present in the aircraft structure. Continuing research is being performed in areas that can have direct impact on aging aircraft. Acoustic Emission (AE) techniques of determining the presence of structural damage are being pursued by military and civil aircraft researchers^(5,6,7). Physical Acoustics Corporation (PAC) of Princeton, NJ, is the leader in the field of using AE techniques for both commercial and military aircraft. For commercial aircraft, PAC has instrumented a Boeing 720B and a 707. Cracks were found on both Boeing aircraft using the AE techniques, including cracks that were missed when inspected 1,000 flight hours earlier. The tests on the Boeing aircraft took one hour to perform after each aircraft was fitted with the AE sensors. The tests were performed on the ground by pressurizing the cabin up to a gage pressure of 27.6 - 58.6 kPa (4-8.5 psi).

In the military arena, PAC has a network of 28 AE sensors attached to critical locations of an Air Force F-111 aircraft. This network of sensors is used as part of the F-111 structural cold proof test program that is performed at specific intervals at McClellan AFB. In this program, each F-111 airframe is cooled to a temperature of -40° C (-40° F) to significantly lower the critical crack size of the high strength steel that is used in critical locations. The sensors then detect any crack growth that occurs as the aircraft structure is statically loaded. The same system that is used on the F-111 is currently being tested on an F-15 full scale fatigue article being tested at the Structures Test Facility at Wright-Patterson AFB. For the F-15 fatigue test, the main areas of interest are the connecting lugs between the wings and the fuselage. Data are collected several times each week from the AE sensors and are reviewed for signs of crack initiation. Any areas that are suspected of containing cracks are then visually and ultrasonically inspected.

Research into the use of fiber optic sensors has greatly increased over the last few years. It is theorized that fiber optic based sensors will be able to yield strain, temperature, pressure, and structural damage information on a real-time basis. This type of setup would be ideal for new aircraft, as the sensors can be either embedded into the structure or simply attached by some mechanical means during aircraft assembly. For aging aircraft, this type of sensor could be attached during a retrofit. In either event, these sensors would be able to warn the pilot and maintenance crew of possible problem areas where maintenance is needed and could give a real-time assessment of battle damage.

The feasibility of using fiber optic sensors for strain measurement on air-

craft has been proven by a number of tests that were performed in the laboratory and three tests that were performed on the same full scale F-15 fatigue test that is being used for the AE testing⁽⁶⁾. All sensors that were used on the F-15 fatigue article were tested under both static and dynamic flight spectrum loading. The strains recorded using the fiber optic sensors matched those that were recorded using conventional electrical strain gages, thus demonstrating their feasibility.

3.2.2 Structural Damage Assessment via Finite Element Analysis

Vulnerability Analysis of Aerospace Structures Exposed to Lasers (VAASEL) is a laser vulnerability analysis code developed for the Air Force by Northrop Corporation. The code is maintained by the Structures Division of the Flight Dynamics Directorate, Wright Laboratory and is available for distribution to contractors and other government agencies. This code was developed primarily to assess the vulnerability and survivability of aerospace structures subjected to laser threats.

To assess survivability and vulnerability, VAASEL has been integrated with a large finite element structural analysis program named ASTROS (Automated Structural Optimization System.) ASTROS is basically used for preliminary design of aerospace structures. Both VAASEL and ASTROS are compatible with NASTRAN. VAASEL consists of six independent engineering modules that are linked together by an executive system. This modular architecture provides VAASEL with the flexibility to do many types of structural analyses. The structural analysis module is based on the ASTROS structural analysis capabilities that were modified to include material nonlinear analysis. This module incorporates commonly used finite elements such as rods, beams, quadrilateral membrane, bending and solid elements. The air loads analysis module assesses the ability of a damaged structure to withstand the design flight conditions. An advanced paneling method, based on the computer program USSAERO, simulates the aerodynamic loads for all flight regimes. The failure analysis module determines the structural vulnerability to a given damage level and identifies critical areas of the structure. All possible failure modes are accounted for and the criteria used to predict these failures are material dependent. Different sets of failure criteria are included for composites, metals and ceramics. The modular nature of the VAASEL architecture permits easy code modification to include other user-defined potential failure models and criteria. VAASEL is currently designed to run on the VAX/VMS operating system.

3.2.3 Gust Research

The Structures Division of the Flight Dynamics Directorate has an on-going investigation to become familiar with the gust analysis referred to as the Statistical Discrete Gust (SDG) method. This

method is being proposed internationally for the certification of aircraft for flight through continuous turbulence. The significant deficiency in the present gust analysis is the inability to rationally include the effects of a highly nonlinear flight control system. SDG can fulfill this requirement. In conjunction with this in-house study, Boeing Aircraft Company is performing a contract to apply SDG to several aircraft. A rigid aircraft analysis will be performed with pitch and plunge degrees of freedom. The advantages and disadvantages of the SDG method will be identified and the merits of the SDG procedure will be compared with other emerging methods. The Structures Division has been the lead in the US Air Force for many years in the area of discrete gusts and turbulence effects on aircraft. In addition to having the recognized experts in this area, this Division has a large library of reference reports.

4. FACILITIES FOR TESTING AGING AIRCRAFT

This section describes the facilities of the Wright Laboratory that are being used or that have the potential for use to address the problems associated with aging aircraft.

4.1 Structures Test Facility

This facility is part of the Structures Division, Flight Dynamics Directorate, Wright Laboratory. It provides the Air Force with a unique in-house capability for experimentally determining the structural integrity and reliability of aircraft primary and secondary structural components and advanced aerospace structural concepts. Emphasis is on implementation of the "full spectrum" concept by providing test support to research, exploratory development, and advanced development.

The test capability of the facility covers the full spectrum from basic evaluation of fracture and fatigue characteristics of joining techniques through intermediate scale testing of conceptual structures and major aircraft subassemblies to full scale testing of complete flight vehicles. This testing is accomplished at cryogenic (-199° C, -326° F), ambient, and elevated (up to 1,760° C, 3,200° F) temperatures as required by each individual test program. The testing of composite, metallic, and hybrid structures are all readily accomplished by the facility. Test loads simulation is accomplished by the use of electronically controlled hydraulic power systems. Hydraulic capability is in excess of 4,920 liters/minute at 20.68 MPa (1,300 GPM at 3,000 psi). Microcomputers and microprocessors along with over 150 analog servo controllers and other special purpose equipment are used to generate test functions and control the application of loads to the test structures. Elevated temperature testing is accomplished by the facility's radiant heating systems. A total capability of 50,400 KW of electrical power is available for short durations within the facility. The range of temperature control is from

ambient to 1,760° C (3,200° F) on large surface areas. Nuclear effects and other special simulation based on flux density rather than temperature can also be performed.

Measurement data from load cells, strain gages, deflection transducers, thermocouples, flux sensors, and other instrumentation devices are acquired from test specimens. The data are then conditioned, converted from analog to digital form, and processed for real-time evaluation by the Alpha Numeric/Graphic Display and for post-test hard copy output. The systems that perform these functions include over 2,000 data channels, five computers, and a number of displays and related equipment.

4.2 Fatigue and Fracture Facility

This facility develops fatigue, crack growth, and structural life prediction methods and experimental data to support the USAF Aircraft Structural Integrity Program. The analytical applications and experimental validation of these methods are accomplished through direct application to aircraft structures, including involvement in durability and damage tolerance problems associated with new and existing systems. The developed methods provide a basis for assessing the effect of variables such as material properties, structural configurations, usage, and loading environment on the strength and durability of airframe structures. These technologies apply to the design, development, operation, and maintenance of aerospace structures in a safe, effective, and economical manner.

4.3 Structural Vibration Facility

The Structural Vibration Branch of the Structures Division performs research to develop advanced vibration prediction methods, control systems to minimize vibration, and aircraft ground vibration testing techniques. Also, new data acquisition and analysis instrumentation and equipment are developed to provide design data for new systems and rapid solutions for existing problems. Data is measured and analyzed covering system ground and flight dynamics phenomena such as vibration, noise, flutter, and other induced loads. Activities of this Branch have a high potential for application to the Aging Aircraft problem.

5. EXAMPLE APPLICATION OF AVAILABLE TECHNOLOGY

The existing technology base can be used to address problems of aging aircraft. For example, in managing the airframe structures in a fleet of aging aircraft, decisions must be made concerning the timing and extent of inspections, repairs, modifications and life extension options. There are significant safety and cost ramifications of these decisions and every possible tool that can assist in making cost effective decisions should be used. The Probability Of Fracture (PROF) risk analysis computer program was written to provide such a tool for appli-

cation in the management of US Air Force structures. This section describes PROF and presents an example of its use on a transport/bomber aircraft.

5.1 Objective of Structural Risk Analysis

To date, the United States Air Force has applied the durability and damage tolerance requirements of MIL-STD-1530A in three areas: 1) designing new aircraft, 2) evaluating the durability and damage tolerance characteristics of aircraft which were designed prior to the current requirements of MIL-STD-1530A, and 3) evaluating structural repairs and modifications. These applications have used fracture mechanics principles in a deterministic manner. That is, flaw growth was predicted using a fixed potential flaw size, a fixed da/dN vs ΔK relationship, and a stress spectrum derived from a predicted average usage. While it was realized that there are many stochastic elements in the initiation and growth of cracks, the applied process was considered conservative. Initial flaw size assumptions were generally severe, tracking programs accounted for variations in usage severity, and inspections (if necessary) were scheduled at half the time required for the specified initial flaws to grow to a critical size. At the time of the assessments, there was a high (but unquantified) degree of assurance that fatigue failures and widespread cracking would not occur within the design operational lifetime. However, the realized life of individual airframes is seldom equal to the design life. The retirement age of an aircraft fleet is determined more by its inherent operational capability and maintenance costs than by the number of flight hours specified during conceptual planning. As the population of airframes ages, fatigue cracks will initiate and grow and all of the stochastic elements which can influence the fatigue process will be active. The exact status of fatigue cracks in any specific airframe will be somewhat unknown even after inspections due to the uncertainties of inspection processes. In an aging fleet, structural damage is stochastic in nature.

PROF is a risk analysis computer code whose objective is to stochastically assess structural integrity. The assessment is made in terms of both safety (as quantified by the probability of fracture of a population of structural details) and durability (as quantified by the expected number and sizes of the cracks requiring repair that will be detected at an inspection). This characterization of structural integrity can be applied as an additional tool in making decisions concerning the timing of inspection, replacement, and retirement maintenance actions. The following paragraphs describe the methodology that has been implemented in PROF and presents an example of its potential application to a representative scenario.

5.2 Air Force Data Base

Implementing structural risk analyses involves compromises between the ability to model reality and the data that is available to feed analytical models. In general, the more detail that is required by the model, the less reliable the available data is. Because of the Aircraft Structural Integrity Program (ASIP) requirements of MIL-STD-1530A, the Air Force has an extensive data base on each system for the deterministic evaluation of structural integrity. Of particular application to risk analysis are the data associated with the damage tolerance^(9,10) and durability^(9,11) analyses that are performed for all potential airframe cracking sites and the data associated with the force management tasks of ASIP⁽¹²⁾.

5.3 Risk Analysis Methodology

The risk analyses model (PROF) is applicable to a population of structural elements which is defined in terms of details which experience essentially equivalent stress histories. Because of the available data, the model was constructed around the growth of a distribution of crack sizes as illustrated in Figure 4^(13,14). The distribution of crack sizes to initiate the analysis would be estimated from the best available data. These data could be provided by the compilation of information from routine inspections, teardown inspections, equivalent initial crack size distributions obtained in a manner previously described, or engineering judgement.

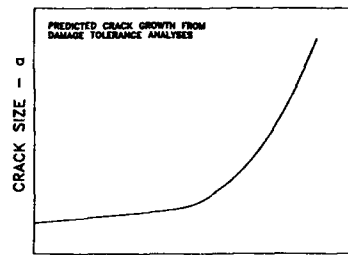
5.3.1 Modeling the Crack Size Distribution

There are two basic crack population calculations: 1) growing the distribution of cracks from a beginning reference time to an arbitrary time within a period of uninterrupted usage, and 2) quantifying the effect of the inspection and repair. These calculations are addressed in the following paragraphs.

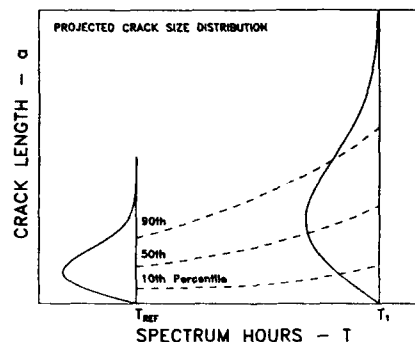
5.3.1.1 Growing Population of Crack Sizes

Given an initial distribution of crack sizes at a reference time, T_r , the program estimates the distribution of crack sizes at $T_r + \Delta T$ flight hours by projecting the percentiles of the initial crack size distribution using the deterministic crack growth versus flight hours relation of the damage tolerance analysis. This calculation is performed in PROF by table look-up. Figure 5 is a schematic of this process. The analytical formulation of the process is as follows.

Let $a_p(T_r)$ represent the p th percentile of the crack size distribution at T_r flight hours, i.e., $P[a < a_p(T_r)] = p$. Let $a = \phi(T)$ represent the a versus T relation (defined for PROF by a table of (a, T) data pairs). Then the p th percentile of the crack size distribution at time $T_r + \Delta T$ is given by

$$a_p(T_r + \Delta T) = \phi[\phi^{-1}[a_p(T_r)] + \Delta T] \quad (1)$$


a) Deterministic a versus T



b) Projections of Crack Size Percentiles

Figure 4. Crack Size Distribution versus Spectrum Hours

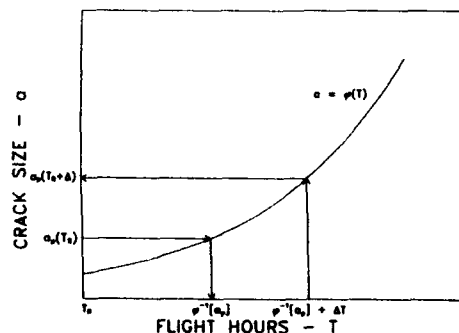


Figure 5. Projection of Percentiles of Crack Size Distribution

This calculation is repeated for all percentiles in the table and defines the crack size distribution.

5.3.1.2 Maintenance Effect on Crack Size Distribution

At a maintenance action, the population of details are inspected and all detected

cracks are repaired. The maintenance action will change the crack size distribution. This change is a function of the inspection capability and the quality of repair as is shown in Figure 6. Inspection capability is modeled in terms of the probability of detection as a function of crack size, $POD(a)$ ⁽¹³⁾. Repair quality is expressed in terms of the equivalent repair crack size distribution, $f_r(a)$. If $f_{before}(a)$ and $f_{after}(a)$ represent the density function of crack sizes in the population of structural details before and after a maintenance action, then

$$f_{after}(a) = P \cdot f_r(a) + [1 - POD(a)] \cdot f_{before}(a) \quad (2)$$

where P is the percentage of cracks that will be detected

$$P = \int_0^{\infty} POD(a) \cdot f_{before}(a) da \quad (3)$$

The post maintenance crack size distribution, $f_{after}(a)$, is then projected forward for the next interval of uninspected usage. The process is continued for as many inspection intervals as is desired.

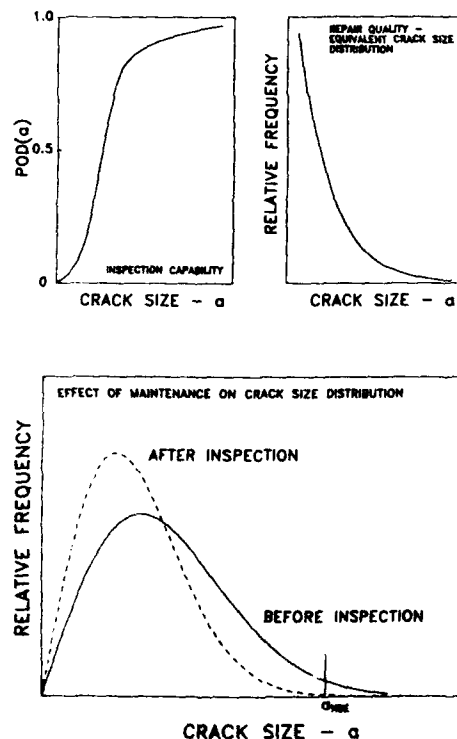


Figure 6. Effect of Maintenance on Crack Size Distribution

5.3.2 Probability of Fracture

Safety is quantified in terms of the probability of fracture (POF) due to the maximum stress encountered in a flight. POF is calculated as the probability that

the maximum stress encountered in a flight will produce a stress intensity factor that exceeds the fracture toughness for a structural detail as shown in Figure 7. This calculation is performed in two contexts. The single flight POF is the probability of fracture in the flight given that the detail has not fractured previously. This number can be compared to other single event types of risks, such as the risk of death in an automobile accident in an hour of driving. The interval probability is the probability of fracture at any flight between the start of an analysis (reference time of zero or after a maintenance action) and the number of spectrum hours, T . This POF is useful in predicting the expected fractures in a fleet of aircraft in an interval and is required for the expected costs associated with a maintenance schedule.

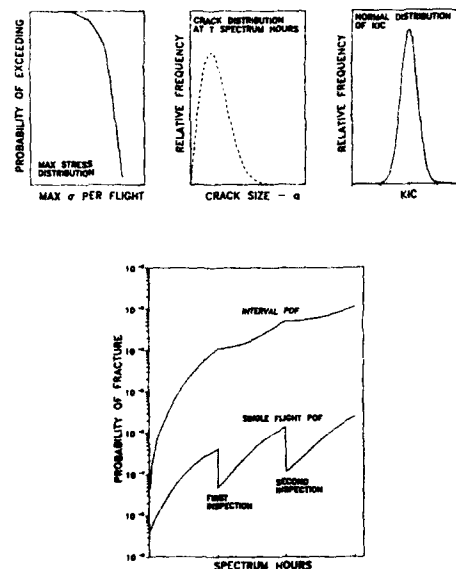


Figure 7. Quantifying Safety in Terms of Probability of Fracture

5.3.2.1 Single Flight Probability of Fracture

The equation for calculating the probability of fracture at a single stress raiser in a single flight at T hours is given by

$POF_s(T)$ = Single element POF during flight at T hours

$$= P(\sigma_{max} > \sigma_{cr}(a, K_c))$$

$$= \int_0^{\infty} \int_0^{\infty} f_T(a) \cdot g(K_c) \cdot \bar{H}(\sigma_{cr}(a, K_c)) dK_c da \quad (4)$$

where

$f_1(a)$ = probability density function of crack sizes at T flight hours;

$g(K_c)$ = probability density function of the fracture toughness of the material; and

$$\bar{H}(\sigma_{cr}(a, K_c)) = P[\sigma_{max} > K_c / \sqrt{\pi a} \beta(a)],$$

i.e., the probability the maximum stress in the flight exceeds the critical stress given " a " and K_c .

The single element POF, $POF_s(T)$, is interpreted as the probability that one of the elements in an airframe with T equivalent flight hours will experience a fracture due to a combination of crack size, fracture toughness, and stress. This calculation is based on the assumption that the size of the crack in the stress raiser of the element and the fracture toughness are independent.

To calculate the single flight probability of fracture from any one of the k equivalent elements (stress raisers) in a single airframe at T flight hours, $POF_s(T)$, it is assumed that the fracture probabilities between elements are independent. Then

$$POF_s(T) = 1 - [1 - POF_s(T)]^k \quad (5)$$

$$= k \cdot POF_s(T)$$

Similarly, $POF_r(T)$, the probability of a fracture in any of the N airframes in the fleet as they age through T flight hours, is calculated as

$$POF_r(T) = 1 - [1 - POF_s(T)]^N \quad (6)$$

$$= N \cdot POF_s(T)$$

All three of these single flight POFs are calculated at ten equally spaced increments in each usage interval. The results are printed in a summary output report.

5.3.2.2 Interval Probability of Fracture

Fracture can result during any flight in a usage period and the probability of a fracture during an entire period is required to estimate the expected costs of a fracture. Since the fracture toughness of an element does not change from flight to flight, single flight POFs as obtained above cannot be combined to obtain interval POF. The assumption of independence needed to make this calculation possible is not valid.

An approach to estimating interval POF which accounts for the constancy of fracture toughness over the interval was formulated as follows: 1) determine the contribution to the total POF from each possible pairing of fracture toughness and crack size at the beginning of the usage interval, $PF(a, K_c)$; 2) weight each contribution by the probability of the crack size-fracture toughness combination, $f(a) da \cdot g(K_c) dK_c$; and 3) sum the weighted contributions over all possible combinations of crack size and fracture

toughness. To calculate the contribution to the total POF from a crack size-fracture toughness pair, the total usage interval is divided into m subintervals. It is assumed that the crack size is essentially constant in a subinterval and the critical stress is calculated for the crack size of the subinterval and the fracture toughness. The distribution of maximum stresses in a subinterval is calculated from the distribution of maximum stresses in a flight and the probability of fracture in a subinterval is the probability that the maximum stress exceeds the critical stress for the subinterval. The POFs from the subintervals are combined to obtain the POF to the total usage interval for the initial crack-fracture toughness combination.

The interval POF process is implemented mathematically by the equation:

$$POF_i(I_j) = \int_0^{\infty} f_j(a) \int_0^{\infty} g(K_c) \cdot PF(a, K_c) dK_c da \quad (7)$$

where

$POF_i(I_j)$ = probability of fracture at a single stress raiser in the j th usage interval;

$f_j(a)$ = probability density function of crack sizes at the start of the j th analysis interval;

$g(K_c)$ = probability density function of critical stress intensity factors for the structural detail;

$$PF(a, K_c) = 1 - \prod_{i=1}^m H_{\Delta T}[\sigma_{cr}(a(T_i), K_c)];$$

$H_{\Delta T}[\sigma_{cr}(a(T_i), K_c)]$ = probability that the maximum stress in ΔT flights is less than the critical stress;

$$= H[\sigma_{cr}(a(T_i), K_c)]^{\Delta T};$$

$H(\sigma)$ = Gumbel distribution of max stress per flight;

$$\sigma_{cr}(a(T_i), K_c) = K_c / \sqrt{\pi \cdot a(T_i)} \cdot \beta(a(T_i));$$

ΔT = number of flights in a subinterval;

$$T_i = i \cdot \Delta T, \quad i = 1, \dots, m.$$

Since the computation time to implement Equation (7) is significant and depends on the number of subintervals, the number of flights allocated to a subinterval is a trade-off between accuracy (change of crack size in the subinterval) and computer time. Crack growth per flight is relatively slow over most of the crack sizes in the crack size distribution and long usage intervals imply slow crack growth per flight. Therefore, the number of flights in a subinterval was based on the total time in a usage interval as follows:

$$\begin{aligned} 0 &< m \cdot \Delta T \leq 1,000, \Delta T = 10 \\ 1,000 &< m \cdot \Delta T \leq 2,000, \Delta T = 20 \\ 2,000 &< m \cdot \Delta T \leq 3,000, \Delta T = 30 \\ &\text{etc.} \end{aligned}$$

The sensitivity of the interval POF to this method for determining the number of flights in a subinterval was evaluated. It was concluded that changes in the interval POF from using smaller subintervals would be insignificant. Interval fracture probabilities for the aircraft and for the fleet are calculated using equations analogous to Equations (5) and (6), respectively.

5.3.3 Expected Maintenance Costs

Given the predicted crack size distribution of an inspect/repair maintenance action and the $POD(a)$ function at the time T_i , the expected number and sizes of the cracks that will be detected can be calculated. In particular, PROF calculates the cumulative proportion of cracks that will be detected as a function of crack size as

$$P(a_i) = \int_0^{a_i} POD(a) \cdot f_{before}(a) da \quad (8)$$

The proportion of detected cracks in the arbitrary range defined by $\Delta a_i = a_{i+1} - a_i$ is given by

$$P(\Delta a_i) = P(a_{i+1}) - P(a_i) \quad (9)$$

Expected costs of maintenance are not calculated in PROF. However, PROF output can be used to estimate the expected costs of a maintenance scenario (as defined by flight hours between inspections, inspection capability, and repair quality). If the total population being modeled comprises k details in each of N airframes, then the expected number of cracks to be repaired at T_i between sizes a_i and a_{i+1} is $k \cdot N \cdot P(\Delta a_i)$. If C_i represents the cost of repairing a crack in size range i , C_f represents the cost of a fracture, and I represents the cost of inspecting each detail, then the expected costs of fracture and repairs in the usage interval are given by

$$E_j(C) = POD(T_j) \cdot N \cdot C_f + k \cdot N \cdot \left[I + \sum_i P(\Delta a_i) \cdot C_i \right] \quad (10)$$

Summing over usage intervals (maintenance periods) yields the total expected maintenance costs. The schematic of Figure 8 illustrates the cost calculations. In this hypothetical example, expected maintenance costs were estimated for a range of inspection intervals.

Table 1 summarizes the input data required by the risk analysis program. Crack size distributions of the cracks which were detected and repaired are output immediately before and after each maintenance action. These crack size distributions can be coupled with cost data to obtain expected maintenance cost information.

5.4 Example Application

To illustrate the application of the risk analysis computer code, representative data for an aging military transport/bomber were used to evaluate the timing

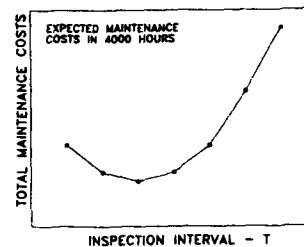
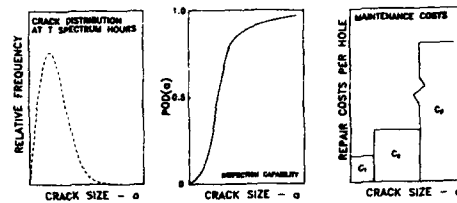


Figure 8. Schematic of Expected Maintenance Costs

DATA TYPE	FORMAT	SOURCE/COMMENT
MATERIAL/GEOMETRY	K/σ vs a	File DTA analysis - stress intensity factor coefficient
	$g(K_{IC})$	Parameter values Normal distribution of fracture toughness
AIRCRAFT/USAGE	Locations	Constants Number of analysis locations per airframe and number of airframes in the fleet
	$f_0(a)$	File Crack size distribution at start of analysis
	a versus T	File DTA analysis - crack growth life curve
	$h(a)$	Parameter values Gumbel distribution of max stress per flight - from L/ESS data or sequences of DTA analysis
INSPECTION/REPAIR	$T_1, T_2, \dots$	Constants Inspection times - user defined
	$POD(a)$	Parameter values Cumulative lognormal POD function for MDE system
	$f_r(a)$	File Crack size distribution of repaired crack sites
	C_i	Constants Costs of inspections and crack repair for ranges of crack sizes

Table 1. Summary of PROF Input Data

of inspections and the capability of the inspection method. In particular, the objectives of the analyses were: 1) to seek the most cost effective inspection intervals for a population of structural details, and 2) to determine if it is cost effective to use a better but more expensive inspection method. The assumptions that there are 75 aircraft in the fleet which experience the same expected operational usage and that all of the aircraft have undergone maintenance at a

fixed reference number of flight hours were made. The risk analysis pertains to periods of operational usage (or inspection or maintenance intervals) after this reference age.

5.4.1 Baseline Input

The assumed population of structural details comprises rows of fastener holes in a fail-safe zone of equivalent stress experience on the upper rear fuselage. Figure 9 presents a schematic of the holes in the region and the geometry correction for crack growth calculations. The critical crack size is approximately 2.50 cm (0.986 inch). Cracks that are detected before fracture can be repaired by a patch. The assumption was made that each airframe contains 50 separate regions such that the repair patch for any single crack in a region repairs all of the cracks in the region. However, if fracture (uncontrolled rapid crack growth) occurs, the entire panel must be replaced. The fracture toughness of the 7079-T6 aluminum alloy has an average value of $97.2 \text{ MPa}\sqrt{\text{m}}$ ($88.4 \text{ KSI}\sqrt{\text{inch}}$) with a standard deviation of $4.8 \text{ MPa}\sqrt{\text{m}}$ ($4.4 \text{ KSI}\sqrt{\text{inch}}$) [14].

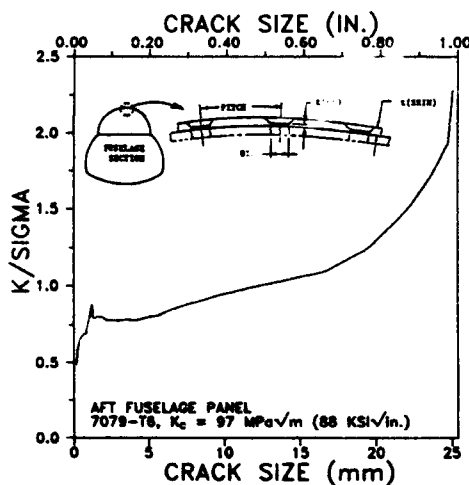


Figure 9. Stress Intensity Factor Geometry Correction for Analysis Region

Figure 10 presents the projection of crack growth from a flight-by-flight spectrum of planned mission usage for the fleet. For the visual inspections of the region of interest, the reliably detected crack size was assumed to be 5.59 mm (0.220 inch). Under US Air Force guidelines for establishing inspection intervals, subsequent inspections would be set at one half the time required for a crack of the reliably detectable size to grow to its critical size. For the example application, the baseline damage tolerance re-inspection interval was set at 7,200 flight hours. The Gumbel distribu-

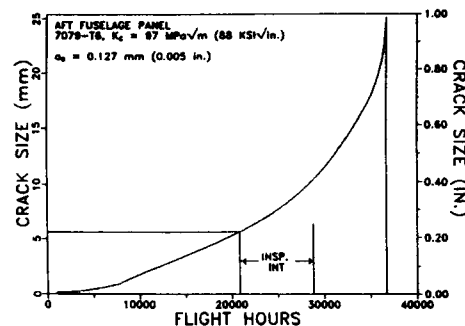


Figure 10. Crack Growth for Projected Usage Spectrum

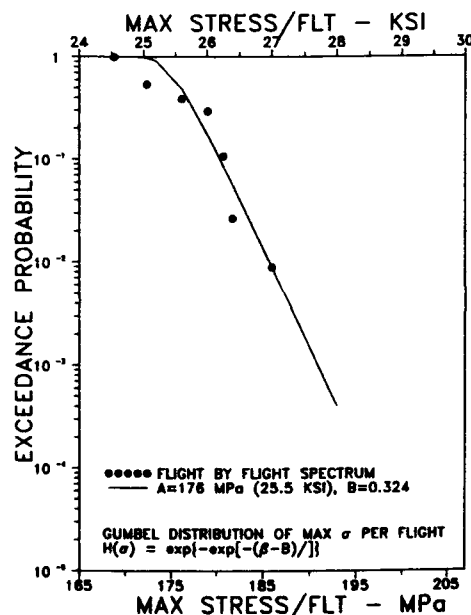


Figure 11. Gumbel Distribution Fit to Maximum Stress per Flight of Projected Spectrum

tion fit to the maximum stress per flight curve of the flight-by-flight stress spectrum is shown in Figure 11.

At the start of the analysis (reference time of zero), it was assumed that the distribution of the largest cracks in each region was described by a Weibull distribution with a scale parameter of 0.151 mm (0.006 inch) and a shape parameter of 0.768. For this distribution, 1 in 1,000 of the holes will have cracks larger than 1.905 mm (0.075 inch) and 3 in 10,000 will have cracks larger than 2.54 mm (0.100 inch). Cracks are repaired by patches and it is assumed that the repair quality of a patch is described by a uniform distribution of

equivalent crack sizes on the interval 0 to 1.27 mm (0 to 0.050 inch). That is, a patch replaces the largest crack in the patched region with an equivalent flaw that is equally likely to be any size between 0 and 1.27 mm (0.050 inch).

For the baseline analysis, the reliably detected crack size of 5.59 mm (0.220 inch) is assumed to be the result of a close visual inspection only. This capability is interpreted as a 90 percent detection capability at 5.59 mm (0.220 inch). Because of the fastener heads, no crack smaller than 2.54 mm (0.100 inch) could be detected, i.e., $POD(a)=0$ for $a \leq 2.54$ mm (0.100 inch). To complete the definition of the $POD(a)$ function, it was also assumed that a 3.81 mm (0.150 inch) crack would be detected half of the time. The cumulative lognormal $POD(a)$ function that meets these specifications is shown in Figure 12. Also shown in Figure 12 is the $POD(a)$ function for a potential eddy current inspection system with a smaller reliably detected crack size (to be discussed in Subsection 5.4.3).

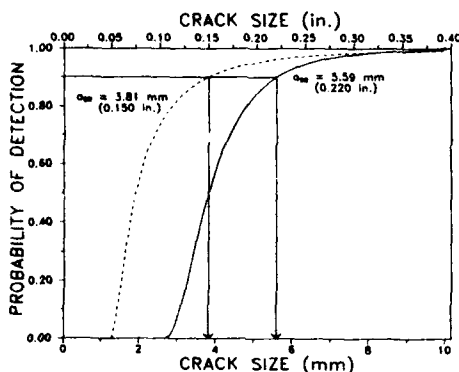


Figure 12. Crack Detection Probability for Competing Inspection Methods

Because of the comparative nature of the analysis objectives, inspection and repair costs need only be specified on a relative basis. For baseline analyses, it was assumed that the cost of the visual inspection of each region is one unit, the cost of patching the region is 100 units, and the cost of replacing a fractured panel is 100,000 units. Expected costs for different maintenance scenarios are normalized in terms of the total expected costs for the baseline inspection interval (7,200 hours) and inspection capability.

5.4.2 Inspection Interval Analysis

The probability of fracture (POF) for any one of the 50 panels on a fuselage under the baseline conditions is presented as a function of spectrum hours in Figure 13. The solid line represents the fracture probability during a single flight and the dashed line (circles) represent the probability of a fracture in any panel of

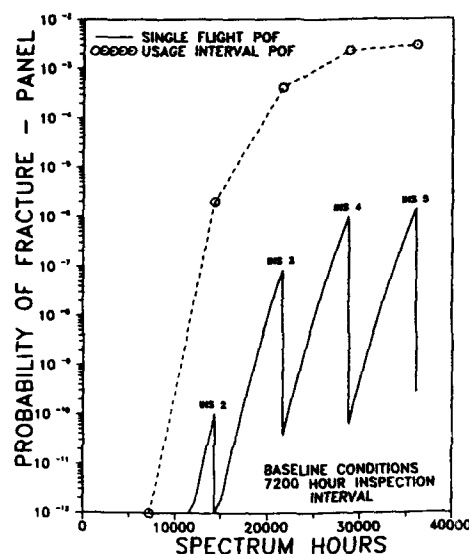


Figure 13. Probability of Panel Fracture in an Airframe for Baseline Conditions

an airframe at any time during the previous usage period. The large changes in single flight probability result from the removal of large cracks at the inspection/repair maintenance cycles and the growth of the population of cracks during the usage periods. PROF does not output fracture probabilities below 10^{-12} , so smaller POF values are plotted at this value. Since the structure under analysis is fail-safe and the costs are driven by the fracture probability in the entire usage period and the costs of maintenance, the single flight fracture probabilities will not be considered further.

To investigate the effect of a constant usage interval between inspections, a total analysis period of 36,000 hours was assumed. Equally spaced inspection intervals were then defined to provide from three to twelve inspections in the 36,000 hour period. Figure 14 presents the probability of fracture in each interval between maintenance (inspection and repair) actions for seven of the inspection intervals. The fracture probabilities display somewhat similar behavior in the early period during which the upper tail of the initial crack size distribution grows to potentially significant sizes. Following this initial period, the interval fracture probabilities tend to stabilize at distinct levels; the shorter the inspection interval, the lower the equilibrium fracture probability.

Because of the equilibrium POF levels, the expected costs associated with the possibility of panel fractures at the longer inspection intervals will be greater than those of the shorter intervals. On the other hand, the costs associated with the more frequent inspections may be greater than the expected costs of panel fracture. To evaluate the trade-off, the total expected maintenance

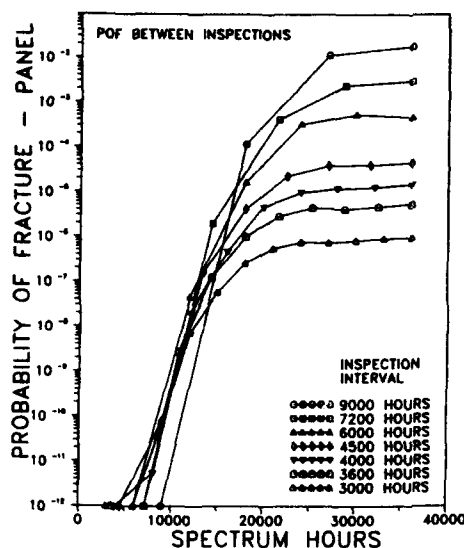


Figure 14. Probability of Panel Fracture in an Airframe Between Inspections for Selected Inspection Intervals

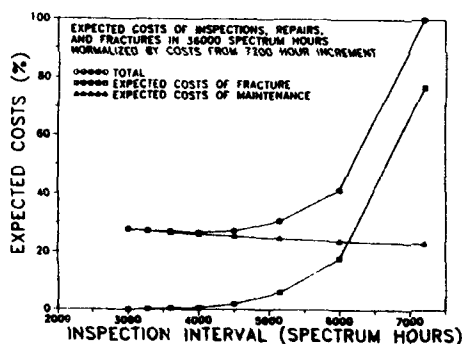


Figure 15. Normalized Expected Maintenance Costs as a Function of Inspection Interval

and fracture cost for each of the inspection intervals was calculated. These expected costs are presented as a function of inspection interval in Figure 15. As noted earlier, the costs are normalized by the total expected cost for the baseline inspection interval of 7,200 hours. (Inspection intervals of 9,000 and 12,000 hours were also analyzed but the expected total costs were, respectively, 4.1 and 25.0 times greater than those of the 7,200 hour increment. These intervals were not included in Figure 15 so that higher resolution could be shown for the shorter intervals.)

The expected total costs decrease with inspection interval down to about a 4,000 hour interval and then tend to increase slightly. The decrease is due to the large decrease in the expected costs

associated with panel fractures at the longer intervals. The equilibrium fracture probability for inspection intervals of 4,500 hours and less produce only minor additions to the total expected costs. The costs due to the inspections and repairs increase but at a very slight rate. From a practical viewpoint, any interval less than 4,500 hours would have essentially equivalent expected total costs.

To investigate the potential for reducing total costs by extending the timing of the first inspection, various combinations of initial inspection times and equal repeat inspection intervals thereafter were analyzed. Table 2 presents a summary of the expected normalized costs due to fracture, maintenance, and the total. As noted earlier, the expected maintenance costs were approximately equal for all scenarios considered. The expected costs due to panel fracture varied somewhat depending on the particular combination. It is interesting to note that the minimum expected total cost was achieved at a 16,000 hour first inspection followed by 4,000 hour intervals thereafter. The expected cost for this combination was slightly less than that of inspecting every 4,000 hours.

First Inspection (Hours)	Inspection Interval (Hours)	Fracture Cost %	Maintenance Cost %	Total Cost %
5143	5143	6.1	24.0	30.7
5000	5000	4.8	24.8	29.6
7200	4800	3.5	25.0	28.5
8400	4800	2.4	25.2	27.6
9000	5400	8.5	24.1	32.6
12000	4800	3.5	24.8	28.3
16000	4000	1.2	24.8	26.0
16000	5000	6.0	23.9	29.9
20000	4000	7.9	23.2	31.1

Table 2. Expected Total Fracture and Maintenance Costs as a Percentage of Total Costs for 7200 Hour Inspection Intervals

For the assumed conditions, the above analyses imply that an inspection schedule with shorter intervals would provide a significant savings in expected fracture and maintenance costs over those determined by the damage tolerance "rule". Although a minimum was achieved under the equal interval analysis, once the inspection interval was sufficiently short, the expected costs did not change significantly. This was true regardless of the timing of the first inspection. This latitude in setting inspection intervals could be important as the actual schedule should be determined by considering the many different populations of structural details in an airframe, each of which may have different optimum schedules.

5.4.3 Inspection Capability Analysis

It was assumed that the inspection for the baseline analysis was a close visual inspection that is inexpensive to perform. The question might arise as to whether it would be cost effective to perform a more expensive inspection with an attendant increase in capability. Toward this end, it was assumed that an eddy current (EC) inspection could be used to inspect for cracks in the regions and that the cost of the EC inspection is 10 times that of the visual. However, this reduces the reliably detected crack size to 3.81 mm (0.150 inch). Because the eddy current probe can detect cracks under the fastener head, it was assumed that the minimum detectable crack size is 1.27 mm (0.050 inch). The 50 percent detectable crack size was assumed to be 1.90 mm (0.075 inch). The cumulative lognormal POD(a) function that meets these requirements is shown in Figure 12 with the POD(a) function of the baseline analysis.

The usage interval fracture probabilities for the two inspection capabilities for a 4,000 hour inspection interval are presented in Figure 16. The eddy current inspection significantly reduces the chances of a panel fracture in the 36,000 hour period. When the expected maintenance costs are considered, however, the inspection and repair costs associated with the eddy current inspection are 2.2 times those of the visual inspection. At this 4,000 hour inspection interval, the expected costs due to panel fracture are small (almost negligible) for both inspection methods. However, the better (EC) inspection system apparently requires more cracks to be repaired at each of the inspections and these cracks are too small to be an imminent threat to the panel.

When the two inspection capabilities were analyzed at the 7,200 hour inspection interval, the reverse conclusion was drawn. The chances of panel fracture at the longer usage interval was sufficiently great that the total expected costs over the 36,000 hour period were significantly reduced by repairing the smaller cracks. This result was tested for sensitivity to the assumed inspection and repair costs. The expected total maintenance costs were obtained for ranges of cost per inspection and cost per patch. The expected total costs were still significantly less when the EC inspection costs were 50 times greater than those of the visual inspection and when repair costs were 500 times greater than those of the baseline calculations.

No clear conclusion can be drawn on the cost effectiveness of the EC inspection system as compared to the visual inspection. When the shorter and more cost effective intervals of this example are used, the visual inspection capability provides the more economical choice. If the longer damage tolerance defined inspection interval is to be used, the additional costs associated with the eddy current inspections would be justified. However, given an inspection interval, the risk analysis program, PROF, can be used to make a cost effective decision.

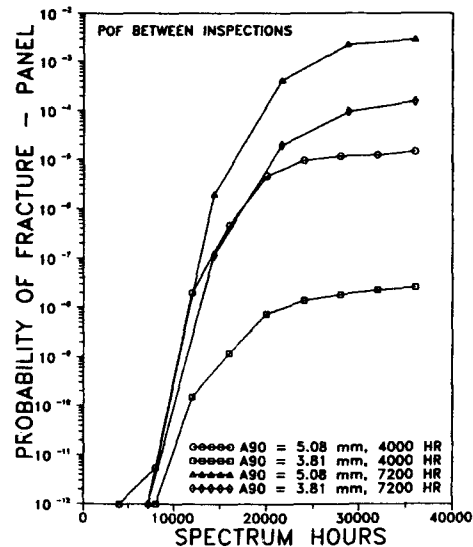


Figure 16. Probability of Panel Fracture in an Airframe Between Inspections for Different Methods and Intervals

6. RESEARCH AREAS REQUIRING ADDITIONAL DEVELOPMENT

Many areas require additional work to be able to better address the problems associated with aging aircraft. Three such areas are the modeling of corrosion in the structure, the modeling of multiple site damage, and the advancement of current Non-Destructive Inspection (NDI) techniques.

Risk analysis methods have been developed to account for populations of single flaws in the structure of an aircraft. However, risk analysis methods must be expanded to account for all significant problems that an aging aircraft experiences, including the effects of corrosion and multiple site damage. The analyses should also be capable of analyzing flight-by-flight and cycle-by-cycle loading and crack growth.

Corrosion models that exist are very simplistic in how they model fatigue crack growth. Additional research is needed to develop models that more accurately predict the initiation and growth of cracks that develop from corrosion pitting, and that account for the deteriorating structural properties due to corrosion.

NDI techniques must be advanced to the point where they can reliably detect flaws that are present under the heads of bolts and rivets and that can detect corrosion damage in hidden areas. They must be developed to a point where it is much less monotonous for the operator to operate and determine where a flaw or area under electro-chemical attack may exist.

The development of "Smart Structures" technology can do much in the area of aging aircraft. The development of advanced sensors and monitoring techniques could alert maintenance personnel of hidden problems, or of probable damage areas that should be inspected immediately.

7. CONCLUSIONS

In this paper, it is shown that the US Air Force does have the technology and up-and-coming research to address individual problems of aging aircraft. However, the technology that exists does not adequately address all major factors. The ultimate analysis would be one that addresses all major factors and their interactions. Current analyses must be extended or new analyses developed to predict the damage modes and states in the aircraft structure at any time. The analyses must be flexible enough to give accurate predictions for any aircraft, and must be simple enough to be used by any trained person.

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A PROBABILISTIC PROCEDURE FOR AIRCRAFT FLEET MANAGEMENT

by

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ABSTRACT

A procedure for determining the probability of structural failure of an airplane at any stage of the operational life is described. This procedure is based on existing models for representing the size distribution of pre-service cracks, the distribution of the maximum service load expected in a single flight and the reliability of the nondestructive method used for inspection. A case study is presented to show how the procedure can help the fleet manager set more realistic inspection and maintenance schedules and adopt more appropriate retirement policies.

1.0 INTRODUCTION

The flight safety of metallic airframes is currently ensured through damage tolerance analysis, based on fracture mechanics concepts. The deterministic nature of this approach implies that only average values of the influence parameters combined with a safety factor or "worst-bound" values are considered in the analysis. This sometimes gives unrealistic operational life assessments resulting in overly stringent inspection requirements.

The probabilistic approach to structural safety quantitatively assesses the risk of failure as a function of time by properly considering the relevant sources of variability such as the size of the initial flaws present at the critical locations, the crack propagation rate, the service loading spectrum, the residual strength and the crack detection capability of the nondestructive inspection system. Such an approach provides a more rational and realistic basis on which inspection and maintenance schedules can be established, at the onset of the operational life, as part of a service life extension program (SLEP) or following a mission change.

The Canadian Forces (CF) are currently supporting full-scale testing, in-service load monitoring and structural modification programs with the aim of extending the operational lives of its aging aircraft fleets into the twenty-first century. This will put emphasis

on timely maintenance decisions that will seek to minimize operational costs while maintaining flight safety. In that perspective, this paper deals with the possible application of probabilistic damage tolerance for fleet management purposes. A brief description is given of a procedure used to analyze a fleet of CF116 aircraft from the Canadian Forces. Results are presented and discussed in a management context. Special considerations in applying the procedure are also highlighted.

2.0 METHOD DESCRIPTION

The probabilistic method adopted in this study consists of the following basic steps:

- 1° Determine an equivalent initial flaw size (EIFS) distribution for each of the critical details to be considered.
- 2° Determine the service crack growth curves for propagating the EIFS distributions.
- 3° Derive the distribution of maximum stress applied at each critical location.
- 4° Obtain the probability of detection curve(s) representative of the NDI technique(s) used in service.
- 5° Calculate the probability of aircraft failure as a function of flight time by combining all critical details and considering the effects of repeat inspection.

These steps are only succinctly described in the next sections. Further details and complete mathematical developments can be obtained from other studies describing similar methods [1-4]. An existing computer program [4] was used for carrying out the probabilistic analysis. The program was modified to yield better accuracy, to account for the "renewal effect" of inspection, and to incorporate common statistical functions for representing the EIFS and maximum stress distributions as well as the log logistics model for representing inspection reliability.

2.1 Equivalent Initial Flaw Size Distribution

The initial fatigue quality of the airframe at each critical structural detail considered in the probabilistic crack growth analysis must be first assessed. Prior to service, these details already contain flaws of randomly distributed sizes that are likely to develop into fatigue cracks during operation. However, the size of the initial flaws cannot be economically determined using current NDT methods, given the high manufacturing quality of modern aircraft structures. For this reason, the initial fatigue quality of a given structural detail is rather quantified in terms of an EIFS statistical distribution. A commonly used procedure for deriving the EIFS distribution is described in [1] and illustrated in Figure 1. Briefly, fractographic data either from laboratory coupon testing or in-service inspections are used to derive a distribution of time to crack initiation (TTCI) for a detectable crack size, a_0 . The EIFS distribution is then obtained by back-extrapolating the TTCI distribution to time zero using a deterministic crack growth law. Parameters of the EIFS distribution can be optimized by means of data pooling of different fractographic data sets. Thus, an EIFS distribution represents an artificial distribution of initial flaws at time zero which corresponds to actual crack sizes at a later time for given crack growth conditions.

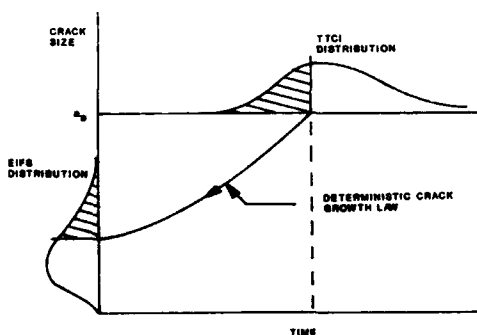


Figure 1 - Derivation of the EIFS Distribution

2.2 Crack Growth Master Curve

Once the EIFS distribution at each critical detail is determined, it is grown forward using a deterministic master curve, representative of the service condition, to obtain the crack size distribution $f_a(a(\tau))$ at any service time τ , as depicted in Figure 2. The master curve is defined either from available fractographic data or using a general crack growth analysis software and must be consistent with the derivation procedure for the EIFS distribution [1].

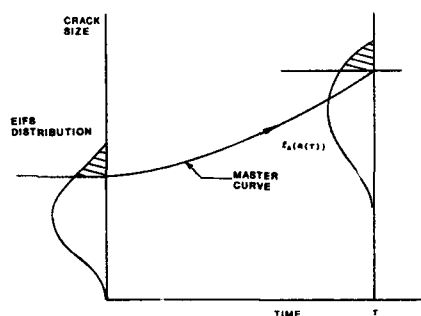


Figure 2 - Deterministic Crack Growth Using the Master Curve Approach

2.3 Inspection Reliability

In the deterministic approach to damage tolerance, an inspection schedule is prescribed at each critical location to detect the dominant fatigue crack before it reaches critical size. Periodic inspections are also required in the probabilistic approach to keep the probability of structural failure below an acceptable level. Inspection reliability must therefore be accounted for since it affects the crack distribution at each inspection.

The effectiveness of an inspection technique is usually represented in terms of a probability of detecting a given size crack. This probability is strongly dependent on the material, the inspection technique, the crack location and shape, the critical detail accessibility and geometry, the ability and attitude of the inspector, as well as the environment in which the inspection takes place. Experimental programs [5-7] have been conducted to generate probability of detection (POD) data for aircraft structures.

In general, the POD curves derived from

these data are best represented by the log logistics or log odds function given by

$$F_0(a) = \frac{\exp(\alpha + \beta \ln a)}{1 + \exp(\alpha + \beta \ln a)} \quad (1)$$

where the parameters α and β can be obtained by linear regression.

In general practice, a 95% lower confidence limit (CL) on the mean POD curve is determined through statistical analysis [8] and used to represent inspection reliability.

In the present methodology, the cracks detected during inspection are assumed to be repaired such that the cracked details are "reset" to the initial fatigue quality. Hence, the probability density function of the crack size immediately after inspection at $t = \tau^+$ becomes

$$f_A(a(\tau^+)) = P_0(\tau) f_A(a(t_0)) + (1 - P_0(\tau)) f_A(a(\tau)) \quad (2)$$

The first part of Equation (2) represented the detected cracks which have been repaired with $P_0(\tau)$, the probability of detecting a crack of any size at $t = \tau$, given by

$$P_0(\tau) = \int_0^{\infty} f_A(a(\tau)) F_0(a) da \quad (3)$$

The second part represents the undetected cracks with $(1 - P_0(a))$ being the probability of missing a crack of size a . The new function $f_A(a(\tau^+))$ is then grown forward using the master curve until a subsequent inspection modifies it.

2.4 Maximum Stress Distribution

The next step in the methodology involves the determination of the probability of exceeding a given stress at a critical location in a single flight. A simple procedure proposed in [9] is followed herein. First, the stress exceedance function used to derive the master curve is factorized to obtain the number of exceedances per flight. Next, truncation is carried out to keep only the maximum stress values exceeded no more than once per flight. Finally, an extrapolation is made to the high loads of rare occurrence by fitting a statistical function that closely represent the shape of the tail of the exceedance curve. The resulting function defines the cumulative probability of exceeding a given stress in a single flight whose derivative gives the maximum stress probability

density function.

2.5 Probability of Failure Determination

The variation of the probability of failure (POF) with time constitutes the most important result from a probabilistic damage tolerance analysis. Based on this information, appropriate actions can be taken to preclude aircraft fatigue failures that may have catastrophic consequences.

The POF at a given flight can be calculated from the joint probability of maximum stress and crack length that are assumed to be independent statistical variables. Thus, the POF for the j^{th} flight at the k^{th} critical location is expressed as

$$POF_{jk} = \int_{R_j} \int_0^{\infty} [f_A(a)]_{jk} [f_s(s)]_k da ds \quad (4)$$

where $f_A(a)$ and $f_s(s)$ are the probability density functions of the crack length and maximum stress, respectively, and R_j is a failure region in the $a-s$ plane defined by the residual strength curve as shown in Figure 3. Hence, the overall probability of failure after N flights for an aircraft structure containing M critical locations is given by

$$POF_T = 1 - \prod_{j=1}^N \{1 - [1 - \prod_{k=1}^M (1 - POF_{jk})]\} \quad (5)$$

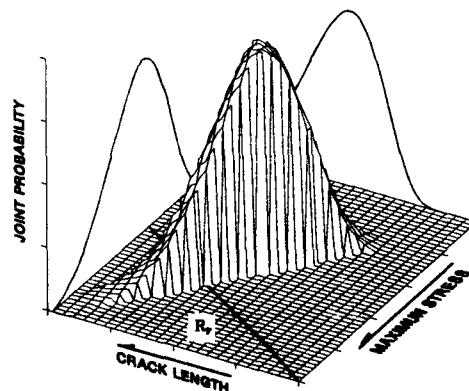


Figure 3 - Probability of Failure Determination

3.0 PRACTICAL APPLICATION

The CF116 fleet of the Canadian Forces was chosen to assess the applicability of the above probabilistic procedure for fleet management. Since 1985, the CF116 is mainly used as a primary trainer for the CF-18 fighter aircraft. A service life extension program (SLEP) was launched with the aim of extending the operational life of the aircraft from 4000 hours to 6000 hours. A wing change after approximately 3000 flying hours has been prescribed as a more economical alternative to repairing the fatigue-corrosion damage occurring on the wing spars. In addition, a change of material from Al 7075-T651 to Al 7475-T651 was recommended for the lower wing skin to improve toughness. However, some of the new wings will still incorporate Al 7075 lower skins since they had already been fabricated before the material change was adopted.

Only one critical detail was considered in this analysis, namely, the lower wing skin radius. A nondestructive inspection of the radius is currently carried out using the eddy-current technique. From a damage tolerance viewpoint, the radius is classified as a slow crack growth, damage tolerant critical location.

The geometry of the wing skin in the radius area is shown in Figure 4. The relevant mechanical properties of aluminum alloys 7075-T651 and 7475-T651 are given in Table 1.

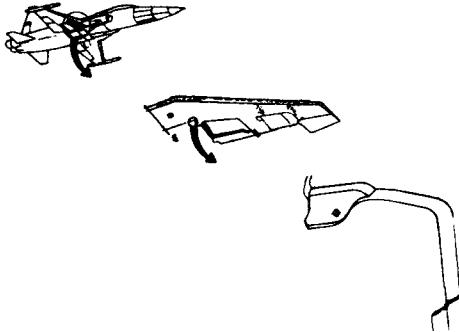


Figure 4 - CF116 Lower Wing Skin Radius

Table 1 - Mechanical Properties of the CF116 Lower Wing Skin Materials

PROPERTIES	7075-T651	7475-T651
Tensile Strength (MPa)	537.6	538.6
Yield Strength (MPa)	482.4	482.4
Fracture Toughness (MPa $\sqrt{m}$)		
- Plain Strain	30.0	45.3
- Plain Stress	67.8	91.4

Since no fractographic data were available, the Weibull-compatible distribution was chosen to characterize the initial fatigue quality because of its successful application in probabilistic durability analysis [2]. The functional form of the distribution is given by

$$f_A(a(t_0)) = \frac{\alpha}{\phi a} [\ln(a_{max}/a)/\phi]^{n-1} \cdot \exp\{-[\ln(a_{max}/a)/\phi]^n\} = 0 \quad (6)$$

$$a > a_{max}$$

Values for the distribution parameters were taken from [10]: $\alpha=1.823$, $\phi=1.455$, $a_{max}=0.762$ mm. A plot of the EIFS distribution is shown in Figure 5.

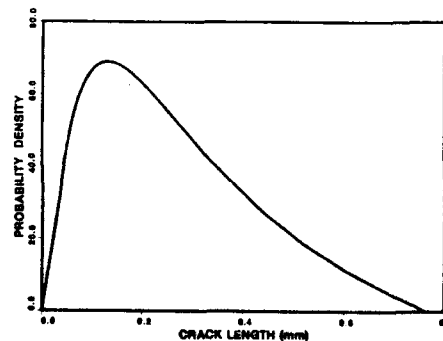


Figure 5 - Assumed Equivalent Initial Flaw Size Distribution

A quarter circular corner crack was assumed to develop at the radius and to remain quarter circular until thickness break-through occurred. From then on, a through-thickness crack with a straight crack front was assumed. The boundary correction factor for this crack propagation case is shown in Figure 6. The stress spectrum derived for the full-scale test on the structure was selected as representative of the fleet average usage as shown in Figure 7. This information and the material properties were used as input to a general purpose, cycle-by-cycle, crack propagation, program [11] to generate the master curves shown in Figure 8. Crack growth calculations were started at an initial crack of .05 mm. The curves were extrapolated for smaller crack lengths.

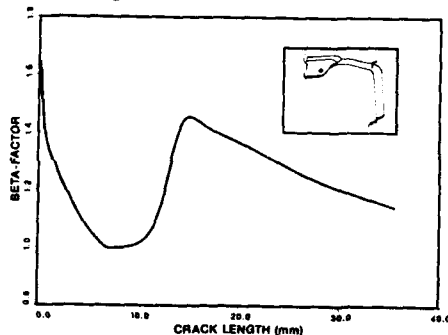


Figure 6 - Boundary Correction Factor for a Corner Crack/Through Crack at the lower Wing Skin Radius

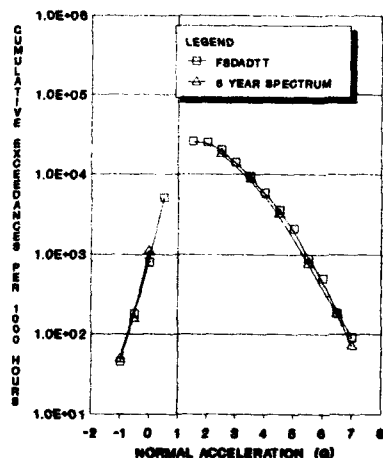


Figure 7 - CF118 Service and PSDADTT Spectra

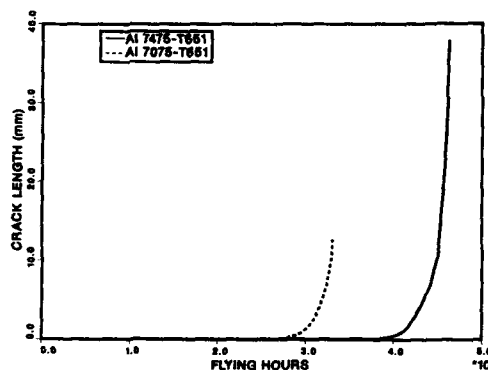


Figure 8 - Crack Growth Curves for the Two Lower Wing Skin Materials

The cumulative probability of stress exceedance per flight is shown in Figure 9. The curve was extrapolated to approximately 8×10^{-4} using a three-parameter Weibull function [12]. This value corresponds to the allowable maximum load in two design service lifetimes at a slow crack growth non-inspectable location, as specified in MIL-A-87221 [13]. The corresponding maximum stress distribution is shown in Figure 10.

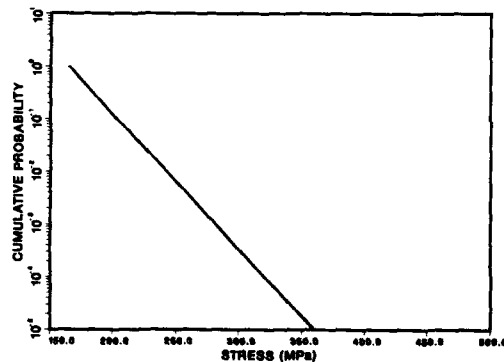


Figure 9 - Cumulative Maximum Stress Distribution

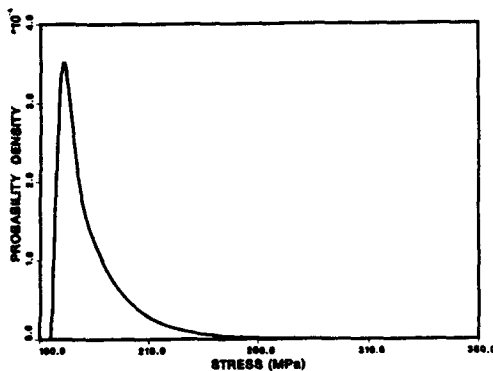


Figure 10 - Probability Density Function of Maximum Stress

Two POD/CL curves were used in the analysis for comparison purposes. Curve 1 shown in Figure 11 was obtained by fitting the log logistics model to a set of inspection data from an experimental program conducted by the Canadian Forces [14] and representative of the lower wing skin radius. Curve 2 plotted in Figure 12 was derived [15] by also making use of the log logistics function for representing NDI data generated from an extensive research program carried out by the USAF [6]. The latter curve has been used in the past to recommend inspection intervals for several CP116 structural details.

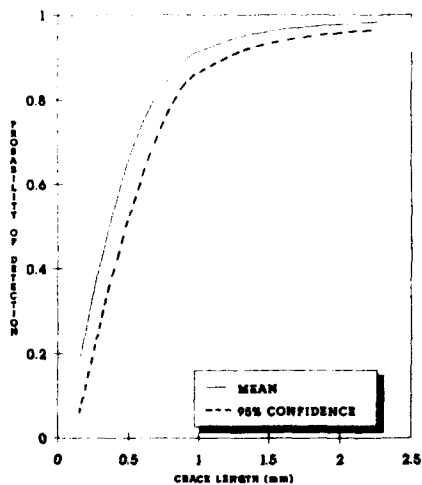


Figure 11 - Probability of Crack Detection at the CP116 Lower Wing Skin Radius - Eddy Current Inspection from CF Study [14]

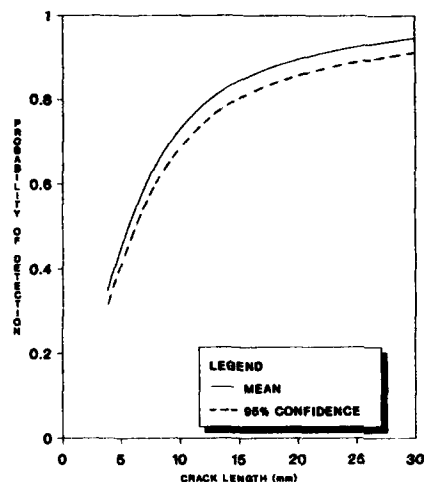


Figure 12 - Probability of crack Detection - Eddy Current Inspection from USAF Study [15]

Finally, the residual strength curve required to define the failure zone and hence calculate the probability of failure considered is shown in Figures 13, for each of the materials considered.

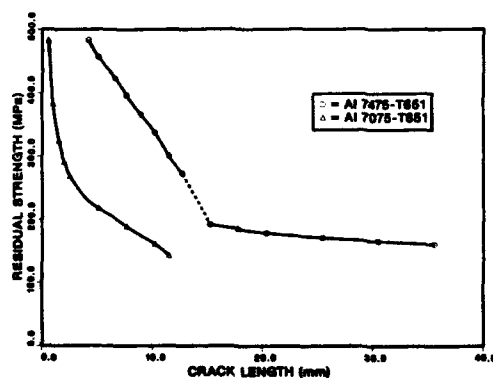


Figure 13 - Residual Strength Curves for the Lower Wing Skin Radius

4.0 RESULTS AND DISCUSSION

The results presented in this section are discussed in relative terms mainly because of the assumption made on the HPS distribution and of the omission of the crack growth variability in the

analysis. These two variables were found to predominantly affect the probability of failure [16-19].

The single flight probability of failure curves for the no-inspection case are shown in Figure 14 for the two materials investigated. This information provides the fleet manager with the average risk of flying an aircraft at every flight. A meaningful interpretation of the POF curves requires that an acceptable level of risk be determined. For a main structural part of an aircraft values between 10^{-7} and 10^{-8} for a single flight POF and between 10^{-3} and 10^{-4} for the whole service life have been mentioned [3, 9, 20]. However, the acceptable risk level greatly depends on the type of aircraft and mission flown such that sound engineering judgment and past experience must be used on a case-by-case basis.

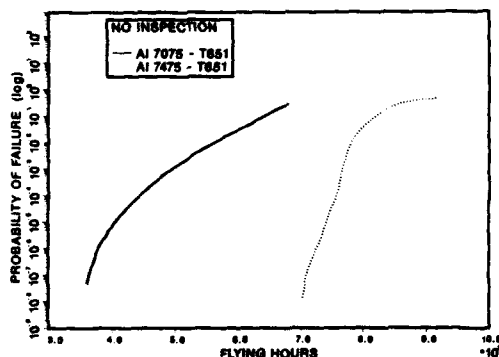


Figure 14 - Comparison of Single Flight Failure Probabilities for the No-Inspection Case

If one assumed a target extended life of 8000 hours for the structure, Figure 14 shows that, for the required lifetime, both materials give probabilities of failure in excess of the acceptable level range previously mentioned. This obviously suggests that inspection must be carried out. On the other hand, Figure 14 provides justification for the change of the lower wing skin material since for any probability of failure the corresponding number of flying hours is always lower for Al 7075 than for Al 7475.

The significant influence of inspection using POD Curve 1 is illustrated in Figure 15 for the 7475-T651 aluminum and for the inspection schedules given in Table 2. This information provides the basis for determining the inspection interval that would ensure flight

safety. In the present case, an inspection every 1500 hours after the new wing installation seems appropriate in order to meet the aforementioned requirements. Similarly, a 500-hour inspection interval is required for Al 7075-T651 as shown in Figure 16.

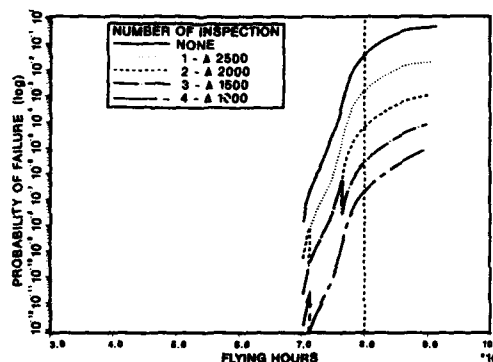


Figure 15 - Influence of Number of Inspections on the Single Flight Probability of Failure for Al 7475-T651

Table 2 - Inspection Schedules for the Lower Wing Skin Radius

Number of Inspections	Inspection Time (Hours)			
	1	2	3	4
1	5500			
2	5000	7000		
3	4500	6000	7500	
4	4000	5000	6000	7000

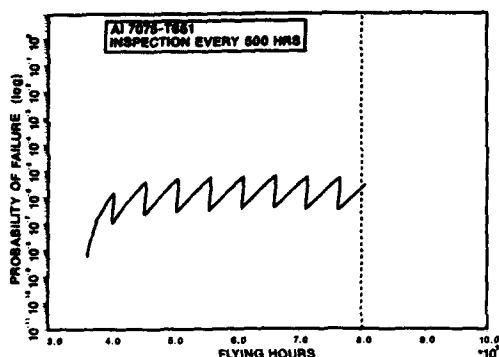


Figure 16 - Single Flight Failure Probability for Al7075-T651 Skin Inspected Every 500 Hours

The probability of aircraft failure after a given number of flying hours is another piece of relevant information on which to base fleet management decisions. Such information is depicted in Figure 17 for the two skin materials with the inspection intervals recommended above. Again here, the fleet manager must decide on an acceptable risk of failure considering the entire life of the aircraft. For the maximum acceptable level of 10^{-3} already quoted, Figure 17 shows that the Al 7075 wing skin would need a tighter inspection schedule whereas the Al 7475 one would satisfy the requirement. However, in the latter case it might be advisable to increase the number of inspections after 7000 hours given the steepness of curve and if additional life extension is desired. Besides, inspection intervals would need to be reduced further if other critical locations need to be considered.

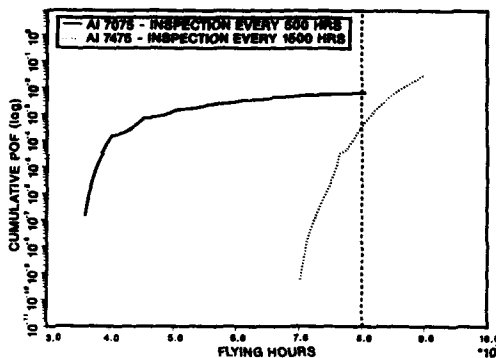


Figure 17 - Comparison of Cumulative Failure Probabilities for the Two Wing Skin Materials with Inspection

The probabilistic approach also allows for discrimination between NDI systems. This is illustrated in Figure 18 where probabilities of failure using POD Curve 1 and POD Curve 2 are compared for the three-inspection schedule. One may observe a three order-of-magnitude decrease in probability of failure when NDI System 1 is used instead of System 2. However, the relatively high POD reliability of System 1 would need to be substantiated with in-service inspection data. In any case, this emphasizes the importance of experimentally deriving POD curves representative of the critical structural details and the actual inspection conditions.

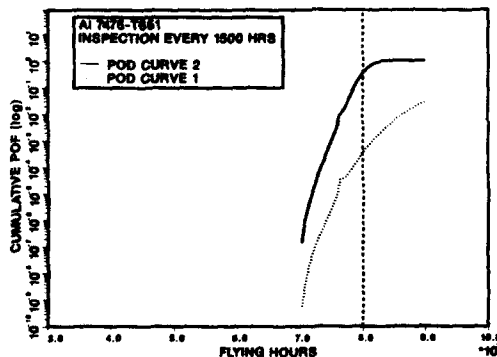


Figure 18 - Influence of the NDI System on the Cumulative Failure Probability

The above results clearly demonstrate the usefulness of probabilistic damage tolerance analysis in helping the fleet manager make technical decisions on a more rational basis. But, this case study revealed that practical issues should be considered before choosing the probabilistic approach. First, the EIFS distribution is one of the most influential and most difficult to obtain input data in the probabilistic analysis. The fractographic data base required for its derivation does not always exist and must be generated from numerous representative coupon tests. Besides, the EIFS is an artificial concept that must be used within certain constraints [1]. Second, neglecting the crack growth rate distribution, such as in the present case study, may yield unconservative results [10]. Stochastic models [3, 10] exist that account for the crack propagation scatter but again these models require that a considerable amount of experimental data be gathered. Finally, the fact that several critical locations are present in an aircraft structure further magnifies the amount of data required. Hence, a full probabilistic analysis can become costly and before it is carried out a proper assessment of cost versus benefit should be made.

Nevertheless, the requirements on the input parameters can be considerably relaxed when the probabilistic approach is used in a relative manner. Thus, changes in the applied load spectrum, detection capabilities of different inspection techniques, material substitutions, and design modifications can be assessed based on a probability of failure criterion. This approach is recommended for the CP fleets until a representative database incorporating EIFS distributions, crack propagation distributions and POD curves becomes available.

5.0 SUMMARY AND CONCLUSIONS

A probabilistic damage tolerance procedure for analyzing aircraft structures has been described. The procedure accounts for the flaw size distribution at the critical details, the distribution of the maximum service load and the probability of crack detection during inspection to determine the probability of failure at each flight and the cumulative probability of failure after a given number of flights.

A numerical example involving a trainer fleet from the Canadian Forces shows that the method can be successfully applied to assist the fleet manager in selecting nondestructive inspection methods and in establishing inspection and maintenance schedules that will

ensure flight safety. The procedure should be upgraded to include the variabilities in crack growth rate, load spectrum and fracture toughness for more accurate probability of failure predictions. In cases where the key input distributions are missing because of a lack of experimental and service data, assumptions can be made and the probabilistic approach can still be used on a relative basis. This was the case in this study. Carrying out probabilistic damage tolerance analysis may prove time-consuming and costly, for a large amount of data must be gathered. However, the analysis may be warranted considering the even higher cost associated with a deterministic "worst-case" approach that often results in unnecessary inspections and repairs.

ACKNOWLEDGMENTS

The authors wish to thank Mrs. Ann Ondrechak for painstakingly typing this paper. The contribution of Mr. Yvan Francoeur to the computer work is also gratefully acknowledged.

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FATIGUE LIFE BEHAVIOUR OF COMPOSITE STRUCTURES

by

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SUMMARY

The general fatigue behaviour of composite structures is presented. The verification and certification philosophy, depending on the special behaviour of this material, is shown. On this basis a lot of structure tests were carried out at IABG, taking test parameters such as damages, environmental influence, load conditions etc. into consideration. The summary of these tests is presented.

1. INTRODUCTION

To ensure the operating safety of aircraft structures, components of aircraft and even complete airframes have to undergo fatigue life tests. Caused by the fatigue sensitivity of metals, this philosophy is undisputed for airframes manufactured of this material. With composite structures on the other hand, the necessity of such an experimental fatigue life verification is disputed, because several investigations show an insensitivity of this material to fatigue. Even test results are known, which show an improvement of the composite material through the applied load cycles and environmental conditions.

But the strength of composite structures shows sensitivities on a lot of other influences, such as for example environmental conditions, impact damages and so on. For the certification of composite aircraft the existing philosophies therefore have to be re-thought and if needed be changed.

2. GENERAL FATIGUE BEHAVIOUR OF FIBER REINFORCED PLASTICS

Compared with metals, the s-n-curve of

composites show another course. In Fig.1 the fatigue behaviour of these two materials is depicted exemplary for aluminium (3.1354.T3) and a composite (Ciba 914C/T300). The curve for the composite shows the typically smaller gradient than the metal curve. This means that the CFRP can bear more load cycles to failure at the same stress level than the aluminium. But on the other hand, this flatter curve

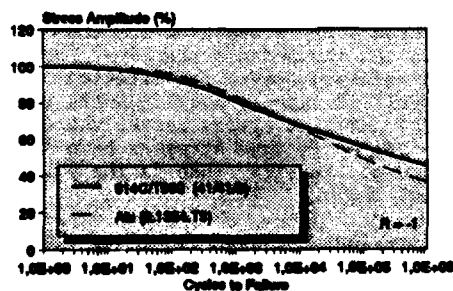


FIG.1 Comparison Aluminium / CFRP (unnotched)

leads to a greater sensitivity of the composite against the scattering of the material, because a small increase of the load level leads to high reduction of the cycles to failure.

The influence of notches in a carbon fiber laminate /1/ is shown in Fig.2. In this case the notch was an open hole. The static compression strength of the composite becomes very much reduced by the notch, but in the region of $1.0E6$ there is nearly no more difference between the notched and the plain specimen. This is because the notched specimen is very unsensitive to fatigue loads and therefore shows a s-n-curve with a very small gradient.

When there are damages in the laminate, the fatigue reaction of the com-

posite is nearly the same as if there are notches in it.

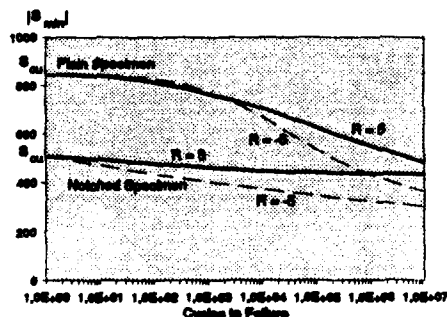


FIG.2 Influences of Notches and of the Stress Ratio R on CFRP /1/

Also the influence of the stress ratio 'R' on CFRP is to be seen in Fig. 2. 'R' is defined as the ratio between the minimum stress to the maximum stress of the applied fatigue loads. S-n-curves for $R = -5$, this means for the compression/tension mode and for $R = +5$, this means the compression/compression mode is depicted for the plain and the notched specimen. For both specimen types, the lifetime of the CFRP is reduced significantly more by the negative stress ratio, where the applied fatigue stresses have a change of the sign, than by the positive ratio.

The previous diagram is valid, when the stresses mainly are acting in the plain of the fibers. But when this is not the case, for example when there are out-of-plane loads and these loads therefore have to be taken over by the matrix, the composite reacts

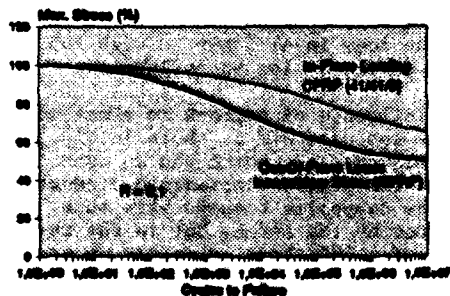


FIG.3 Influence of Out-Of-Plane loads /1/ /2/

more sensitively to fatigue loads as can be seen in Fig. 3. The interlaminar shear fatigue stresses, received with the short-beam-shear-test /2/, are compared with the fatigue results /1/ of an in-plane tension/tension test. It can be seen that the fall off of the curve for the out-of-plane loaded specimen begins very early and is much higher than the one of the in-plane loaded specimen. These results are typical for these loading modes, although in the diagram two different materials are depicted.

In Fig. 4 the fatigue behaviour of a tension/compression strut /3/ is shown. The strut was wound with carbon fibers. The load introduction parts are, as usual, out of metal. In the static tests and also in the fatigue tests with high loads the strut failed after relatively few load cycles in the CFRP part. But in the fatigue tests with a low stress level, the CFRP proved to be resistant to the fatigue loads and the strut failed in the metal part after about tenthousand load cycles.

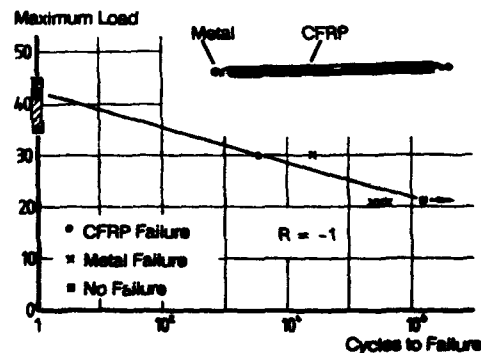


FIG.4 Tension-Compression Strut /3/

To investigate the influence of combined load and temperature cycles, we tested fuselage frame components /4/. The material for the components was glassfiber epoxy and the components had a length of for about 1.2 m.

We applied temperature cycles from -55°C to 72°C and at the same time we cycled the components with the mechanical loads of a fatigue life spectrum. The initial static strength and

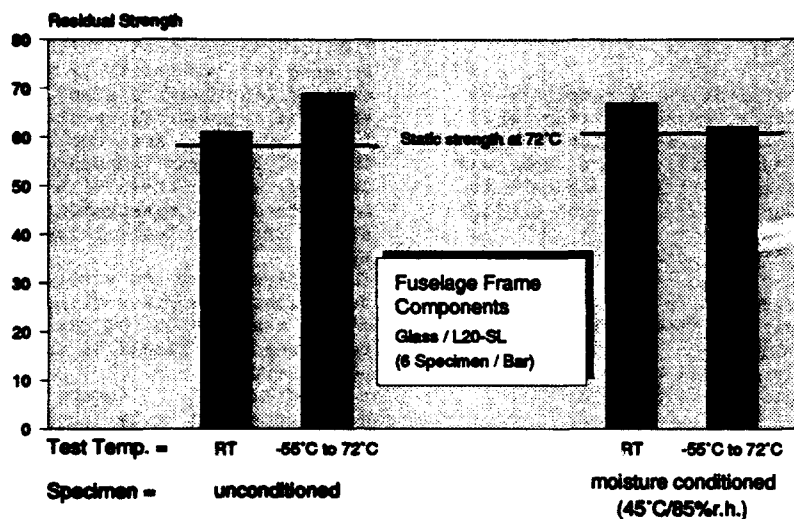


FIG.5 Influence of Combined Load- and Temperature-Cycles
Residual Strength Tests at 72°C after 3*30.000 Flight Hours /4/

the residual strength after 3*30,000 simulated flight hours was investigated. We tested unconditioned (as received) and conditioned (at 45°C and 85% relative humidity) components. The results are shown in Fig. 5.

There were no significant differences between the fatigue tests carried out at room temperature and those carried out with temperature cycles, because this difference was within the scattering of the test results. Even a reduction of the initial static strength through the fatigue loads could not be recognized.

Taking into consideration all these results, it could be said generally, that the s-n-curves of composites and therefore the fatigue behaviour of this material basically is determined by the following factors:

- The stress ratio R
A change of the sign of the applied stresses leads to a higher gradient of the s-n-curve and thus leads to a lower number of cycles to failure.
- The load direction
Stresses which act not mainly in the fiber direction reduce the number of

cycles to failure (for example out-of-plane loads).

- Damages, holes
Damages and holes reduce the static strength of laminates significantly, but with fatigue loads this strength does not become reduced much more.
- The fiber and matrix types
For example glass fibers react more sensitively to fatigue loads than do carbon fibers.
- The laminate type
90° layers react more sensitively to fatigue loads than 0° layers, where the loads could be transferred by the fibers.
- The structure
Naturally there is an influence of the design on the fatigue behaviour of the structure. But unfortunately it is not possible to give a general statement, whether a structure type is sensitive to fatigue loads or not. And because a very, very large number of structure configurations are possible, this point is a very uncertain with regard to the fatigue behaviour.

3. REQUIREMENTS FOR FATIGUE VERIFICATIONS

For the certification of composite aircraft structures the military and the civil authorities require experimental fatigue verifications. Mode and extent of these verifications have to be fixed specifically for each project and depend upon

- the design philosophy (for example fail safe or safe life)
- the experience with similar structures, received from tests and also from service
- and of the general experience of the company which developed the aircraft.

Official requirements like for example the 'Federal Aviation Regulations' (FAR) /5/ and additional specifications like the 'Advisory Circular No. 20-107 A' /6/ of the Federal Aviation Administration (FAA) only give general guidelines.

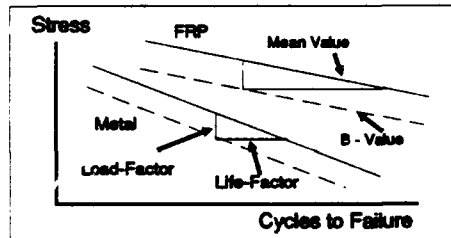
The nature and the extent of the fatigue tests for the certification have to be fixed in technical negotiations between the authorities and the manufacturer on the basis of a proposal from the aircraft company (see also Tab. 1 to Tab. 3).

These fixings for the certification fatigue test have to contain the following definitions:

- The definition of the test specimen, this means whether a full scale test is to be done or whether a component test is sufficient.
- The definition of the test parameters
 - Because the s-n-curves of fiber reinforced plastics show, compared with metals, a flatter course (see Fig. 1), a greater scattering for the lifetime of composites has to be taken into account. The life factor, this means the safety factor for the lifetime simulation, normally is between 2 and 4. This factor depends upon the used load factor. For the fatigue life tests of the general aviation, the load spectrum usually is increased by the load factor 1.1. The general correlation between life factor, load factor and scattering is

shown in Fig. 6.

- The loading spectrum, which is to be used for the tests, is to be defined (for example standard spectrum, randomised spectrum, flight-by-flight spectrum).
- The test program (sequence of testing, static tests with strain and deformation measurements, residual strength test, inspections etc.).
- The influence of the environmental conditions (temperature, humidity) may be examined on the fatigue test component itself or with coupon tests.
- The definition of the damage tolerance program is to be fixed. To prove manufacturing defects and impact damages, the location, the type and the number of these damages has to be agreed. Concerning the impact damages the impact energy is to be fixed.



F.G.6 Correlation Between Life Factor, Load Factor and Scattering

4 FATIGUE LIFE VERIFICATIONS OF FRP AIRFRAMES

In the recent years a lot of FRP aircraft structures have been developed and tested and many of these structures are already flying.

In the tables 1 to 3 an overview of fatigue tests on composite structure parts and the fatigue results are shown. It can be seen, that at most of these composite structures no fatigue damages occurred.

For example, the fatigue tests with the SEASTAR wing, ALPHA-JET horizontal tail and of the TORNADO main landing gear door are presented in the following.

Part Name	Load Spectrum	Environmental Conditions (Test Conditions)	Damage Tolerance Investigation		Fatigue Test Result	
			Number of Impact Damages	Number of Manufacturing Defects	Damage	
ALPHA-JET / Speed Brake Carbonfibre / Epoxy	1.0 / 50000 FH	RT / wet	-	-		x
ALPHA-JET / Horizontal Tail Carbonfibre / Epoxy	1.0 / 20000 FH	RT / wet	-	-	x (Note Out-of-specification)	
* ADVANCED FIGHTER AC / Wing Integral Tank Box Carbonfibre / Epoxy	1.0 / 2.0	RT / dry	-	-		x
* ADVANCED FIGHTER AC / Fin Box Carbonfibre / Epoxy	1.0 / 3.0	RT and 100°C / wet	-	-		x
* ADVANCED FIGHTER AC / Fuselage Integral Tank Box Carbonfibre / Epoxy	1.0 / 4.0	RT and 90°C / wet	-	-		x
TORNADO / Main Landing Gear Door Carbonfibre / Epoxy	1.0 / 4.0	RT and 100°C / wet	1	-		x

* Tests for Verification of new structure concepts and materials

Tab. 1 Overview of Fatigue Tests on Composite Structure Parts performed at IABG
Part I: Military Application

Part Name	Load Spectrum	Environmental Conditions (Test Conditions)	Damage Tolerance Investigation		Fatigue Test Result	
			Number of Impact Damages	Number of Manufacturing Defects	Damage	
GROB G 115 / Wing Glassfibre / Epoxy	1.10 / 3.0	RT and 72°C / wet	11	9	x (caused by unsymmetrical support)	
SPEED CANARD / Wing Glassfibre/Carbonf./Epoxy	1.10 / 3.0	RT / dry	13	12	x (Metal Part)	x (Composite)
SEASTAR / Wing Glassfibre/Carbonf./Epoxy	1.13 / 2.0	RT / dry	12	18		x
SEASTAR / Fuselage Glassfibre / Epoxy	1.13 / 2.0	RT / dry	13	22		x
SEASTAR / Horizontal and Vertical Tail Glassfibre / Epoxy	1.10 / 3.0	RT / dry	6/3	12/10		x
SEASTAR / Horizontal and Vertical Tail Glassfibre / Epoxy	1.10 / 3.0	RT and 72°C / wet	6/3	12/10		x
EGRETT / Wing Glassfibre/Carbonf./Epoxy	1.10 / 3.0	RT / dry	12	22		x
EGRETT / Fuselage Glassfibre/Carbonf./Epoxy	1.10 / 3.0	RT / dry	15	43		x
EGRETT / Horizontal Tail Glassfibre / Epoxy	1.10 / 3.0	RT / dry	13	21		x

Tab. 2 Overview of Fatigue Tests on Composite Structure Parts performed at IABG
Part II: Civil Application (General Aviation)

Aircraft / Part / Material	Loadcase / Lifespan	Environmental Conditions (Ref. Conditions)	Damage Tolerance Investigation		Fatigue Test Result	
			Number of Impact Damages	Number of Manufacturing Defects	Damage Yes	Damage No
* DORNIER 328 / Rear Fuselage Section Carbonfibre / Epoxy	1.15 / 3.5	RT / dry	35	90	X (only 1 uncracked damage)	
AIRBUS A 320 / Outer Landing Flap	1.15 / 1.5	RT / wet	51	38		X
AIRBUS A 310 / Fin Box Carbonfibre / Epoxy	1.15 / 3.0	RT and 45°C / wet	14	Different Defects and Repairs		X
AIRBUS A 320 / Fin Box Carbonfibre / Epoxy	1.15 / 1.5	RT and 50°C / wet	big no. of damages	Different Defects and Repairs	X (limited damages caused by cut of plane loading)	

* Development Test

Tab. 3 Overview of Fatigue Tests on Composite Structure Parts performed at MBB and Dornier (Civil Application)

4.1 FATIGUE LIFE TEST WITH THE SEASTAR WING

A fatigue certification test was carried out at IABG with a wing of the SEASTAR CD02, a two engine aircraft.

The schematic test set up is shown in Fig. 7. The wing was fixed at the test support in the inverted flight position with the original strut support. For the simulation of the flight loads 6 hydraulic jacks were used. The tension and compression loads were introduced by loading trees and contour boards into the specimen. Two jacks,

simulating the engine thrust and mass load, were directly connected to an engine dummy structure. To simulate the maximum service temperature of 72°C for the static tests, an isolation chamber was erected around the wing. The heat was produced by electric air heaters.

In accordance with the certification requirements of the authorities (LBA and FAA), the following qualification steps were performed:

- While manufacturing the wing, some manufacturing defects (weak bonds) were artificially produced. The

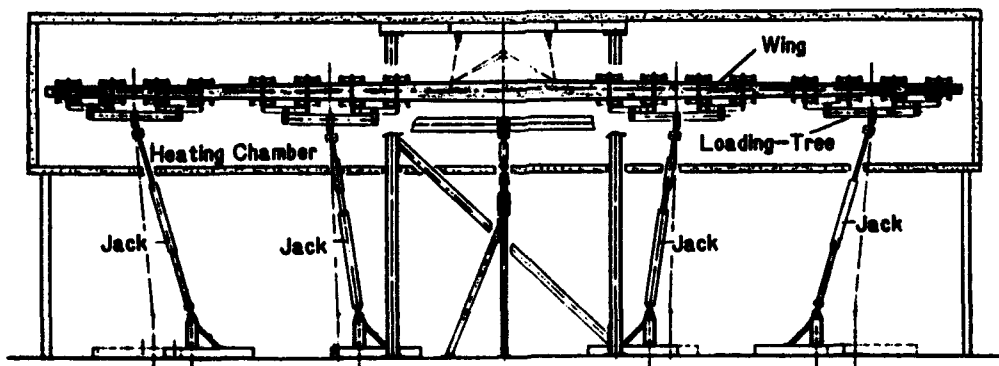


FIG.7 Schematic Test Set Up of Seastar CD02 Wing Test (mounted in inverted flight position)

- front spar and the main spar became damaged by impacts.
- The finished wing was inflicted with impact damages.
 - Static tests at room temperature and 72°C up to limit load $j=1.0$ (for strain gauge and deflection measurement and qualification of the manufacturing defects and impact damages).
 - Simulation of two lives, each of it with 30,000 flight hours. These tests were carried out at room temperature. The loads were increased by a loadfactor of 1.13.
 - Static test at room temperature up to limit load $j=1.0$ (for strain gauge and deflection measurement and qualification of the manufacturing defects and impact damages).
 - Static test at room temperature up to ultimate load $j=1.5$.
 - Residual strength test at 72°C.

The SEASTAR wing passed this certification tests successfully /6/:

- After simulation of the 2 lives no propagation of the inflicted damages could be identified. Also no additional defects could be detected.
- The residual strength test was successful up to 2,25 (225% limit load) Because the test component was not conditioned with moisture, this test result still has to be corrected by a factor considering the moisture influence. This influence was determined by coupon tests.

4.2 FATIGUE LIVE TEST WITH THE ALPHA-JET HORIZONTAL TAIL

The CFRP-horizontal tail of the Alpha-Jet was tested at IABG in a certification fatigue test.

Fig. 8 shows the schematic test set up. The horizontal tail was fixed at the original support fitting by struts in the inverted flight position. For the simulation of the flight loads 4 hydraulic jacks were used. The tension and compression loads were introduced into the specimen by loading trees and pads, which were bonded on the upper side of the horizontal tail. An isolation chamber was erected around the horizontal tail to simulate the required temperatures of -55°C and

+70°C for the static tests. The heat was produced by an electric air heater and the cold by a cooling unit.

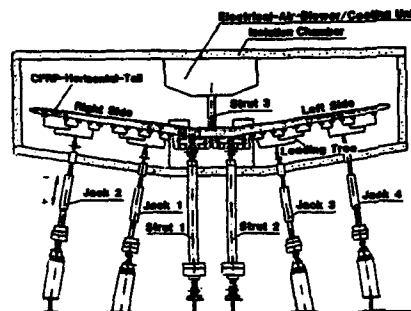


FIG.8 Test Set Up of the Alpha Jet CFRP Horizontal Tail

The following qualification steps were performed in accordance with the authorities:

- Static test at room temperature (specimen as received) up to 125% limit load.
- Static test at -55°C (specimen as received) up to 115% limit load.
- Conditioning of the specimen (1% moisture content).
- Static test at 70°C (conditioned specimen) up to 115% limit load.
- Simulation of 20,000 flight hours at room temperature.
- Static functional test at room temperature, -55°C and 70°C up to limit load.
- Reconditioning of the horizontal tail (1% moisture content).
- Residual strength test at 50°C at the conditioned specimen.

With the Alpha Jet horizontal tail the following results /7/ were received:

- During the fatigue test a number of the countersunk holes in the leading and trailing edge box got delaminations (see Fig. 9). But these damages proved to be uncritical for the fatigue test. Nevertheless these delaminations were repaired by bonding potted metal washers into the countersunk holes before the start of the residual strength test.
- The residual strength test was successful up to 1,65 (165% limit load).

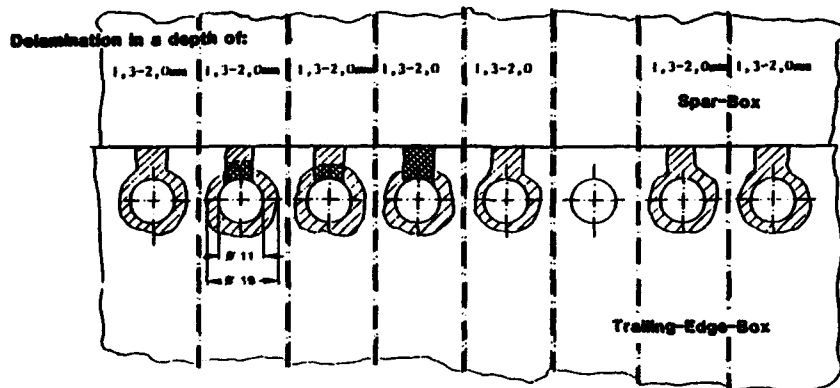


FIG.9 Example for the Delaminations in the Countersunk Holes of the Alpha-Jet Horizontal Tail

4.3 FATIGUE LIFE TESTS WITH THE CFRP TORNADO MAIN LANDING GEAR DOOR

Fatigue tests with two Tornado main landing gear doors, manufactured of graphite/epoxy, have been carried out at IABG.

The test was set up in the way that the door could be representative loaded as well in the closed as in the opened position (see Fig. 10). A hydraulic jack was used for the loading. The tests were carried out in an isolation chamber to simulate the environmental conditions (humidity and temperatures between room temperature and 100°C).

Two main landing gear doors have been tested under fatigue loads, one with a repaired and the other with an unrepaired impact damage. Corresponding to the certification requirements of the authorities, the following test steps were performed.

- Introducing of an impact damage into both doors.
- Repaire of the impact damage of one door.
- Moisture conditioning of both specimen at 70°C/85%r.h.
- Static tests with both specimen at room temperature and at 100°C up to limit load.
- Fatigue test with both doors. The load cycles became superimposed by

temperature cycles between room temperature and 100°C and by humidity (for about 85% r.h.). During the test the gear door was opened and closed. Four lifes were simulated, each of them with 4,000 flights.

- Residual strength test with both doors at 100°C.

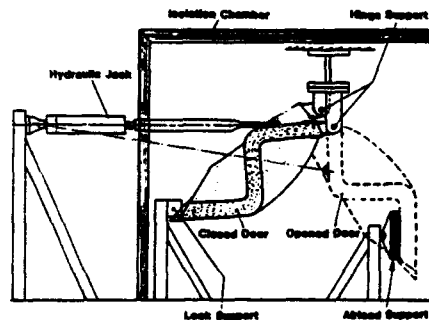


FIG.10 Test Set Up of the Tornado Mean Landing Gear door

In these tests the following results were obtained:

- During the fatigue test no growth of the impact damage.
- No damages at all, also not in the repair.
- The residual strength of both doors was 2.1 (210% limit load).

A third Tornado mean landing door, which was statically tested, showed the same safety factor. Thus it can be

concluded that the fatigue loadings had no influence on the strength of this component.

5. SUMMARY OF THE IABG-EXPERIENCES FROM THE FATIGUE LIFE TESTS ON COMPOSITE STRUCTURES UP TO NOW AND CONCLUSIONS

- If the calculation of the structure is perfect and when the structure reaches in a static test the ultimate load without any problems, then this structure shows also a good fatigue life behaviour. Under this prerequisite no fatigue damages occur without of small local and uncritical damages, mostly caused by the load introductions and the bearing application (which cannot be always representative) of the test sample.
- When the strains in the structure are within the allowed region, the damage tolerance behaviour of the structure normally is uncritical, this means that no damage growth occurs.
- A negative influence of the environmental conditions (temperature, humidity) on the fatigue life behaviour could not be recognized.

But in spite of these good experiences with composite structures in lifetime tests, we think that also in the future it cannot be done without fatigue tests because:

- In many cases there are still combinations of metals and composites in the structure, and it is possible, that this structure then fails by fatigue loads in the metal part.
- The metal fasteners in bolted or riveted joints may become critical, also bonded joints (shear stresses).
- Out-of-plane fatigue loads often prove to be very critical for composites. To detect these loads in a theoretical way is nearly not possible or very costly.
- There is a very, very wide design,

manufacturing and material variety and it is always possible that one of these parameters proves to be critical on fatigue.

- Structure anomalies and design mistakes can never be excluded.
- The introduction of new design and manufacturing philosophies and technics bring new risks, which cannot be estimated without tests.

Nevertheless the mode and the extension of the fatigue tests always have to be adjusted according to the extended and improved expertise. We see here the following possibilities:

- Fewer fatigue tests with large components may be carried out, for example
 - no fatigue tests with movable surfaces like rudders and flaps, etc.
 - reduction of the fatigue tests with the empennages (horizontal stabilizer and fin), when the same design and material was experimentally verified with a wing or with other empennages.
- The main focus should be put on fatigue tests with subcomponents or critical structure areas. In this way normally the number of the specimen is increased and thus it is possible to create statistics. These subcomponents may be
 - wing-fuselage-connection
 - load introductions in high loaded structures
 - other critical structure areas such as cross section change, panels with big open holes, high stress areas etc.
- Simple tests without the simulation of environmental influences.
- Short test times by using the load/life factor assessment.

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A PARAMETRIC APPROACH TO SPECTRUM DEVELOPMENT

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SUMMARY

This paper describes the parametric approach being pursued for the development of the test spectrum for the CF-18 center fuselage. The paper provides an overview of the F/A-18 International Follow-On Structural Test Program (IFOSTP) with emphasis on spectrum development. The specific technical elements of the spectrum development task are introduced and detailed information is provided on the CF-18 usage monitoring system, the approach to usage processing and the manoeuvre identification methodology which is forming the basis of the parametric approach to CF-18 spectrum development. A specific example of the manoeuvre identification process is provided and recommendations for future monitoring systems are offered.

BACKGROUND

With respect to aircraft structural design, there has been an explosion of technology. More flexible design requirements, for example, the USAF Mil Prime approach [1], allow advantage to be taken of new developments in materials, manufacturing processes, control systems and structural optimization techniques.

Integration of new software systems [2], which allow a multi-disciplinary approach to structural optimization, has resulted in improvements in aircraft performance and reductions in production costs. Sensitivity analyses are performed as part of this process, however, the risk of increased sensitivity to structural damage if the aircraft is operated off-design remains.

This situation is exacerbated for aircraft with digital flight control systems (DFCS), particularly fighter aircraft. The local stress histories at critical areas of these aircraft are dependent not only on the role/mission similitude to design but also on the assumptions with respect to points-in-the-sky (PITS) at which manoeuvres are flown and the control algorithms used by the aircraft to fly these manoeuvres. While the pilot command and the rigid body motion of the aircraft may be essentially identical for the same manoeuvre flown at two disparate PITS, the flight computer decides the most optimum control motions to accomplish the manoeuvre. An example is a roll at low dynamic pressure which has a high aileron input and the same manoeuvre at high dynamic pressure which may be accomplished predominantly by differential empennage control movements. The resulting stress/strain distributions are quite different.

The combined effect of changes in roles/missions and in-service refinements to the control algorithms to gain optimum performance, can invalidate the engineering data originally used to establish the static, fatigue and damage tolerance response of the aircraft.

Another important variable is pilot technique. DFCS offer the pilot increased manoeuvrability that fighter pilots are quick to discover and use. The frequent use of high angle of attack manoeuvres by the CF-18 and F-16 aircraft are examples of this increased manoeuvrability. These techniques, while effective from a performance viewpoint, can introduce new load cases or increased frequency of assumed load cases, to the detriment of the structure.

Aircraft usage monitoring has also matured. Sophisticated multi-channel recorders are able to capture time-based strain reversals, accelerometer peaks and valleys, control positions, and aircraft configuration and flight condition parameters. These latter systems are fundamental to the maintenance of an effective fleet tracking program for modern aircraft with DFCS. The sophistication of these systems and the large amount of data arising from even one flying hour raises new technical issues. Data handling, cataloguing and storage are expensive and time consuming. Data errors can be subtle and require elegant validation techniques which add to the general costs associated with data tracking. Maintenance of transducers, strain gauges, and the hardware and software of the system itself is an additional burden to the field operators.

There are also challenges from the viewpoint of spectrum development. Several issues must be addressed in developing spectra from this data. Generation of usage statistics in tabular or graphical form is relatively straightforward, particularly if the data is analyzed and collated as it arrives at the data processing facility. These statistics provide an excellent summary of usage and allow direct qualitative and quantitative comparisons to the design assumptions. The amount of data is voluminous, for example, each CF-18 yields about 25 Megabytes of usage data per year. Typically, after initial analysis, the data from an individual aircraft is stored to some mass storage device such as tape. Therefore, if some aspect of usage is to be investigated that is not a part of the general data reduction and collation process, it is a time consuming process to identify the data sample to be investigated, re-establish access to it, verify that all requested data has been

retrieved and perform the analysis. This issue is becoming less of a constraint as computer hardware and software systems improve.

The greatest challenge is in deriving a representative sequential usage spectrum. Depending on the data collection philosophy adopted for a particular monitoring system, the derivation process may be based on the direct strain/loads monitoring of critical areas or on the interpretation of sequential flight control and response data [3]. Both approaches have their philosophical advantages and their technical disadvantages. Direct monitoring of strains automatically adjusts for changes in role and configuration. It provides a direct indication of the stress sequence for the critical area in question and allows the most accurate derivation of its usage. Disadvantages include the difficulty in extrapolating the data to determine stress histories in alternate areas, the serviceability and stability of the strain gauges, and the requirement to calibrate the system.

Parametric monitoring offers the advantage of allowing the loads sequence to be calculated for any location in the structure. In practice, this is an extremely difficult task and requires knowledge of external aerodynamic distributions, inertial distributions, the motion of the aircraft and the control surface inputs that caused this motion. With this knowledge, external load distributions in terms of bending moment, shears and torques are derived that can be used to calculate stress histories at critical areas. This is a daunting task.

In practice, a hybrid system which uses both strain data and aircraft parameters is recommended. This allows a closed loop approach to be used in that there is some matching of strain histories with actual aircraft manoeuvring for the first approach and some matching of actual and calculated strain histories for the latter. In addition, to simplify the analytical requirements associated with the parametric approach, it is possible to associate systematic parameter variations with defined manoeuvres. In this way the process of parametric analysis can be reduced to the identification and characterization of manoeuvres and the stress derivation process can be directed towards these specific manoeuvres. This is the approach adopted for the CF-18 spectrum development task for the center fuselage.

IFOSTP PROGRAM-OVERVIEW

The Canadian Forces (CF) and the Royal Australian Air Force (RAAF) are pursuing joint follow-on structural test programs of the F/A 18 Hornet aircraft [4]. This program, formally called the International Follow-On Structural Test Program (IFOSTP), was established after joint review of RAAF and CF fleet usage statistics and an evaluation of these statistics against the compliance testing and analysis programs carried by the manufacturer.

The Aeronautical Research Laboratory in Australia is responsible for the development of an aft fuselage and empennage spectrum and the accomplishment of an aft fuselage and empennage test. It is planned to apply dynamic loads and manoeuvre loads simultaneously. The test is further complicated by a marked shift in the dynamic spectrum as a result of the installation of a fence on the leading edge extension (LEX fence) that will necessitate a spectrum change during the test.

In Canada, Bombardier Inc. will be performing a center fuselage test using a spectrum developed by the Institute for Aerospace Research (formerly the National Aeronautical Establishment) of the National Research Council. It is planned that the Institute for Aerospace Research (IAR) will develop the spectrum and conduct a wing test.

This paper will focus on the approach used by the IAR to define the center fuselage test spectrum.

GENERAL APPROACH TO CF-18 SPECTRUM DEVELOPMENT- CENTER FUSELAGE

The CF-18 aircraft has a digital flight control system designed to maximize performance and maintain stability. It has some "carefree manoeuvring" features that inhibit overstressing the aircraft. The primary flight controls consist of ailerons, horizontal stabilizers, rudders and leading and trailing edge flaps. The control system takes inputs from the pilot and from sensors measuring selected aircraft parameters and then initiates the most appropriate control surface movements to respond to the pilot's demands and still maintain stability. As discussed, this means that at different PITS, the same manoeuvre may be accomplished by different control movements. This results in different external aerodynamic load distributions which in turn means different internal loads distributions.

Each CF-18 aircraft is equipped with a Maintenance Signal Data Recording System or MSDRS. This system continuously samples and records flight, engine and systems parameters as well as seven strain channels on a magnetic tape cartridge. For most aircraft parameters of interest, the recording rate is once per second. The limited strain data and the difficulty in developing relevant transfer functions negated full reliance on a direct monitoring approach for the center fuselage and wing spectrum development. Instead, a hybrid parametric approach was adopted which makes extensive use of the aircraft flight parameter data and available strain data to identify manoeuvres. This approach encountered some difficulties because of the low sampling rates for some parameters. For example, roll rate is measured only once per second and roll accelerations are not measured at all. To reduce the task to manageable dimensions, it was decided that approximately 300 hours of MSDRS data representative of CF/RAAF operations would be subjected to a detailed parametric analysis and this data base would be

identified by reference to general fleet statistics.

The spectrum development program is discretized into the major elements shown in Table 1 [5].

WBS	TASK
D.A	Usage Processing
D.B	Data Analysis and Review, Qualitative
D.C	Data Analysis and Review, Quantitative
D.D	Ground Loads Analysis and Review
D.E	Internal and External Loads Distributions
D.F	Spectrum Generation
D.G	Spectrum Sensitivity Studies

Table 1: Task Breakdown

The Usage Processing Task encompasses:

- i) the development of read, check and analysis capabilities for MSDRS data;
- ii) the identification of the most appropriate criteria for evaluating the wing and center fuselage usage;
- iii) the generation of CF fleet statistics for wing and center fuselage; and
- iv) the generation of CF fleet statistics for the aft fuselage and empennage.

Finally, approximately 300 hours of MSDRS data whose general statistics are representative of the type of operations the CF fleet managers wish to be represented by the IPOSTP test, will be selected for parametric analysis. The RAAF will provide a similar data base representative of their operation.

Under the task, Data Analysis and Review, Qualitative, a qualitative review of CF-18 and RAAF F/A18 usage was initiated. The purpose of this review was to develop an understanding of the squadron operating techniques and the pilot procedures that may have an effect on the interpretation of the numerical data from the MSDRS. All operating squadrons in Canada and Australia were surveyed and detailed interviews of squadron pilots were carried out. Using the information from the interviews and the fleet/squadron statistics from the contractor MSDRS data reduction, a qualitative indication of manoeuvre/mission frequencies and distributions was developed and used to provide a preliminary estimate of the most fatigue damaging operating conditions. The most specific conclusion was the confirmation that the PITS at which a manoeuvre is flown has a significant effect on CF-18 loads. Wide variations in peak wing root

strain were found for the same symmetrical "g" manoeuvre at different speeds and altitudes due to the DFCS response to variations in speed and altitude [6]. Since RAAF usage must also be considered in developing the spectrum, a second goal of this activity was to compare the operating procedures and mission codes used by the CF and RAAF and develop a normalized mission code system that can be used for direct comparison of mission/role/manoeuvre distributions [7].

The most technically innovative activity is the development of methods for manoeuvre identification. This work is being done under the Data Analysis and Review, Quantitative task. The object of this task is to quantitatively identify and rank the fatigue damaging manoeuvres flown by the CF-18 and the RAAF F/A 18. It consists of a series of interrelated work packages that will lead to a capability of performing parametric analyses of the 300 hour MSDRS data bases identified under the Usage Processing task. The objectives of these analyses are to isolate and count damaging manoeuvres and to categorize them with respect to specific parameters such as points-in-the-sky and configuration.

There are two prime activities:

- i) the development of the capability to isolate selected manoeuvres from the MSDRS tape records; and
- ii) the use of this capability on the selected MSDRS records.

The final product is the identification of the critical manoeuvres and a listing of these manoeuvres in an order of occurrence for test spectrum development. It should be noted that a conservative approach will be taken in that even comparatively low damage manoeuvres will be considered since final selection of truncation and clipping levels will occur under a separate task known as Spectrum Sensitivity Studies. A detailed description and preliminary results obtained from this work form the basis of this paper.

Ground manoeuvres (which include ground and taxi loads as well as landing and takeoff loads) are being considered separately under the Ground Loads, Analysis and Review task. The MSDRS system does offer some data that allows the general description of landings in terms of sink speeds and landing configuration, but these data do not directly relate to a loads sequence for the undercarriage. No information is available on ground handling loads. Current efforts are directed towards the collection of data from flight test programs in Australia and Canada. The output of this activity will be the identification of critical areas and ground manoeuvre cases for consideration in the development of the test spectrum.

The tasks described thus far will lead to the identification of manoeuvres and the order in which they occur for each mission. There is a requirement to

translate these manoeuvres into sequential balanced loads conditions in terms of bending moments, shears and torques from which the test rig can be designed. This work, identified as Internal and External Loads Distributions, is the responsibility of Bombardier Inc. To accomplish this task, inertial and aerodynamic loading, including the effect of control movements, must be determined for each critical manoeuvre. Where possible, existing data from the design iterations will be used. The other sources of data that will be used include flight test data (manufacturer, RAAF, and CF), wind tunnel data, and aerodynamic analyses. This is a complicated, expensive and time consuming task and efforts are being directed to defining criteria for the grouping of like manoeuvres and reducing the number of manoeuvres that must be analyzed. An important consideration in this grouping study is the control law boundaries from the DFCS because of their affect the strain distributions.

The final tasks are Spectrum Generation and Spectrum Sensitivity Studies. The Spectrum Generation task will finalize the order of missions and the order of manoeuvres within each mission. The actual order of flights and missions from the MSDRS representative data base will form the basis of the spectrum. The purpose of the Spectrum Sensitivity Studies task is to carry out parametric sensitivity studies of the CF, RAAF and the final test spectra. These studies include both analytical modeling and coupon tests. Issues to be investigated include truncation and clipping levels, cycle class/manoeuvre elimination, requirement for marker blocks, and sensitivity to changes in stress level. Tests using the manufacturers design fatigue test spectrum will be performed as a baseline for comparison.

The test and analysis methodologies determined during this Task will form the basis of the follow-on Crack Growth Methodology work that will investigate other issues such as sensitivity of fracture analyses to variations in mission mix and PITS distributions.

CF-18 MAINTENANCE SIGNAL DATA RECORDING SYSTEM

The MSDRS was developed by McAir to provide fatigue usage, flight incident records, engine usage data and associated maintenance data. The system is used on the AV-8B and EA-6B, as well as the F/A-18. Components of the system comprise an on-board processor and a data recorder that writes to a magnetic tape cartridge. A ground station is used to strip the data from the cartridges and make it available for engineering use.

Various parameters are grouped together in MSDRS messages and identified by record codes. These messages are recorded when triggered by an exceedence of a threshold on selected channels. The fatigue Code 49 is triggered when the normal acceleration reaches a peak or valley. Other codes are triggered by engine events or weapons release. (Note

that several messages may be triggered by the same event such as a landing. If this happens, there is a hierarchy for defining the recording sequence. Data can be lost if the number of messages stacked exceeds the buffer size). The flight incident record (Code 46) is written every second, whilst the continuity message (Code 120) that contains the state of the weight-on-wheels switch, is recorded every five minutes and at take-off and landing. A list of codes that are pertinent to the CF-18 fatigue load spectrum development is given in Table 2. All recorded data are time related.

RECORD CODE	DESCRIPTION
4	Fatigue Monitoring - Weapons inventory
21	Recorder Initialization
22	Recorder Summary Message
31	Engine Data Life Cycles
46	Flight Incident Records
47	Fatigue Landing
48	Fatigue Monitoring Initialization
49 to 62	Fatigue Sensor Peaks and Valleys
65	Configuration Message
120	Continuity Data

Table 2: MSDRS codes used for usage processing and manoeuvre identification.

The MSDRS records used for usage processing are Codes 4, 46, 47 and 49 to 62. The parameters of interest are given in Tables 3, 4, and 5.

Parameter	Frequency(Hz)
IAS	1
Pressure altitude	1
Roll rate	1
Angle of attack	1
Longitudinal stick position	1
Lateral stick position	1
Rudder pedal position	1
Normal acceleration	1
Fuel quantity	0.2
Control surface positions	0.2

Table 3: Flight Incident Parameter List.

Normal acceleration *
 Forward fuselage strain *
 Wing root strain
 Wing fold strain *
 Left stabilator strain *
 Right stabilator strain *
 Left fin root strain *
 Right fin root strain *
 Fuel quantity
 TAS
 Altitude
 Roll rate

*Fatigue Sensor Triggered
 Parameters recorded on every peak valley of these parameters

Table 4: Fatigue Sensor Triggered Parameter List.

Max. n_z *
 aircraft weight
 Max. vertical velocity *
 First weight-on-wheels
 Max $n_z W$ *

*These parameters are the maximum values in the 2.05 seconds before weight-on-wheels.

Table 5: Landing Parameter List.

USAGE PROCESSING TASK.

MSDRS data reduction of the CF-18 fleet is carried out by Bombardier Inc. using the Structural Life Monitoring Program (SLMP). This system takes MSDRS files as input and performs data quality checks before outputting quarterly reports on fleet statistics. The IAR used the Fatigue Life Expended Indices (FLEI) from the SLMP reports to select a sample of the fleet on which a more detailed usage analysis is currently being carried out. This sample is made up of data from four aircraft from each CF squadron, two with "average" usage and two with "severe" usage based on the squadron mean FLEI rate. This will provide an indication of the variability of the usage across each squadron. Other criteria considered in the selection were:

- i) a dual seat aircraft was to be included in each of the average pairs for the predominantly "single seat" squadrons and a single in 410 squadron (the OTC);
- ii) each aircraft was required to have been in the squadron for the whole of the sample period;
- iii) the aircraft used for flight test were not included;
- iv) the aircraft were not to have received special fatigue life management procedures;
- v) the squadron must have been established at least 6 months prior to the sample period.

As discussed, the aim of the Usage Processing Task is to develop an MSDRS database of approximately 300 hours of flying (250 sorties) representative of the usage severity the CF and RAAF want simulated on the test. Three hundred hours is the typical annual cycle for the CF-18 and is also consistent with the block size used by the manufacturer's full scale test. Separate data bases representative of usage prior to and after the fitment of leading edge extension (LEX) fences are required. Figure 1 illustrates the methodology being used to identify the data base(s) necessary for the parametric analyses.

The raw MSDRS data for each of the thirty-two aircraft selected was processed at IAR to give four data files for each flight. Two of these files contained the flight incident records, one the fatigue code output and the fourth a summary of the stores carried/released and the landing parameters. These four files are the basic input for all the IAR developed usage processing and manoeuvre identification software.

The usage analysis of these aircraft included generating n_z exceedance tables, FITS tables, strain gauge peak-valley tables and roll rate exceedance tables grouped by IFOSTP mission code. Statistics for squadrons and the fleet are obtained by summing the individual aircraft tables. Initial examination of the strain gauge peak valley tables indicated that some aircraft had unserviceable gauges. These aircraft would not be included in the final database for manoeuvre identification.

The baseline usage for spectrum development, most likely from the squadron with the worst average usage, will be selected by studying the n_z and roll exceedances whilst ensuring that the mission mix is similar to the balance of the fleet. It should be noted that while each CF squadron has either air defence or strike as their prime task, each maintains currency in the non-prime task and therefore all squadrons fly all missions. An additional ten aircraft from the baseline squadron will be chosen to form the data base from which the final selection of missions will be made. The aim was to get a block of flights, not necessarily all by the same aircraft, that is representative in terms of stores configurations and mission mix.

PARAMETRIC APPROACH TO MANOEUVRE IDENTIFICATION

The CF-18 MSDRS recorder offers aircraft parameter and fatigue data that can be combined in a time ordered sequence. With reference to this data file, manoeuvres are defined as having occurred when significant parameters exceed defined threshold values. A series of standard manoeuvres are defined based on common descriptions used by the design/manufacture and operator communities. A "Non Standard" series of manoeuvres has been added to handle linked manoeuvres. A description of each variation has been prepared to enable each manoeuvre to be characterized [8]. Also, simulation models which use the MSDRS flight incident data file as input, have been developed to allow a visual presentation of a flight for detailed examination of manoeuvres. Table 6 lists the manoeuvre types required to describe the usage of the CF-18.

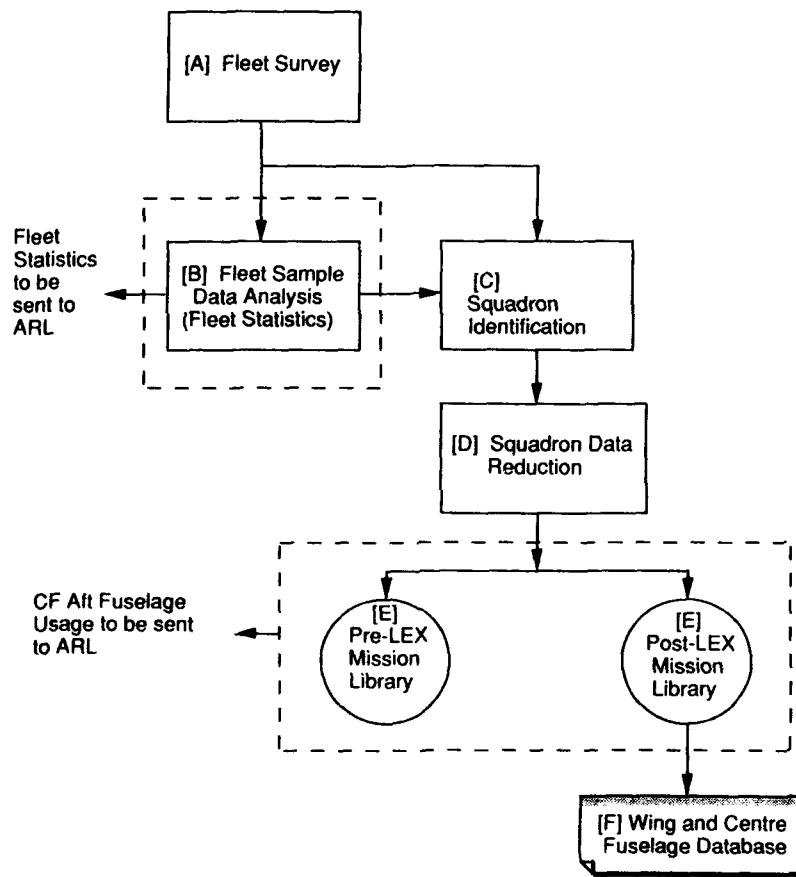


Figure 1 Data Processing Task

Standard Types

Name	Description	Content
Turn	Linked roll, pull, opposite roll	Roll Cycle, g cycle, roll cycle
Pull	From 1g to g peak, back to 1g	g cycle
Push	From 1g to g min, back to 1g	g cycle
Rolling Pull	Pull and roll (roll with g)	roll cycle (optional), roll with g, roll cycle (optional)
Roll	Start then stop roll	roll cycle

Non Standard Types

Name	Descriptions	Content
Extended Pull	Pull with more than one g peak	repeated g cycles
AOA Excursion	AOA alone above threshold	no significance for center fuselage fatigue
Roll Sequence	Roll same direction with changing rates	repeated roll cycles
Rolling & Pulling	Linked rolling and pulling, no roll w/g	roll cycles + g cycles repeated
Roll with g	Roll with g, more than one roll rate peak	roll cycles with g optional roll cycling, g cycles
Roll then Pull	Roll linked with pull	roll cycle, g cycle
Pull then Roll	Pull linked to roll	g cycle, roll cycle

Table 6: Manoeuvre Categories and Description

Within the automated process, additional description is added to the identification report to flag unusual events such as abrupt manoeuvres, high angles of attack, and/or unusual changes in speed or altitude over the course of a manoeuvre.

Detailed review of MSDRS data and coincident flight test data has shown that most of the non-standard fighter manoeuvres typical of air combat manoeuvring and ground attacks are combinations of the basic elements of the standard manoeuvres. For example, a typical basic fighter manoeuvre such as an Immelmann repositioning turn involves a symmetric pull to the vertical, an aileron roll on the way up, a pull to level flight and a 180 degree roll back to wings level and upright. The goal of this work is to define basic manoeuvre elements and to discretize the manoeuvres from the representative data base into combinations of these basic elements. This grouping is vital to the economic determination of the balanced loads conditions and to the definition of a realistic number of load lines for the test control system.

MANOEUVER CHARACTERISATION AND IDENTIFICATION

Manoeuvres may be characterized by describing the sequence and magnitude of significant parameters, such as roll rate and g level. For example, a symmetrical pull will involve some minimum start g (ie. near 1g), a g peak (over a defined threshold value), followed by a minimum end g (again near 1g). Roll rate will be insignificant throughout the manoeuvre. Another common basic manoeuvre is a turn. The aircraft is rolled and checked as the g rises to a peak value. The peak is held as long as desired. The g drops down from its peak as the aircraft is rolled back to level. Both roll rates and g peak must be over the appointed threshold values.

A two part approach is used to identify manoeuvres. First, the data is scanned and screened to identify time slices in which manoeuvres are not occurring. This is done by identifying the time slices when the roll rate is near zero, the angle of attack (AOA) is below 10 degrees and the g is approximately 1. All other time slices contain some sort of manoeuvre. To eliminate insignificant manoeuvres, threshold (or dead) bands are established around the zero roll rate and 1g levels. If the parameters are within these bands, no manoeuvre significant for fatigue is occurring. Current bands are 0 +/-20 deg/sec and 1g +/-1g for roll and g respectively. These values are very low and will be revised upwards as more analyses and coupon tests are carried out.

A logic process is used to set a flag whenever g, roll rate, and/or AOA exceed the defined thresholds. Thus, individual roll cycles, g cycles or AOA cycles will be identified if they rise above, then fall below the threshold values while the other parameters remain within the thresholds. With the active time slices identified, the second part of the process begins. Each time slice is evaluated by testing for g ranges, noting roll directions, calculating roll-through angles, the sequence of roll rate and g peaks and valleys. Manoeuvres are identified by comparing the observed data against the predefined manoeuvre characteristics. Data required to completely describe the manoeuvre is sent to an output file. Figure 2 presents an overview of the identification logic.

Rate of change of g and roll acceleration are calculated by differentiating the g and roll traces respectively. With the low recording rate for these parameters, this differentiation can be very inaccurate if only the discreet data points from the MSDRS are used. Comparison to detailed flight test data has shown that the slope of the straight line joining two discrete data points can underestimate the rate of change of g and roll accelerations by a factor of three. Results of this derivation can be improved by performing a cubic spline fit [9] of the initial g and roll traces before derivation. This procedure has been successfully applied, and comparisons between MSDRS and detailed flight test data show a maximum discrepancy of 38% and an average discrepancy of within 10% of actual values.

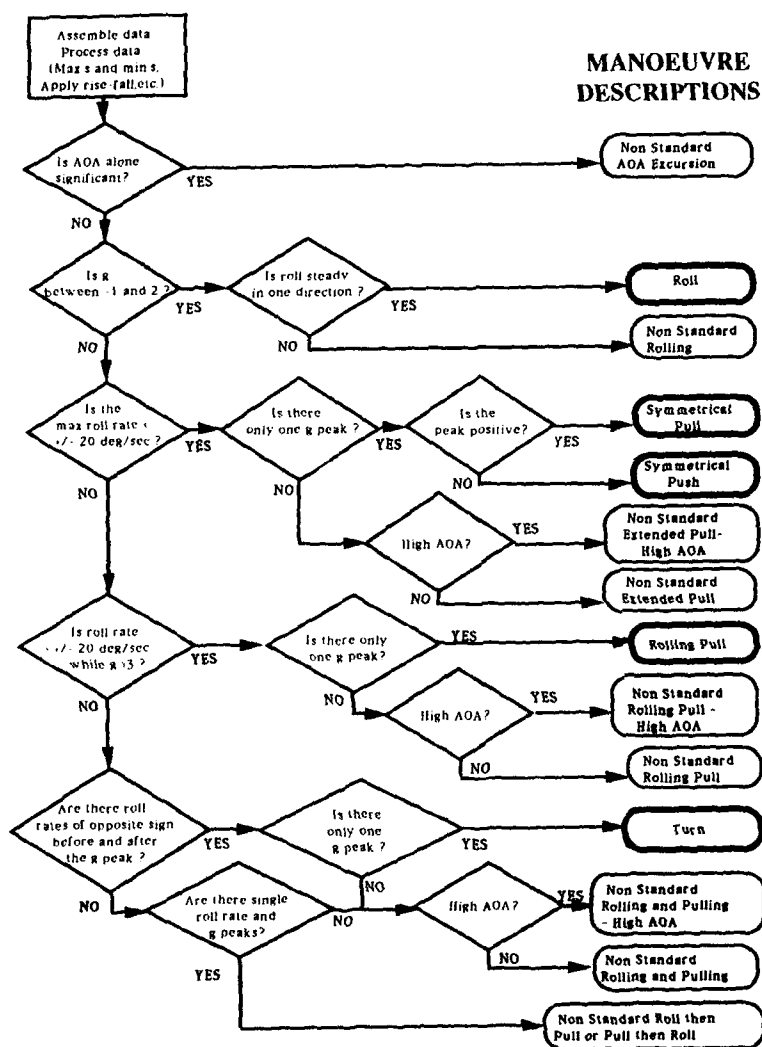


Figure 2 Manoeuvre Identification Logic

and time of occurrence. Fortunately for the derivation of the roll acceleration values, extra roll rate data is often recorded at times of roll initiation and roll check as the trigger values associated with the strain gauges on the aft fuselage are exceeded. This improves the roll rate splining process, particularly in the time frame of most interest, and therefore improves the accuracy of the derived roll acceleration.

MANOEUVRE DESCRIPTION

The goal of the manoeuvre description is to convey the information necessary to determine the significant aerodynamic and inertial loads on the test article. For each identified manoeuvre, this includes the following:

- Manoeuvre type
- Configuration
- Fuel Quantity
- Point-in-the-sky (altitude and speed)
- Roll data - peak rates
 - peak accelerations
 - roll-through angles
 - roll directions
 - roll with g data
- g data
- g peaks and valleys
- g onset (max rate of change of g)
- AOA data
- Stick positions
- extremes of position
- lateral and longitudinal

Table 7: Manoeuvre Categorization Information

To quantify fleet usage, a set of representative MSDRS files will be scanned. The significant data listed above will be recorded for every manoeuvre. Using the mission type identification codes assigned to each file, a database on each mission type will be prepared. The different missions will be assembled into a sequence of approximately 250 missions representing about 300 hours of flight. This data base will be used to determine the load sequence applied to the test article.

EXAMPLE OF MANOEUVRE IDENTIFICATION

- 6.75 g Turn

This example of a high-g turn is taken from early in a CF ACM mission. The manoeuvre identification process first eliminates the inactive part of the flight by determining periods of g, roll and/or high AOA activity. In this case, a roll precedes a g excursion which is followed by another roll opposite in direction to the initial roll. This is the typical sequence expected for a turn. At no point is there a significant roll rate (>20 deg/sec) while g is over 3, which is the current transition point to the "roll with g" manoeuvre description. Nevertheless, the initial roll-in, pull and final roll-out are seen as "linked" since the finish of the roll-in overlaps the start of the pull, and the roll-out begins before the pull is completed.

The pattern is clearly illustrated in Figure 3. This displays some of the significant parameter traces for the 12 seconds over which this manoeuvre occurs. The initial roll-in is seen on the roll rate trace, the negative value corresponding to a roll to the left. The roll rate is checked back to about zero, as simultaneously the g begins to rise (see the Nz trace). The pilot has banked his aircraft, and is now pulling the g. In this ACM manoeuvre, as the pilot held the pull, he allowed the speed to bleed down by 125 knots and the AOA to rise to about 25 degrees. After holding the g for several seconds, he released back pressure on the stick and rolled out back to the right, as seen on the roll trace.

Table 8 shows the Flight Incidence data that was recorded once per second by the MSDRS system during the manoeuvre. Table 9 shows the Fatigue Triggered data that was recorded upon strain reversals and upon g peaks and valleys over the course of the manoeuvre. This data is sampled at a rate of 20 Hz, and thus the minimum step between data points is .05 sec. There is also some duplication as different gauges reach peak or valley strains simultaneously. There were 16 unique lines of data recorded for this manoeuvre, out of 24 triggered.

Table 10 presents a summary file pulling the Flight Incidence and unique Fatigue Triggered data together. It is upon this data that the manoeuvre identification routine (Figure 2) is applied. The fatigue triggered data includes only time, Nz and roll rate information. There is no roll acceleration or rate of change of g information recorded. The values shown are derived by differentiating a curve fit to the data points by a cubic spline interpolation routine [9]. The routine selected does not produce spurious inflection points or overshoots; it merely fits the existing data with a smooth curve.

Table 11 presents the detailed version of the manoeuvre identification output for this manoeuvre. The header contains a summary of the pertinent information required to define, in this case, a turn. The start and end times are listed, as are the initial g valley, maximum g peak and final g valley. The roll-in information includes the maximum initiation acceleration, maximum roll rate and maximum check acceleration. The same information is provided for the roll-out at the end of the turn. Bank angles are calculated by integrating the roll trace. What follows the header is a time-sequenced list of all relevant parameters. Only extreme values (turning points) are listed, in addition to the initial and final parameter values (in this case at 1368.05 and 1380.0 seconds from take-off respectively). All the parameters are run through a rise-fall routine to eliminate insignificant fluctuations. This detailed list also includes AOA data and stick position extremes. Point-in-the-sky information is recorded. Similar lists of relevant data are derived for each of the other manoeuvre types discussed earlier.

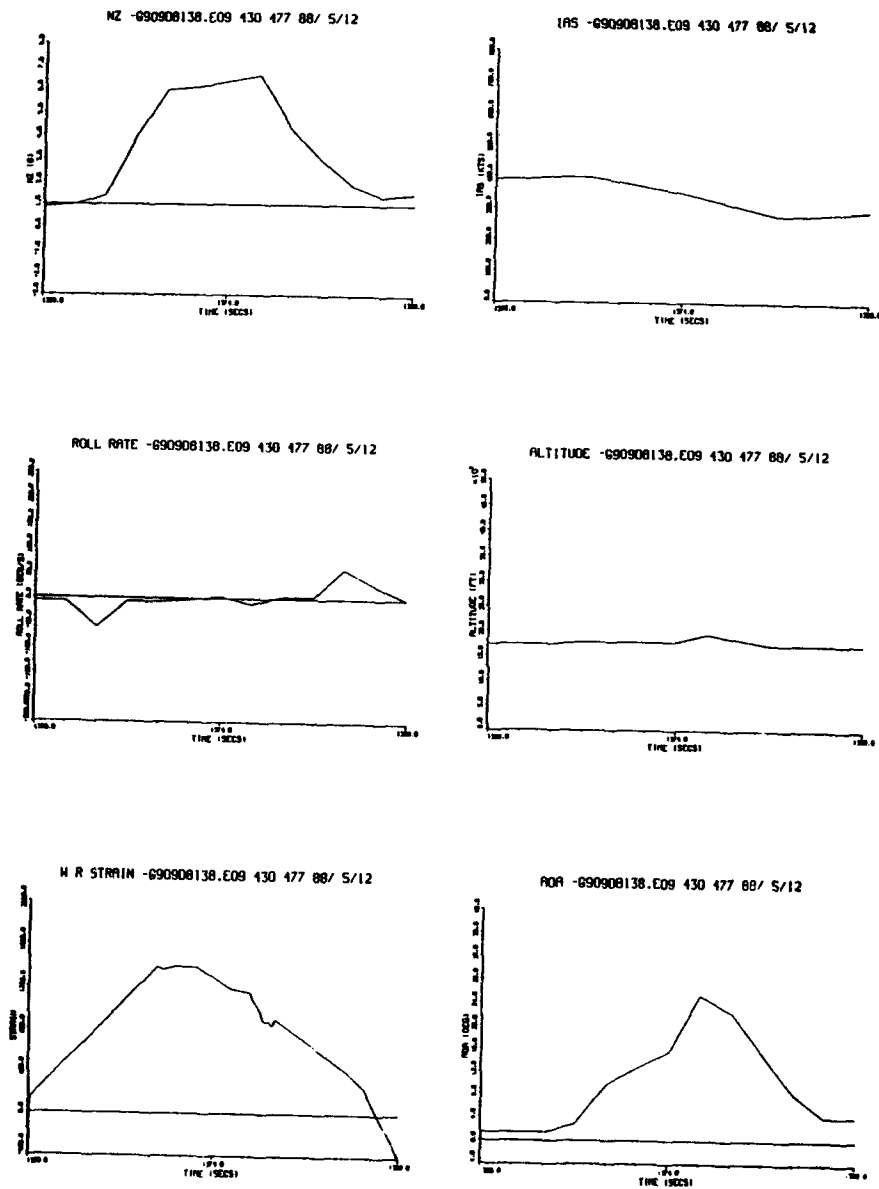


Figure 3 6.75g Turn Manoeuvre

ICF-18 NEDAS DATA RETRIEVAL - "AIRSTRIP VERSION 4.0 3/8/90
 ***** CURS 48 (A) *****
 FLIGHT INCIDENT RECORDING - AIRCRAFT PARAMETERS
 TAIL ID : 908 FLIGHT TIME : 3214.00 MISSION TYPE : 40 DATE : 88/ 5/12 STORES : 11000000110000000000
 SUMCHECK 0478083140000000000
 ADP FILE 880808139.809
 FROM BLOCK 430 TO BLOCK 477

HC	INT	BARO	PRNS	RAD	WDG	LONG	L	R	L	OUT	R	OUT	P	PITCH	DOWN	PRNS	L	R	LAT	L	R	ROLL
TIME	AIR	ALT	ALT	ALT	ALT	ALT	STK	T.R.	T.R.	L.R.	L.R.	RAIZ	ACC	ACC	ACC	STAB	STAB	STAB	STK	ATL	ATL	ATL
SPEED							POS	POS	POS	POS	POS	POS	(7/32)	(7/32)	POS	POS	POS	POS	POS	POS	POS	POS
1366	1369.4	382	17408	0	1.4	-0.2	0	1.4	-0.2	0	1.4	-0.2	0	1.4	-0.2	0	1.4	-0.2	0	1.4	-0.2	0
1369	1369.4	382	17408	0	1.4	-0.1	0	1.4	-0.1	0	1.4	-0.1	0	1.4	-0.1	0	1.4	-0.1	0	1.4	-0.1	0
1370	1370.4	400	17408	0	1.4	1.0	0	1.4	1.0	0	1.4	1.0	0	1.4	1.0	0	1.4	1.0	0	1.4	1.0	0
1371	1371.4	400	17408	0	2.8	2.3	4.88	4.88	2.8	2.8	12	2.8	12	2.8	12	32	-3.1	-2.4	-0.4	-1.0	0.4	-8
1372	1372.4	384	17920	0	9.8	2.7	0	9.8	2.7	0	9.8	2.7	0	9.8	2.7	0	9.8	2.7	0	9.8	2.7	0
1373	1373.4	384	17920	0	12.7	3.1	0	12.7	3.1	0	12.7	3.1	0	12.7	3.1	0	12.7	3.1	0	12.7	3.1	0
1374	1374.4	382	17920	0	15.5	4.0	0	15.5	4.0	0	15.5	4.0	0	15.5	4.0	0	15.5	4.0	0	15.5	4.0	0
1375	1375.4	328	18464	0	25.3	1.8	0	25.3	1.8	0	25.3	1.8	0	25.3	1.8	0	25.3	1.8	0	25.3	1.8	0
1376	1376.4	304	18432	18944	0	22.5	0.8	0.00	0.00	30.8	30.8	-4	-2.8	-140	352	-7.4	-7.7	0.3	0.0	-0.7	4	274.2
1377	1377.4	280	17408	0	15.5	-0.1	0	15.5	-0.1	0	15.5	-0.1	0	15.5	-0.1	0	15.5	-0.1	0	15.5	-0.1	0
1378	1378.4	280	17408	0	8.4	-0.2	0	8.4	-0.2	0	8.4	-0.2	0	8.4	-0.2	0	8.4	-0.2	0	8.4	-0.2	0
1379	1379.4	288	17408	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0
1380	1380.4	296	17408	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0	4.2	0.4	0

Table 8 Flight Incident Data

TIME	IAS	ALT	AOA	LONGSTK	WZ	LATSTK	ROLL	WZDOT	ROLLDOT	RUDDER
1366.00	392	17408	1.4	-0.2	0.87	-0.1	-8	-3.1	121	0
1366.02									73	
1366.05	391	17408	1.4	-0.2	0.77	-0.1	-4	0.1	26	0
1366.90									-22	
1369.00	392	17408	1.4	-0.1	0.99	-0.3	-8	0.3	-25	0
1369.50									-71	
1370.00	400	17408	1.4	1.0	1.37	-0.5	-60	0.8	-2	5
1370.55									74	
1371.00	400	17920	2.8	2.3	3.98	-0.4	-8	2.4	17	8
1371.95									14	
1372.00	384	17920	9.8	2.7	5.97	-0.3	-8	1.4	19	4
1372.05									70	
1372.15	381	17920	10.2	2.8	5.63	-0.3	0	0.7	2	4
1372.25									-54	
1372.35	378	17920	10.8	2.8	5.89	-0.2	-7	0.4	4	4
1372.60									5	
1372.75	372	17920	12.0	3.0	6.04	0.0	-5	0.2	4	6
1372.85									4	
1373.00	368	17920	12.7	3.1	6.09	0.1	-4	0.1	3	7
1373.25									-5	
1373.45	361	17920	14.0	3.5	6.06	0.2	-5	0.1	5	7
1373.80									20	
1374.00	352	17920	15.5	4.0	6.34	0.3	4	0.2	13	7
1374.05									12	
1374.55	339	18942	21.2	2.9	6.32	0.5	8	0.3	0	6
1374.80									-65	
1375.00	328	19456	25.3	1.8	6.59	0.8	-12	0.8	-3	5
1375.05									28	
1375.15	324	19302	25.5	1.7	6.75	0.6	-9	0.7	-4	4
1375.20									-14	
1375.25	322	19200	25.5	1.6	6.75	0.6	-10	-0.7	2	3
1375.35									4	
1375.45	317	18995	25.1	1.4	6.57	0.5	-9	-1.1	7	2
1375.50									10	
1375.55	314	18892	24.9	1.3	6.44	0.4	-8	-1.9	8	1
1375.57									18	
1375.65	312	18790	24.5	1.2	6.23	0.4	-8	-2.0	29	1
1375.70									29	
1375.75	310	18688	24.0	1.1	6.03	0.3	-5	-2.1	29	0
1375.80									29	
1375.85	307	18585	23.5	1.0	5.78	0.3	-2	-2.5	29	0
1375.87									26	
1375.95	305	18483	22.8	0.9	5.49	0.3	-1	-3.3	24	0
1375.97									16	
1376.00	304	18432	22.5	0.9	4.35	0.3	4	-2.8	8	0
1376.05									7	
1377.00	280	17408	15.5	-0.1	2.98	1.3	4	-0.1	-4	1
1377.15									-11	
1377.35	275	17332	13.0	-0.3	2.99	1.3	1	-0.6	-2	1
1377.70									135	
1378.00	280	17408	8.4	-0.2	1.88	1.1	80	-0.6	4	1
1378.15									-80	
1378.25	282	17408	6.9	-0.1	2.04	0.9	51	-0.5	-18	2
1378.55									28	
1378.85	286	17408	4.8	0.3	1.31	0.5	57	0.2	-34	7
1378.90									-223	
1379.00	288	17408	4.2	0.4	1.37	0.4	28	0.2	-102	8
1379.05									-95	
1380.00	296	17408	4.2	0.4	1.49	0.2	0	0.0	54	4

Table 10 Composite Flight Incident/Fatigue Data

MANOEUVRE TYPE	TIME DATA	SPEED DATA	ALT DATA	G DATA	ROLL-IN DATA	ROLL-OUT DATA
6.75G TURN	START TM-1368.06 AT 381 KTS, 17400FT	START G= 0.77	ADOT(INIT)= -71.	ADOT(INIT)= 135.		
EXTRA ROLL LEFT	END TIME-1380.00 AT 296 KTS, 17400FT	PEAK G= 0.76	RR MAX = -80.	RR MAX = 60.		
	2.4 G/SEC DURATION= 11.964T AVG 333.KTS/10100.FT	END G= 1.31	ADOT(CHEK)= 74.	ADOT(CHEK)= -223.		
			ROLL-IN BANK = -80.	ROLL-OUT BANK = 87.		
RICK AOA	SPEED VARIATION = 125. KNOTS	ALTITUDE VARIATION = 2124. FEET				
DETAILS:						
FUEL WEIGHT OVER MANOEUVRE	= 4096 LBS.					
	TIME -1368.06	G VALLEY = 0.77 G				
	TIME -1368.06	MINIMUM AOA = 1.4 DEG				
	TIME -1368.06	ROLL ACCELERATION PEAK = 20. DEG/SEC ²				
	TIME -1368.06	ROLL RATE PEAK = -4. DEG/SEC				
	TIME -1368.06	MAXIMUM LATERAL STICK DEFL.= -0.1 INCHES				
	TIME -1368.06	MINIMUM LONGITUDINAL STICK DEFL.= -0.2 INCHES				
	TIME -1368.06	MINIMUM RUDDER PEDAL FORCE = 0.0 POUNDS				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 391 KNOTS WITH G = 0.77				
	TIME -1369.50	ROLL ACCELERATION VALLEY = -71. DEG/SEC ²				
	TIME -1370.00	ROLL RATE VALLEY = -80. DEG/SEC				
	TIME -1370.00	MINIMUM LATERAL STICK DEFL.= -0.6 INCHES				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 400 KNOTS WITH G = 1.37				
	TIME -1370.55	ROLL ACCELERATION PEAK = 74. DEG/SEC ²				
	TIME -1371.00	MAXIMUM RUDDER PEDAL FORCE = 8.0 POUNDS				
	AT POINT IN THE SKY: ALT = 17820 FT	SPEED = 400 KNOTS WITH G = 3.98				
	TIME -1371.05	ROLL ACCELERATION VALLEY = 14. DEG/SEC ²				
	TIME -1372.05	ROLL ACCELERATION PEAK = 70. DEG/SEC ²				
	TIME -1372.15	ROLL RATE PEAK = 0. DEG/SEC				
	AT POINT IN THE SKY: ALT = 17820 FT	SPEED = 381 KNOTS WITH G = 5.03				
	TIME -1372.25	ROLL ACCELERATION VALLEY = -54. DEG/SEC ²				
	TIME -1372.35	ROLL RATE VALLEY = -7. DEG/SEC				
	AT POINT IN THE SKY: ALT = 17820 FT	SPEED = 376 KNOTS WITH G = 5.89				
	TIME -1373.00	ROLL ACCELERATION PEAK = 20. DEG/SEC ²				
	TIME -1374.00	MAXIMUM LONGITUDINAL STICK DEFL.= 4.0 INCHES				
	AT POINT IN THE SKY: ALT = 17820 FT	SPEED = 362 KNOTS WITH G = 6.34				
	TIME -1374.55	ROLL RATE PEAK = 8. DEG/SEC				
	AT POINT IN THE SKY: ALT = 18942 FT	SPEED = 339 KNOTS WITH G = 6.32				
	TIME -1374.80	ROLL ACCELERATION VALLEY = -85. DEG/SEC ²				
	TIME -1375.00	ROLL RATE VALLEY = -12. DEG/SEC				
	AT POINT IN THE SKY: ALT = 18456 FT	SPEED = 328 KNOTS WITH G = 6.59				
	TIME -1376.05	ROLL ACCELERATION PEAK = 20. DEG/SEC ²				
	TIME -1376.15	G PEAK = 0.76 G				
	TIME -1376.15	MAXIMUM AOA = 26.5 DEG				
	AT POINT IN THE SKY: ALT = 19302 FT	SPEED = 324 KNOTS WITH G = 6.76				
	TIME -1376.20	ROLL ACCELERATION VALLEY = -14. DEG/SEC ²				
	TIME -1376.75	MINIMUM RUDDER PEDAL FORCE = 0.0 POUNDS				
	AT POINT IN THE SKY: ALT = 18000 FT	SPEED = 310 KNOTS WITH G = 6.03				
	TIME -1376.85	ROLL ACCELERATION PEAK = 29. DEG/SEC ²				
	AT POINT IN THE SKY: ALT = 18605 FT	SPEED = 307 KNOTS WITH G = 5.78				
	TIME -1377.15	ROLL ACCELERATION VALLEY = -11. DEG/SEC ²				
	TIME -1377.35	MAXIMUM LATERAL STICK DEFL.= 1.3 INCHES				
	TIME -1377.35	MINIMUM LONGITUDINAL STICK DEFL.= -0.3 INCHES				
	AT POINT IN THE SKY: ALT = 17332 FT	SPEED = 276 KNOTS WITH G = 2.99				
	TIME -1377.70	ROLL ACCELERATION PEAK = 135. DEG/SEC ²				
	TIME -1378.00	ROLL RATE PEAK = 60. DEG/SEC				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 260 KNOTS WITH G = 1.86				
	TIME -1378.15	ROLL ACCELERATION VALLEY = -80. DEG/SEC ²				
	TIME -1378.25	ROLL RATE VALLEY = 51. DEG/SEC				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 262 KNOTS WITH G = 2.04				
	TIME -1378.55	ROLL ACCELERATION PEAK = 20. DEG/SEC ²				
	TIME -1378.85	G VALLEY = 1.31 G				
	TIME -1378.85	ROLL RATE PEAK = 57. DEG/SEC				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 260 KNOTS WITH G = 1.31				
	TIME -1378.90	ROLL ACCELERATION VALLEY = -223. DEG/SEC ²				
	TIME -1379.00	MINIMUM AOA = 4.2 DEG				
	TIME -1379.00	MAXIMUM LONGITUDINAL STICK DEFL.= 0.4 INCHES				
	TIME -1379.00	MAXIMUM RUDDER PEDAL FORCE = 0.0 POUNDS				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 260 KNOTS WITH G = 1.37				
	TIME -1380.00	G PEAK = 1.40 G				
	TIME -1380.00	ROLL ACCELERATION PEAK = 54. DEG/SEC ²				
	TIME -1380.00	ROLL RATE VALLEY = 0. DEG/SEC				
	TIME -1380.00	MINIMUM LATERAL STICK DEFL.= 0.2 INCHES				
	TIME -1380.00	MINIMUM RUDDER PEDAL FORCE = 0.0 POUNDS				
	AT POINT IN THE SKY: ALT = 17400 FT	SPEED = 260 KNOTS WITH G = 1.40				

Table 11 Detailed Manoeuvre Identification Output - 6.75g Turn

MANOEUVRE DISTRIBUTIONS BY MISSION TYPE

Typical flights representative of the IPOSTP mission categories have been subjected to the manoeuvre identification process. Table 12 provides the breakdown of manoeuvres by mission type. The most frequent manoeuvre for all mission types is the roll. Typically 60-65% of the manoeuvres are standard rolls. This may be partially explained by the use of wing flashes and wing rocking for communicating intent between aircraft during formation flying under radio restrictions. More likely, it is the result of the very low roll rate threshold (20 deg/sec) used for these early results. The roll is a manoeuvre that is sensitive to the PITS at which it is flown and it is also the manoeuvre that the CF-18 MSDRS system is most restricted in quantifying since roll accelerations are not recorded and roll velocities are recorded only once per second.

Based on the limited data analyses done to date, 80-85% of the manoeuvres can be identified as standard. It is expected that refinement of the current discrimination boundaries will lead to even more manoeuvres being categorized as standard, thus reducing the number of unique cases that must be considered for the balanced load calculations.

CONCLUSIONS

A parametric based approach to spectrum development for the CF-18 International Follow-On Structural Test Program has been introduced. Manoeuvre classification and identification methodologies using the CF-18 MSDRS data files have been developed that allow the sequential identification of structurally damaging manoeuvres.

Advanced aircraft with DFCS require innovative techniques in developing usage spectra. In particular, the sensitivity of the aircraft control laws to the PITS at which manoeuvres are flown has introduced new variables that must be considered when developing local stress histories.

Multi-channel computer based data monitoring systems are vital to the development of relevant usage statistics on these aircraft. At an early stage in development of the data collection philosophy, the data reduction philosophy must be considered. This data reduction philosophy must address both the selection and generation of fleet usage statistics and the generation of local stress histories to support analyses and tests. Reference to this data reduction philosophy will assist in the determination of the number and type of data channels, the frequency of recording and the determination of the data collation and storage formats.

Specific limitations, with respect to the parametric approach adopted for spectrum development, of the MSDRS system used in the CF-18 have been identified.

These include:

- a) no angular acceleration data;
- b) no useful yaw information (only very crude yaw rate data is available);
- c) low sampling rate of 1/sec is inadequate for some abrupt (rapid) manoeuvres;
- d) landing information does not allow determination of sink speed at touchdown;
- e) parametric system data not recorded for strain reversals.

ACKNOWLEDGMENTS

This work has been performed under a Statement of Agreement between the Institute of Aerospace Research of the National Research Council of Canada and the Canadian Department of National Defence. The project is partially funded by DND Financial Encumbrance 846588FAHRA and by the IAR Structures, Structural Dynamics and Acoustics Program.

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MANOEUVRE / MISSION BREAKDOWN									
MISSION CODE	DESCRIPTION	TURNS	PULLS	PUSHES	ROLLS	ROLLING PULLS	NON STANDARD	TOTALS	
A	CLEARHOOD	3	23	0	80	6	24	136	
B	IFR	0	28	0	67	1	6	102	
C	AIR INTERCEPT	18	82	0	357	13	83	553	
D	ACM	24	49	0	243	19	62	397	
E	A2G WEAPON	25	49	0	245	17	82	418	
F	A2S TACTICS	15	30	0	219	15	76	355	
G	A2A GUNNERY MISCELLANEOUS	7	26	1	129	4	37	204	

Table 12 Manoeuvre/Mission Breakdown

FATIGUE TESTING AND TEAR DOWN OPERATIONS ON AIRBUS A320 FORWARD FUSELAGE

by

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SUMMARY

The structural design substantiation of commercial transport airplane is provided by full static and fatigue analyses supported by testing of complete primary structures.

The fatigue and damage tolerance testing on full scale specimen representative of production airplane and the teardown inspections performed at the end of the test permit to collect a lot of informations.

Main objectives of fatigue tests are on one hand, to early identify weak points in primary structure and to quickly define corrective actions on in-service and production airplanes, on the other hand to check the efficiency of the inspection methods, to validate the calculation methods, to justify allowable damage (scratches/dents) and typical repairs of structural repair manual, to study the propagation of artificial damages which are introduced during the test. The aim of tear down inspections is to verify and validate the inspection methods applied during the test on assembled structure and in addition to find hidden cracks with special non destructive inspection methods.

Furthermore from fatigue testing followed by tear down, we are expecting to find areas where there is a risk of wide spread fatigue damage and to perform damage tolerance assessment on the basis of realistic cracking scenarios.

Finally, findings are analysed for revision of established maintenance tasks in the maintenance review board document and repercussions on design.

1 - INTRODUCTION

The philosophy on which the structural design of a commercial airplane is based on, takes into account the airlines interest in three major fields:

- Safety.
- Long service.
- Low cost.

Structural fatigue and damage tolerance certification and design concept are governed by JAR25 (FAR25) regulations, chapter 25.571 and subsequent advisory circulars.

In the framework of airplane certification, full scale fatigue tests are strongly recommended (JAR 25.571).

Main objectives of fatigue testing is to early identify weak points in the primary structure and hence to quickly define corrective actions to eliminate them and finally to check that the design objectives are met to fulfil the requirements concerning the structural life of the airplane in commercial service.

Fatigue testing followed by tear down inspections also enable the structural inspection programme to be readjusted by taking into account the measured crack propagation rates, fail-safe behaviour and crack history.

2 - PHILOSOPHY

The fatigue and damage tolerance testing on full scale specimen representative of production airplane and the tear down operations performed at the end of the test permit to collect a lot of informations.

MAIN OBJECTIVES OF FULL SCALE FATIGUE TESTING ARE TO :

- Early identify weak points in primary structure.
- Quickly define correctives actions on in service and production airplanes.
- Check and validate the inspection methods.
- Validate the calculation methods applied in the framework of certification and maintenance tasks.
- Study the propagation of artificial damages which are introduced during the test on major structural components and support the crack propagation analyses.

- Justify allowable damages (scratches / dents) and typical repairs of structural repair manual (S.R.M.) in different locations at the beginning of the test.
- Furthermore, expect from fatigue testing followed by tear down, to find areas where there is a risk of wide spread fatigue damage (multiple site damage and multiple element damage) and perform damage tolerance assessment on the basis of realistic cracking scenarios.

MAIN OBJECTIVES OF TEAR DOWN OPERATIONS ARE TO :

- Confirm findings and determine the real value of crack length found during fatigue testing.
- Find hidden cracks on full scale specimen and develop new non-destructive inspection procedures.
- Find small cracks in a specific area (multiple site and/or multiple element damage susceptibility).
- Verify and validate the inspection methods performed during the test.
- Investigate minimum detectable crack length provided by selected inspection method, and to correlate it with the fractographic results.

The use of tear down results permit to confirm fatigue and damage tolerance analyses, to justify the selection of structural significant items (SSIs), to revise the established maintenance tasks in the maintenance review board document (M.R.B) and to evaluate repercussions on design.

3 - A320 : FULL SCALE FATIGUE TESTS

- GENERAL

- Five major full scale fatigue tests have been carried out on the Airbus A320 :
 - Forward fuselage
 - Wing/centre fuselage
 - Rear fuselage
 - Horizontal stabilizer
 - Vertical stabilizer

A multi-section testing permits a faster execution of the programme and a fewer compromises in the load spectrum.

Additional component fatigue tests have been carried out on pylon, rear pressure bulkhead, slats, engine mounts and landing gears (nose and main gears).

- AIM

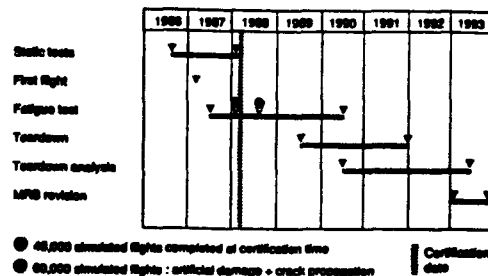
The aim is to perform 120 000 simulated flights corresponding to 2.5 lives (fatigue life design aim : 48 000 flights).

Testing is realized in two main phases :

Phase 1 : up to 60 000 simulated flights.
Phase 2 : from 60 000 simulated flights to testing completion, introduction of artificial damages and repairs.

The test schedule given below shows the dead lines of fatigue and damage tolerance testing, tear down results and analysis, maintenance review board revision, in the framework of airplane certification.

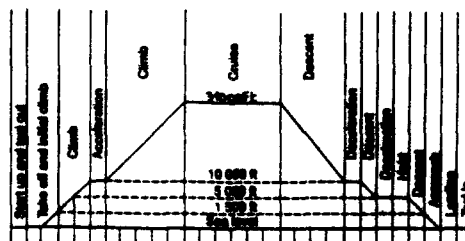
Major structure test schedule



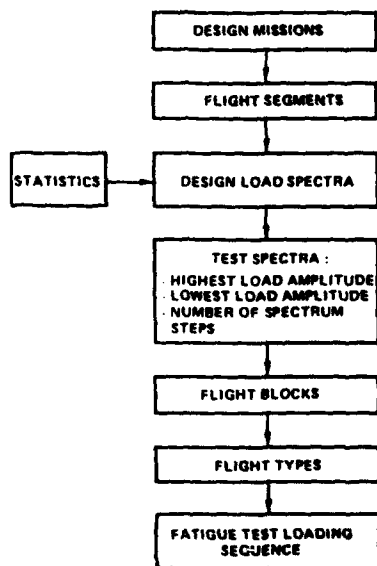
- LOAD SPECTRUM

These tests are performed according to the "flight by flight" procedure, which has been recognized for some years to be the most suitable procedure for simulating the loads encountered by airplane in service.

The typical mission to be considered for defining the ground, air and pressured load spectra is described below. This mission also provides the occurrences of gust for one flight and the occurrences of loads on the ground for one flight. This must be taken into account when defining the cumulative frequency distribution.



The following flow chart describes the selected procedure for developing the loading programmes of A320 major fatigue tests.



To respect as far as possible the definition of the loading spectrum, the test philosophy takes into account the following basic assumptions :

- The choice of the size of the blocks is based on the wish to exclude load values that are too close to the static sizing limit loads from the loading spectrum, since these loads may stop damage initiation or propagation, by retardation effects.
- The average number of gust cycles and ground load cycles per flight are voluntarily limited in order to get a convenient testing duration.
- These omission levels are justified elsewhere by separate tests intended to quantify the effect of the omission of small cycles on fatigue lives and crack propagation, or by the application of equivalent damage laws.

The number of typical flights for building up the sequence can be chosen at will : by example we

propose 8 types of flights, designated A to H to describe the airborne environment and 3 sub-types P to R, to describe the environment on the ground. Finally the fatigue test loading sequence is built, considering the following major points :

- the number of flight types (AP to HR)
- the number of flights in a block
- the sequence of loads in one block
- the sequence of flights in a block

- RESULTS

The inspections are performed by the test laboratory or the manufacturer, responsible of the test process according to an inspection programme which defines :

- the areas to be inspected,
- inspection methods,
- classes and levels of inspections.

When a damage is found, the following informations are given in a damage documentation sheet :

- damaged part description
- component's manufacturer responsible
- precise area and the applied inspection method (visual, NDT...)
- drawing number
- damage extent
- number of simulated flights
- inspection date
- sketches and/or photos of the damaged area
- possible causes of the damage given by the test laboratory
- test repercussions
- damage time history
- stress office comments about the damage causes and proposed repercussion on production and in service airplanes.

Microfractographic expertises are required to have a thorough knowledge of the causes and the characteristics of the damages (crack propagation curve, initiation date). 160 findings (fatigue damages) have been found up to the end of fatigue tests. The majority of findings are minor and found on secondary structure or on non-safety related items.

The early findings on primary structure, have been corrected before the first airplane delivery. Late findings may lead to in-service and production modifications, or inspections.

- TEAR DOWN OPERATIONS

At the end of the test, a final inspection is performed by the test laboratory according to the

basic inspection programme, then the full scale specimen is cut out if necessary for transportation. The AEROSPATIALE parts are stocked in a specific building in Blagnac, called "Hall tear-down".

We are expecting from the tear down operations :

- To verify and to validate the inspection methods applied during the test on assembled structure,
- To find hidden cracks with special non-destructive inspection methods,
- To find areas where there is a risk of wide spread fatigue damage,
- To confirm findings and estimated crack lengths,
- To validate and to improve the fatigue and crack propagation models.

The tear down inspection plan is established by the stress office and is proposing the following inspection tasks :

- In assembled condition, visual inspections of damages or defects, followed by special detailed inspections (non-destructive inspection methods selected : ultrasonic, eddy current, Xray...).
- In disassembled condition into detail parts, the same inspection levels are performed, visual research followed by special detailed inspection.
- Microfractographic expertises are performed in case of request from the stress office, for the determination of the crack propagation curve and the initiation date.

At least, a report including inspections results and photographic shots taken during the tear down is edited.

The major informations contained are :

- sketches, references of drawings,
- non-destructive inspection results,
- microfractographic examinations,
- crack propagation curves and initiation date,
- causes of damages.

All damages are introduced in a computer data bank, which summarizes all the informations gained.

The data bank is used by :

- the design office to improve design of new airplanes,

- the stress office for in-service repercussions, maintenance tasks revision, supplementary inspection programme,

- non destructive inspection specialists to verify the efficiency of inspection procedures.

4 - FULL SCALE FATIGUE TEST : FORWARD FUSELAGE

4.1 General

This test called EF1 (essai de fatigue n° 1) is a part of the major fatigue tests carried out on the Airbus A320.

The loads applied during the test include :

- take off, landing, taxiing
- air loads : gust and manoeuvre
- cabine pressure

120 000 simulated flights corresponding to 2.5 lives have been performed during the test .

DEFINITION OF TEST SPECIMEN

The specimen called EF1 consists of the fuselage nose sections 11 to 14 from frame 1 to frame 35.

The specimen rear part which is reinforced between frames 33 and 35, carries a "T" section circumferential fitting providing a bolted connection with the jig.

Sections 11 to 14 primary structure is equipped with :

- 2 LH/RH front passenger doors,
- a simplified on board stairs door,
- the R/H cargo door,
- all flight compartment/cabin/forward cargo compartment floor support structure up to frame 35,
- 4 cabin windows,
- all flight compartment windshields,
- all windshields are fitted with their heating element network,
- a dummy nose landing gear.

TEST SEP-UP

The loading of the specimen is carried out through the following devices on the fuselage : loading jacks, loading trees of the cabin floor and the pressurization is achieved through the test jig.

The ground loads are applied to the nose landing gear by three jacks.

4.2 Test load spectrum

According to the procedure followed for developing the loading programmes of A320 major fatigue tests, the test philosophy and the basic assumptions are :

- as the forward fuselage fatigue life design aim is 48 000 flights, the test has to cover 120 000 simulated flights, applied in identical blocks of 4 800 flights. 25 blocks in all have been applied to the specimen.
- the average number of gust cycles per flight has been limited to seven in order to get a convenient testing duration. The result is that the cumulative distribution for one life time (48 000 flights) of gust loads has been limited to $48\,000 \times 7 = 336\,000$ cycles.

The number of typical flights for building up the block of 4 800 flights has been chosen at will : for the forward fuselage test, 8 types of flights, called A to H, to describe the airborne environment and 3 sub-types P, Q and R, to describe the environment on the ground.

The occurrences of the different typical flights per block of 4 800 flights with the ground load flight load combinations are shown in the following table : 21 types of flights in all ranging from the most severe flight AR to standard flights HP, HQ and HR.

Typical Flights	A	B	C	D	E	F	G	H
P	-	-	1	3	7	10	48	139
Q	-	2	5	10	41	99	202	737
R	5	7	14	47	123	295	753	2210

Finally the average number of cycles per flight is 22, and the average number for an airplane life is 1.056 000 cycles.

A conservative value of 2 % of the flights is considered as training flights.

Briefly, the test loading programme considers the following major points :

- 1) 21 flight types AR to HR have been defined

- 2) one test block of 4 800 flights (start with type HR and end with type HR)
- 3) The sequence of cycles in one flight has the following restrictions
 - the order of gust is randomized,
 - all ground load cycles start with a downwards taxi bump load,
 - all gust load cycles start with an upgust load,
 - an up gust is always followed by a downgust.
- 4) For the distribution of flight types within one block of 4 800 flights, the rough flights AR to DR are positioned based on an "equally" spaced distribution and the smooth flights EP to HR are positioned by a random number generation programme.
- 5) The cabine pressure differential loads have been linearly applied to the specimen for mission segments between climb and descent with a maximum at cruise ($\Delta P = 556$ mb).

4.3 Findings (fatigue damages)

During the full scale fatigue test, inspections of the specimen are performed, according to an inspection programme which defines classes and levels of inspections for the whole structure.

The results of inspections are recorded and each detected damage is the subject of a damage documentation sheet which details the damage and defines the repair or modification to be embodied on in service or production airplanes.

At the end of the test, 31 findings have been discovered. (2 under tear down inspections)

- 6 findings treated in the maintenance review board revision,
- 9 minor findings,
- 1 finding leading to retrofit before airplane delivery,
- 13 findings leading to 10 service bulletins.

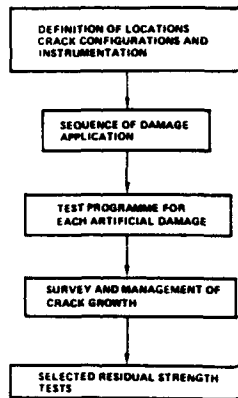
4.4 Damage tolerance tests

After having performed 60 000 simulated flights on the specimen, crack propagation tests have been carried out on specially selected areas of the structure by the introduction of artificial damages.

The objectives are to :

- obtain data to substantiate the analysis
- obtain data to confirm inspection intervals of the maintenance review board document

The flow chart hereafter presents the damage tolerance test process.



4.5 Tear down operations

An overall tear down inspection is performed after the end of the test. The tear down results and analysis are in progress, the target for completion is scheduled for end of 1993.

For section 11/12, 63 structural sub-assemblies will be reviewed for visual inspections.

For section 13/14, 31 structural sub-assemblies will be reviewed for visual inspections.

5 - CONCLUSION

For new design, the AEROSPATIALE policy is to perform fatigue testing on full scale specimen representative of production airplane with subsequent tear down operations.

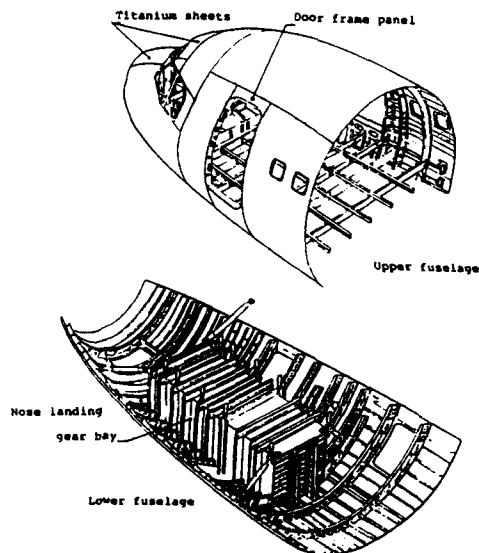
Indeed, fatigue testing is the privilegiate way :

- To substantiate fatigue and damage tolerance analyses (for instance to improve the maintenance programme based on real findings).
- To quickly define the corrective actions on the fleet.

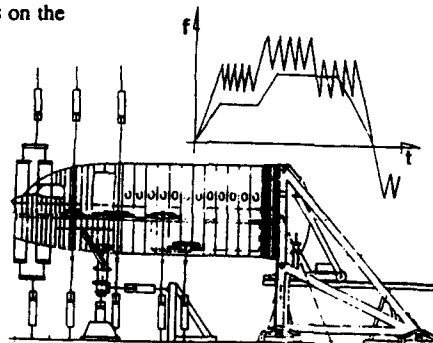
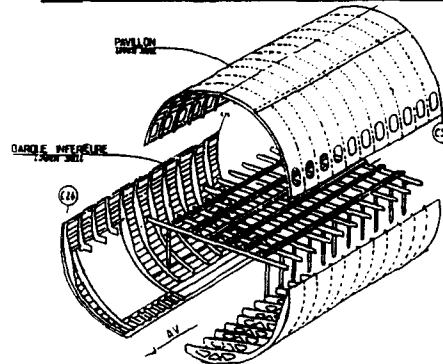
6 - ANNEXES

TEST SET-UP

SECTION 11/12 STRUCTURAL ASSEMBLY



SECTION 13/14 STRUCTURAL ASSEMBLY



PROPOSAL FOR THE NEW FATIGUE MANAGEMENT SYSTEM FOR THE AMX

by

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1. SUMMARY

The structural monitoring systems produced by ALENIA for Italian Air Force are briefly presented and discussed. From the previous experience a new proposal for AMX aircraft monitoring arises and is showed in the present paper. The proposed system is based mainly on in flight recording of strain gauges measurements with, in addition, storage of some flight parameters for special investigations or simplified back-up analysis in case of failures of the main system. Elaboration methods to be used both on board and on ground are described. Measurements on static and fatigue tests on ground were studied for a correct tuning of the system.

ABBREVIATIONS LIST

D.A.U.	Data Acquisition Unit
E.F.H.	Equivalent Flight Hours (Damage x Fatigue Life)
F.M.S.	Fatigue Maintenance System
G.S.	Ground Station
L.S.I.	Load Severity Index (E.F.H. / FLOWN HOURS)
P.G.S.	Portable Ground Station
S.R.M.	Simplified Recovery Monitoring
U.D.M.	Unit Damage Matrix
W.A.F.T.	Whole Aircraft Fatigue Test
s.g.	strain gauges

2. BACKGROUND

For several years Italian Air Force decided to monitor the structural fatigue consumption of the fleet in order to optimize the management of the aircraft and to have a correct visibility of the fatigue status of the airframes. It was noted that a monitoring based on mean data of the entire fleet cannot be considered sufficiently useful because of the different utilizations of the single aircraft; consequently an individual fatigue monitoring was chosen as a correct method to obtain informations on the life of aircraft. Until about 10 years ago the aircraft monitoring were based on fatigue meter readings, practically simply comparing the service g spectrum with the design one; of course this method had two defects:

- it does not permit to take into account a lot of parameters that modify the loads on the structure (e.g. Mach, altitude, movable surfaces).
- it does not permit to monitor at all the items not g-related as fin and stabilizer.

So, also in consequence of the improvements in technology, at the beginning of the service life of Tornado aircraft a program to design a new monitoring system started. This system, that is now operating in Italy, is based on the recording of many flight parameters: the time histories of the parameters are stored on a tape cassette on board, at the end of the flight the cassette is removed and processed by a ground station where loads and then stresses are calculated from the recorded parameters, a "rain flow" analysis is performed and finally a fatigue life consumption is obtained by means of the usual relative Miner rule referred to the fatigue test results. The calculations are performed for 21 locations (10 right hand, 10 left hand, plus a "dummy" component) from whose it is possible to deduce the fatigue consumption of the entire structure.

The project was successful against the main design targets i.e.: individual tracking of each plane and fatigue evaluation of all the main parts of the aircraft, nevertheless the development of the system showed that many problems arised to convert parameters into loads because of the great amount of data to be stored and then processed, and the great complexity of the computer program that required also a powerful hardware in order to reduce the elaboration time. So for the new maintenance system for AMX aircraft it was proposed to improve the performances by the adoption of strain gauges that are now much more reliable than 15 years ago, and the use of solid state memories with on board calculation capabilities.

3. SYSTEM PHILOSOPHY

The proposed system is based on an integrated use of strain gauges and a limited numbers of flight parameters.

The "main" monitoring activity is demanded to the strain gauges readings to be elaborated on board by "rain flow" counting method in order to obtain the fatigue stress cycles; these will be converted by means of proper fatigue S/N curves into damages.

Moreover a limited number of flight parameters are recorded when a significant variation on Nz occurs; this will permit to perform a recovery monitoring evaluation if a failure on an unacceptable number of strain gauges lines occurs; in addition some special investigation (e.g. roll rate spectra) will be possible .

The data will be stored in a solid state memory on board, milked on ground and then transferred to a ground station for further elaborations and analysis in order to decide the proper maintenance actions.

4. MONITORED LOCATIONS

It is clear that the greater the number of monitored locations the better the knowledge of the fatigue situation of the entire structure; nevertheless the number of strain gauges to be fitted (and consequently the number of monitored locations) must match the availability of input channels in the on board electronic equipments, taking into account that the system will monitor engine and systems also.

The best compromise chosen and proposed foresees the installation of 12 strain gauges able to monitor the main parts of the aircraft (i.e. wing, front, central and rear fuselage, empennages).

The detail of the location is the following : (See also fig.1)

1	Wing rear spar near wing root	L.H.
2	Wing front spar near wing root	L.H.
3	Wing rear spar between I/B and O/B pylon	L.H.
4	Wing central fitting frame	L.H.
5	Fuselage upper longeron near cabin	
6	Fuselage upper longeron near front joint	
7	Fuselage upper longeron near rear joint	
8	Rear fin attachment	
9	Front fin attachment	
10	Stabilizer spigot	L.H.
11	Stabilizer spigot	R.H.
12	Wing rear spar near wing root	R.H.

Straingauges 1 to 3 permit to monitor the wing and indirectly the front and rear fitting frames.
The strain gauge 4 monitor the fuselage central fitting frame and indirectly the central part of the wing root.

Strain gauge 5 is useful to monitor the fatigue consumption due both to the front fuselage loads and cabin pressurization cycles because the longeron on which the strain gauge is fitted is sensible to the cabin pressurisation.

Strain gauges 6 and 7 monitor the front/central and central/rear fuselage connections respectively.

Strain gauges 8 to 11 monitor the empennages and contain an intrinsic fail safe concept because the monitoring remain possible even if a strain gauge line fails. Strain gauge 12 is useful to have an evaluation of the possible different fatigue consumption between left and right hand parts.

The locations above are chosen to obtain the best compromise between good accessibility during retrofit installation and closeness to the expected 'critical' points.

Moreover the recorded flight parameters (i.e. Nz, Mach, altitude, roll rate, mass, store configuration) will permit to monitor, if necessary, g loaded parts like wings and fuselage.

5. DATA PROCESSING

5.1 Data processing on board

Strain gauges:

The strain gauges will be read basically with a sampling rate of 16Hz; the possibility to increase the sampling rate on 1 or 2 strain gauges will be added in order to monitor possible buffet phenomena.

The s.g. signals will be processed by the on board computer to identify the turning points of the time histories and then processed with the 'rain flow' counting method. The number of occurrences of the identified cycles will be stored in 30x30 matrices with rows and columns identifying alternate stress and ratios respectively.

Since the 'rain flow' algorithm does not ensure to reduce to cycles the entire time history of flight, the remaining part of this (turning point only) will be stored in proper memories and will be added to the next stress sequence in the next flight.

At the end of the flight, the matrices calculated during the flight will be summed to others representing the entire life of the aircraft.

Both kind of matrices will be multiplied by same dimensions matrices containing the damages of each kind of cycle (U.D.M.) obtaining the damage cumulated in the last flight and in the entire life of the plane.

Fig.2 contains a schematic view of the Strain gauges process.

Parameters:

As anticipated in previous paragraph the parameters are used only to have some information about the flight and to be able to perform a simplified recovery monitoring (S.R.M.) in case of failure of the s.g. lines. The first utilization is not strictly a subject of this memo and must be discussed in detail with the customer.

About the S.R.M. the Nz will be used as guide parameter. It will be continuously recorded with the normal sampling rate of 16 Hz and processed in order to identify the turning points of its time history with a filter to avoid to take into account variations less than 0.5 g. When a turning point will be identified the Nz value will be memorized together with the values of other considered parameters present at that moment (Mach, roll rate ...); so at the end of the flight a list of instants with the corresponding values of parameters will be available.

Fig.3 contains a schematic view of the process.

Moreover some parameters are analyzed only to verify that certain flight envelopes were not exceeded; if this occurs the relevant data and time are stored in a dedicated memory location.

The mission time, (From power on to power off conditions on M.R. equipment), and the flight time (From take-off to landing), are also recorded.

5.2 Data processing on ground (see also fig. 4)

The data stored on board will be transferred by milking on ground in a portable equipment (P.G.S.). This P.G.S. will be able to check immediately if some flight envelope was exceeded in order to give the possibility to the ground crew to decide if the aircraft can return immediately to fly or some maintenance/verification action must be done (GO - NO GO check); if requested the P.G.S. can also show the residual life of the monitored component with minimum residual life and its fatigue consumption in the last flight. These data, expressed in Equivalent Flight Hours, are simply calculated multiplying the cumulated damage by the life of the item (obtained by W.A.F.T. or theoretical analysis). Then the P.G.S. can transfer the data to the fixed Ground Station. The G.S. will perform the following jobs:

- a) Check of data recorded on board to verify their validity
- b) Possible correction of the on board calculated damages of the flight, taking into account (if significant) the residual part of stress history not converted into cycles by "rain flow" algorithm. This computation is of course useful only for statistical analysis and to compare severity of different kinds of mission
- c) Storage of data in a proper file to track the entire life of each plane of the fleet
- d) Simplified Recovery Monitoring by means of recorded parameters and simple theoretical equations if necessary. Parameters and simple theoretical equations will be used also to help the interpolation of s.g. data when only some s.g. line fails.
- e) Display of results both in written and graphic form. Outputs in written form will be equal to outputs used for Tornado Maintenance Recorder (see fig.5 to 7).
- f) Statistic evaluation of fatigue consumption of the fleet and linkage with other maintenance systems of the Air Force to optimize maintenance activities.

6. CALIBRATION OF THE SYSTEM

From a schematic point of view 3 Kinds of calibrations/verifications are necessary in order to ensure a correct working of the system based on strain gauges:

- a) Calibration of the stress level read by the s.g. fitted in a certain position against the stress present in the "critical" points.
- b) Verification of what the s.g. are actually reading, that is different for each aircraft because of possible differences in the installation and in the individual characteristics of each s.g. and associated wire lines
- c) Verification of the status of the s.g. lines during the service time against possible modification of measurement characteristics (e.g. shift of the zero).

About point a) an "ad hoc" instrumentation was installed on W.A.F.T. in order to get stress levels both in points expected to be significant for monitoring and in points candidate to be fitted with s.g. on Service aircraft. The location to be monitored in service are 10 (2 s.g. are used as spare, or to monitor asymmetric behaviours) but on W.A.F.T. 50 dedicated s.g. were fitted, this should guaranty to have sufficient informations also if other monitored locations must be chosen. From these strain gauges a complete stress history during W.A.F.T. will be obtained too and these data will be used to tune the calculation of damage for service aircraft (Relative Miner rule).

In order to get the actual measurement characteristics of the system installed on each aircraft it is foreseen the construction and the use of a rig able to apply simple known loads to an aircraft after the installation of the monitoring system. Each aircraft will be loaded by this rig and all relevant characteristics measured and stored in the on board computer in order to have all the data referred to a standard (e.g. stresses measured during ground tests).

To check the status of s.g. during service, the "check of the zero" (actually check of reference point) will be performed automatically before each flight: when the monitoring system is switched on it identifies the configuration, measures stresses and assumes the corresponding values as reference.

In addition the possibility of "random" checks in flight will be implemented.

30 flight conditions (defined in terms of Nz, Mach, Altitude, mass etc.) will be stored together with the associated stresses in the monitored locations; when the aircraft will pass for one of these conditions (the verification is made with the monitored parameters) the stresses read by s.g. will be compared with the stored ones; if the difference exceeds a prefixed percentage, correlated with the uncertainties given by the limited number of parameters used to identify the flight condition, it means that something is wrong and consequently a warning is given and the supposed wrong measurements are stored in the computer memory for a detailed analysis on ground.

CONCLUSIONS

Starting from the assumption to monitor individually each aircraft of the fleet (with clear benefits for the management/utilization) the proposed F.M.S. supplies a system with several advantages on previous ones:

- Great accuracy in stresses and consequently in fatigue evaluation for the entire mission (from power on to power off)
- Relatively simple calculation algorithms
- Complete independance from store configuration
- Minimum requirements of ground operations
- Auto - test facilities
- Simple recovery system and sufficient redundancy in monitored points
- Possibility to take into account directly buffet phenomena
- Very small elaboration time on ground with main informations immediately available after the flight if requested
- Complete commonality of outputs with previous monitoring systems
- Possibility of link with other computerized maintenance systems of the Air Force.

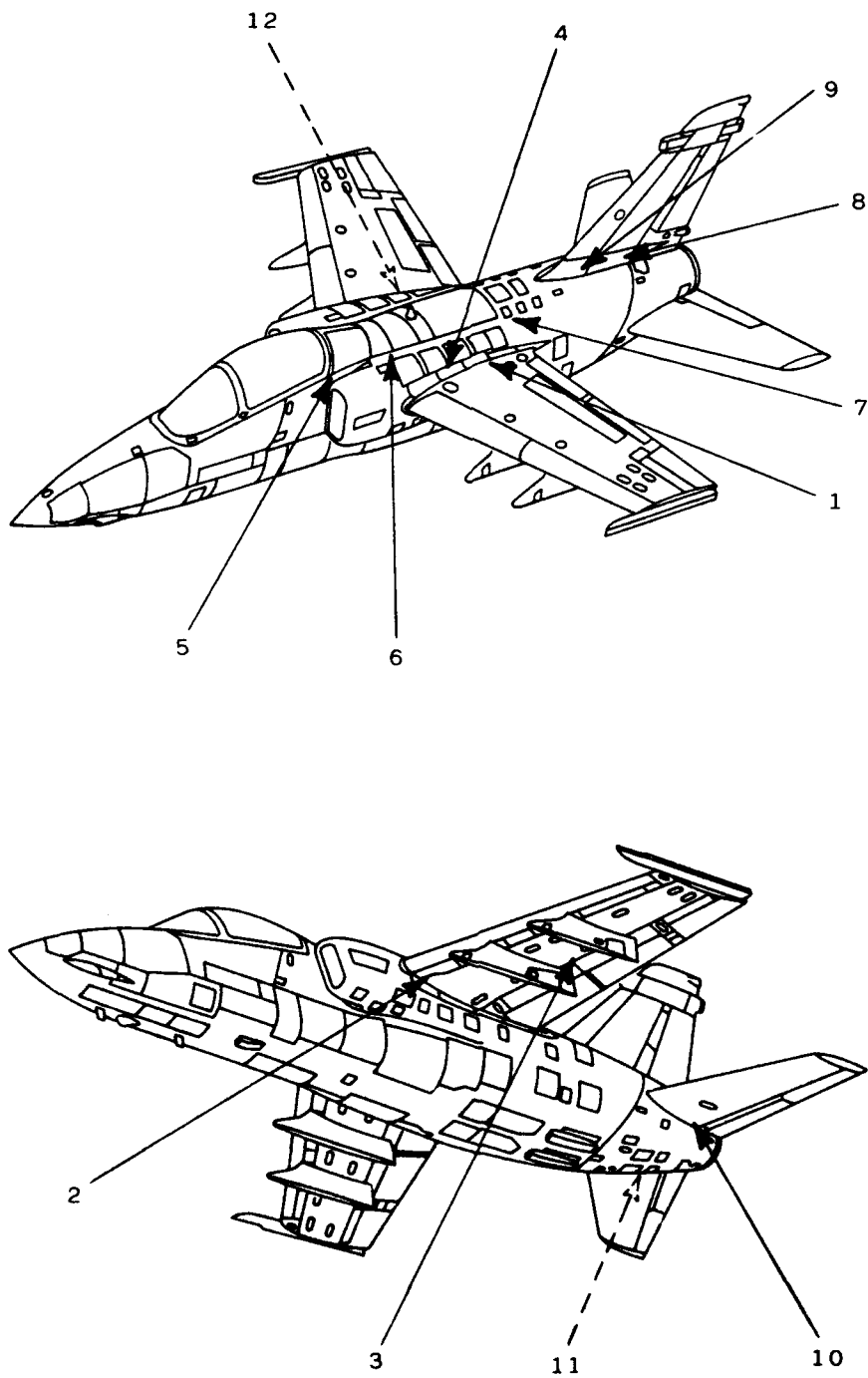


FIG 1: MONITORED LOCATIONS

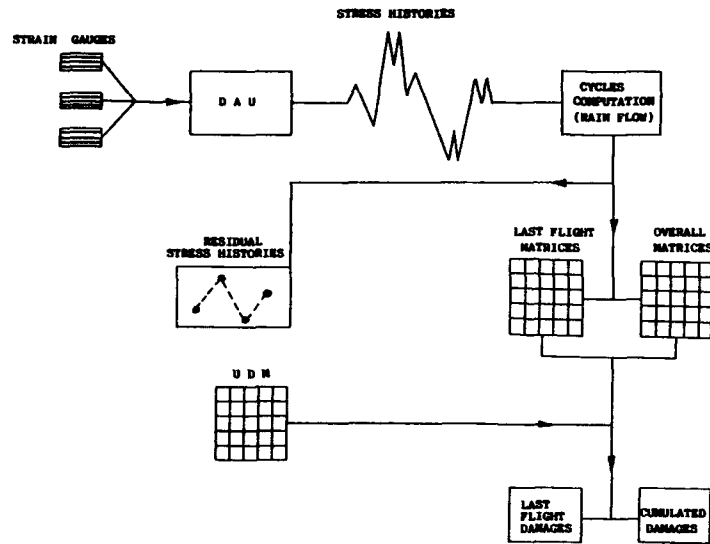


Fig.2: Strain Gauges Process

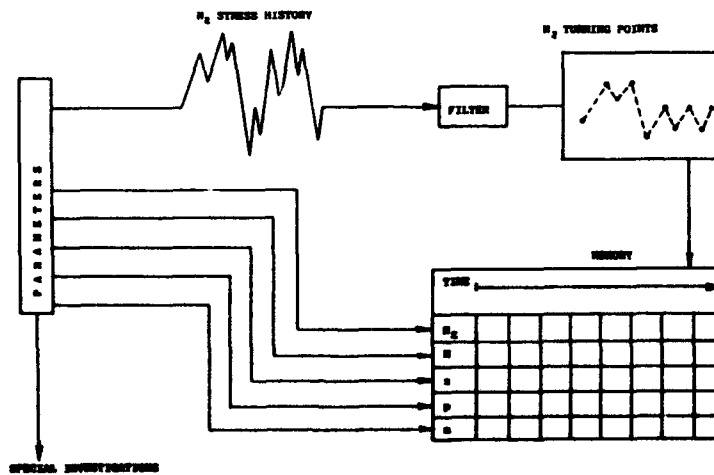


Fig.3: Parameters Process



FIG.5 : Output (Written Form)

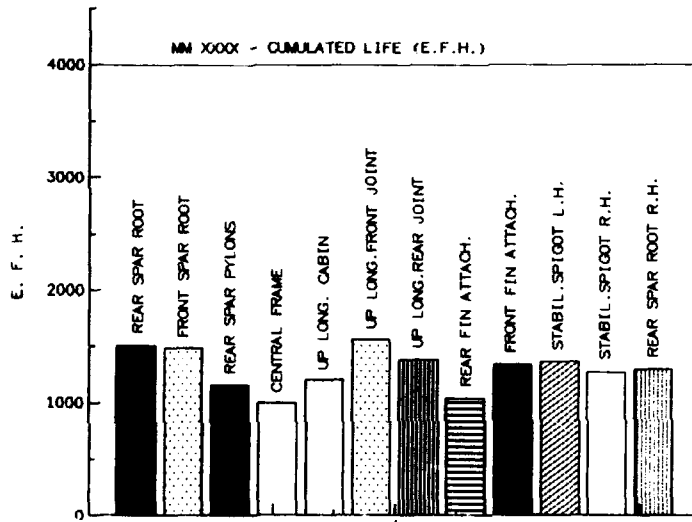


FIG.6 : Output (Cumulated Life)

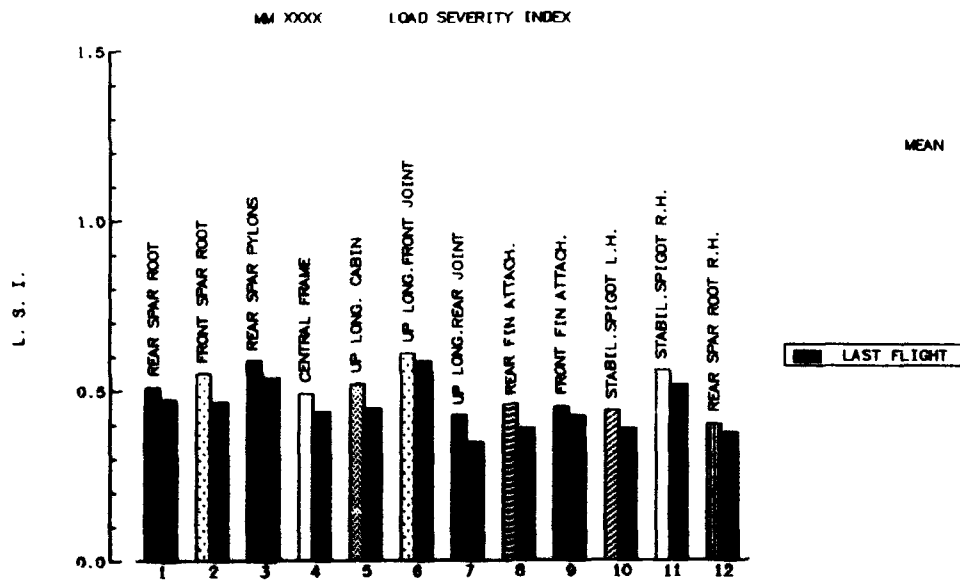


FIG.7 : Output (L.S.I.)

DURABILITY AND DAMAGE TOLERANCE TESTING AND FATIGUE LIFE MANAGEMENT: A CF-18 EXPERIENCE

by

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SUMMARY

The CF-18 multi-role fighter entered service in the Canadian Forces (CF) in 1982. Since then, 138 of these aircraft have been delivered and are operating out of seven squadron located in Canada and in Europe. Each aircraft is equipped with a multi-channel data acquisition system which records a number of parameters including measured strains, engine data, and various flight incident data. The analysis of this recorded data has enabled the CF to very closely monitor aircraft usage and the rate at which each individual aircraft accumulates fatigue damage. Early in 1986, in-service analysis of CF-18 fleet usage indicated that the aircraft were being operated in a significantly different manner than assumed for design, and that severity of the usage approached or exceeded the spectrum used for certification testing. Furthermore, the CF-18s were accumulating fatigue at a higher rate than expected which lead to concern about the potential for the airframe to reach its required life expectancy to the year 2003.

To ensure that the maximum economic life of the CF-18 fleet is achieved, the CF have implemented an aggressive Fatigue Life Management Programme (FLMP) and are proceeding with full scale Durability and Damage Tolerance Testing (DADTT) activities. The test is a joint Canadian and Australian venture usually referred to as the International Follow-On Structural Test Programme (IFOSTP). This paper describes the development of various components of the FLMP, including the aircrew fatigue awareness programme, CF-18 Individual Aircraft Tracking (IAT) activities, the development and implementation of fatigue damage control measures, and lessons learned from the management of the FLMP. The CF-18 DADTT philosophy, methodology and management approach adopted by Canada as well as the requirements that warranted the test will also be discussed.

1.0 INTRODUCTION

The CF-18 was designed to a United States Navy (USN) specification [1] that follows a safe life philosophy. The USN required the F/A-18 airframe service life of 6,000 Flight Hours (FH) to be demonstrated by successful completion, without major failure, of 12,000 Spectrum Flight Hours (SFH) of full scale fatigue testing to a "severe" usage spectrum. In general, the use of a severe spectrum (normally 2 or 3 standard deviations from the mean) and a scatter factor of two provides a confidence level on safe life similar to that of the USAF and RAF which are currently applying a scatter factor from 3 to 5 to full scale test result [2]. The USN establishes the safe life of a fracture critical component by defining a period during which a crack is assumed to initiate to a nominal size of 0.01 inches, and the remainder of the component life is expended in crack growth. The Canadian Forces (CF) have implemented a different airworthiness policy which follows a durability and damage tolerance philosophy. Safety of flight is ensured through the implementation of Non-Destructive Inspection (NDI) performed at appropriate intervals. However, the design philosophy of the CF-18 has made it difficult to apply the durability and damage tolerance philosophy to most fracture critical components on the CF-18 due to the inaccessibility of critical areas, small flaw sizes involved, and rapid crack growth from detection limits (typically of the order of 1,000 SFH for the Flight Station (FS) 488 wing carry through bulkhead [3]). Therefore, the CF intends to operate the CF-18 as a safe life aircraft, however, the safe life will be re-evaluated based on the results of further testing.

2.0 DADTT AND FLMP REQUIREMENTS

Early in 1986, in-service analysis of CF fleet usage indicated that the CF-18 aircraft were being operated in a significantly different manner than assumed for design, and that severity of the usage approached or exceeded the spectrum used for certification testing. Figure (1) shows normal acceleration exceedance curves for both CF-18 fighter and trainer aircraft compared to the test spectrum. The curves clearly indicate the increased severity of CF usage, with the average load exceedance frequency data exceeding the test spectrum

consistently up to the 6 to 7 G range. It was also noted from yearly fleet statistics, that the fatigue damage rate had been steadily increasing since the aircraft entered into service. Although the escalation in usage severity could be attributed to increasing aircrew experience in flying the aircraft, it is believed that the implementation of a G-limiter in later version of the Flight Control Computer (FCC) software in the mid 80's (which provided aircrew with a care free manoeuvring capability) was also a contributing factor. Table (1) provides a summary of the CF-18 fleet fatigue severity relative to the test spectrum since 1984 with its peak values reached in 1988.

Usage data also indicated that areas of the flight envelope where the aircraft was being operated were different from those assumed in testing. The manufacturer's full scale test was based on a three Point-In-The-Sky (PITS) distribution in which maximum aircraft loading on critical aircraft components was deemed to occur. Table (2) and (3) show the assumed distribution of time spent in the different PITS in comparison to the CF fleet usage. As indicated the differences are significant, the full scale test being unrepresentative of the CF distributions. These differences in PITS distribution may have a major impact on structural life predictions for an aircraft like the CF-18 which makes extensive use of digital flight controls. Mach, altitude, dynamic pressure and rate parameters strongly influence control surface scheduling, which in turn have a major impact on aircraft load distribution.

Furthermore, there are concerns over the structural configuration of the tested airframe. As certification testing progressed and failures were encountered, configuration changes were introduced on the production line or proposed as fleet retrofit to improve the fatigue characteristics of deficient components. A number of these structural improvements to fracture critical components were certified based on analysis or limited coupon testing only and never subjected to full scale testing to a representative CF spectrum. It is expected that these modifications will enhance the fatigue characteristics of the fleet aircraft, however life improvements cannot be accurately quantified without proper full scale testing.

As a result of these uncertainties the CF is currently applying a scatter factor of three to the manufacturer's full scale test result. Consequently the CF-18 safe life has been reduced to the equivalent of 4,000 hours. The reduced certified life and the increasing usage severity have raised concerns regarding the potential for the CF-18 aircraft to reach its required life expectancy to the year 2003. In order to resolve this serious fleet management problem, the CF implemented an aggressive Fatigue Life Management Programme (FLMP) to minimize fatigue damage accrual and decided in 1989 to proceed with follow-on full scale testing of the CF-18 airframe to establish its safe life under a representative spectrum. Full programme go-ahead was provided in February 1990. The following sections will discuss the planned full scale testing activities and the ongoing fatigue life management programme.

3.0 CF-18 DURABILITY AND DAMAGE TOLERANCE TESTING

3.1 Overview

The CF have established an airworthiness policy that is based on the implementation of an Aircraft Structural Integrity Program (ASIP) for each aircraft type in service. The Canadian ASIPs [4] are based principally on the USAF specification with some flexibility in the degree of application to each aircraft type to reflect the various fleet's pedigrees. The structural integrity management is based on the characterization of the Durability and Damage Tolerance (DADT) features of the aircraft. The aspect of durability relates to fatigue damage accumulation and crack initiation time. The damage tolerance approach is concerned with the performance of the structure when a flaw is present in the material. Consequently follow-on structural testing of the CF-18 will take the form of a DADT test. It is realized however that full application of the DADT principles may not be possible because of the safe life philosophy adopted during the design of the CF-18. The CF-18 DADTT is being conducted as a collaborative effort between Canada and Australia who share the same structural integrity concerns regarding a reduced certified life and high rate of fatigue accrual. This joint programme is usually referred to as the International Follow-On Structural Test Programme (IFOSTP). Within the programme Canada will be responsible for testing the centre fuselage and wing structures while Australia will be testing the aft fuselage and empennage structure. The aft fuselage fatigue damage is due to manoeuvre loads and dynamic loads induced by LEX vortices, while the centre fuselage and wing damage is mainly dominated by manoeuvre loads. The requirement to realistically simulate the empennage buffet loads environment leads to this division of responsibilities. The collaborative nature of the test programme offers major advantages in terms of test schedule, by enabling parallel testing; project cost, by offering cost sharing; and reduced technical risk, by the sharing of corporate experience gained in previous tests undertaken by both partners. In

Canada the test effort has been distributed between industry and government laboratories while overall project management and technical direction remains a Canadian Forces responsibility. The Institute for Aerospace Research of the National Research Council located in Ottawa has been assigned the responsibility for spectrum development tasks, conduct of the wing DADTT, and wind tunnel testing activities in support of the loads derivation. Bombardier Inc, Military Aircraft Division located in Mirabel, Quebec is responsible for conduct of the centre fuselage testing and derivation of the test loading. Figure (2) shows the Canadian project organization chart and assigned responsibilities.

3.2 Test Objectives

The objectives of the F/A-18 IFOSTP as determined by the project partners are to:

- determine the safe life of fracture critical structures;
- determine the economic life of the aircraft structure;
- where possible, obtain crack growth data to support application of a DADT approach;
- validate existing repairs and structural modifications and determine future modification requirements; and
- complete sufficient testing by 1994, to facilitate preliminary life cycle management decisions.

The first four test objectives are relevant to the engineering features of the full scale test while the last objective has been established in response to fleet management concerns. For the Canadian fleet, 1994 corresponds to the date at which several of CF aircraft will be reaching their safe life of 4,000 hours. It is therefore imperative for fleet management that this milestone be achieved in order to avoid having aircraft restrained from flying while awaiting test results.

3.3 Test Spectrum

The load spectra to be applied to the each test must be representative of both CF and RAAF in-service usage [5]. The test spectrum should simulate an average usage in preference to a severe one. Such test conditions have the advantages of leading to representative failure sites and crack growth rates. Fortunately, in-service operation of the F/A-18 is quite similar in both countries and it is expected that the final test spectra will be highly representative of in-service usage, enabling accurate test interpretation. Figure (3) compares N_1 exceedance curves for both RAAF and CF fighter aircraft.

For any aircraft type, the task of developing a test spectrum is a complex undertaking that requires indepth knowledge of past in-service operation, careful consideration of future roles, and a good insight on the projected maintenance plan. For the F/A-18 this task is further complicated by the fact that dynamic loading plays a major role in the aft fuselage fatigue damage, and that both fleets have been subjected to large variations in usage severity since their entry in service. For the aft fuselage the problem is further aggravated by the implementation of the LEX fence modification which has significantly reduced the magnitude of dynamic loading. As a result of the large amount of damage accumulated prior to LEX fence incorporation it is likely that the aft fuselage spectra will be composed of two different load sequences: one representing pre-LEX loading and a second representing post-LEX loading. For the centre fuselage case the test spectrum will be composed of an appropriate mix of early, current and possible future usage. The wing test will follow a similar philosophy and may include some form of dynamic loading simulating for the outer wing structure. Test spectra will be developed to achieve the maximum coverage to the fleet.

Another problem related to the development of the CF-18 spectrum is the complexity of calculating external balance loads. The Flight Control Computer (FCC) algorithms and control surface scheduling results in the load distribution over the entire aircraft being strongly influenced by rate parameters, Mach, and altitude variables which makes traditional methods of load scaling highly inaccurate.

To alleviate this problem, the determination of the aircraft load distributions will rely heavily on flight test. In addition an engineering development programme has been initiated in Canada as part of the IFOSTP to develop a F/A-18 analytical load derivation capability. All the load models developed will be validated using existing flight test loads and aerodynamic pressure distributions validated using wind tunnel measurements. The load modelling uses an aeroelastic approach [6] and includes proper consideration of the control surface deflection as programmed in the FCC (aerocervoelasticity).

Early spectrum development activities on many aircraft fleets have typically been hindered by the lack of a comprehensive fleet usage data base. Usage analysis relied heavily on pilot interviews, mission profile analysis, and data processing of limited accelerometer (mainly Nz) data. For the CF-18, a very large data base has been generated using flight data collated by the Maintenance Signal Data Recording System (MSDRS). Approximately 80% of all flight data has been captured since the aircraft entered service. For IPOSTP the problem has become one of data management and processing, requiring careful selection of representative samples of in-service data in order to make usage characterization possible within engineering resources, and cost and schedule constraints.

3.4 Test Philosophy

Full scale fatigue tests are usually cycled until either predefined goals for certification are achieved, or the airframe economic life is reached. Previous F/A-18 testing was conducted with the former approach in that the aircraft were certified to a specified design lifetime without attempting to define an economic life limit for the airframe. Moreover, the transition from the USN fatigue and fracture criteria to a damage tolerance approach may prove to be difficult according to a study conducted by the manufacturer for the CF-18 ASIP [7]. A modified damage tolerance procedure will therefore be adopted for the CF-18 DADTT to determine the economic life. The following aims will be pursued:

- a. identify and confirm crack initiation sites. This can only be achieved if the duration of the durability test is such that natural cracking occurs at critical locations. Data regarding the site, sequence, and extent of cracking is invaluable and is as important as component to the data obtained from deliberately induced flaws;
- b. confirm crack growth predictions to define in-service inspection interval and methods;
- c. determine the modifications and repairs that will be necessary and evaluate their cost effectiveness. Airworthy repairs will be applied to the test article to the maximum extent possible; and
- d. the previous safe life limit of the airframe will be extended to determine its economic life. Test life time will be based on considering potential service life extension up to 8,000 hours and estimated life expectancy to the year 2007.

To meet these objectives for the CF-18, defects must be found while they are very small and corrective action must be underway before the residual strength of affected components become critical. To satisfy this requirement it is intended to make extensive use of on-line damage monitoring devices. These devices will make use of Acoustic Emission Monitoring (AEM) principles to detect crack initiation, or for known critical areas alternating Current Potential Drop (ACPD) to monitor crack growth. CF experience using AEM during the CF-116 Full Scale DADTT has demonstrated that this approach is highly successful in detecting and monitoring small flaws over a large area of structure [8]. In addition, research programmes with Canadian Industry and Universities are underway to develop new crack detection/growth sensors based on ACPD technique and strain sensing devices using fibre optic sensors (Figure 2).

It is currently planned to carry out durability testing for three lifetimes to 24,000 SFH under a representative spectrum followed by damage tolerance testing to determine the maximum economic life. However, a change in the current scatter factor of three may be necessary to determine a safe life of the structure because it is likely that the test spectrum will be representative of average usage. The CF considers that the approach presented above would provide an optimum balance between test cost, duration and returns.

3.5 Test Specimen

An overview of the F/A-18 production lot for both RAAF and CF fleet effectivity is presented at Figure (4). The most desirable source for a test article has been determined to be a structural configuration representative of a Lot VIII aircraft. The CF intends to retire a service aircraft to be used for the center fuselage test article. A suitable Lot VIII airframe has been identified by considering suitability to the required configuration, in-service usage, and completeness of the MSDRS strain history. The test airframe selected has accumulated approximately 680 flying hours and has been subjected to a service load history which is representative of the average for CF-18 fighter aircraft. The complete aircraft will be used for test purposes with the exception of the outer wing which will be replaced by dummy loading structure. Figure (5) shows the center fuselage test section. The wing test will be conducted as a stand alone test to enable concurrent testing with the centre fuselage. At this time it is not known if the wing testing will be conducted

using a wing root reaction fixture or a complete centre fuselage to act as transition structure. A wing with no previous service history has been identified for wing testing purposes.

The collaborative nature of this test has necessitated the establishment of joint repair policy with RAAF with respect to the treatment and disposition of all real or simulated structural damage in the test article. The joint repair policy and will address:

- the design and implementation of simulated damage;
- the design, manufacture and installation of appropriate airworthy repairs;
- implementation of fleet repair or fatigue improvement retrofits; and
- the removal of repairs for damage tolerance and residual strength testing.

3.6 Test Schedule

The logistic issues and future plans for a mid-life update programme has require that sufficient testing be completed on the test sections (centre fuselage, wing and aft fuselage) by late in 1994. Previous experience with full scale tests has shown that schedule requirements are difficult to achieve because of the problems in estimating the repair requirement. However, for planning purposes a ratio of 25% to 75% has been estimated to evaluate the percentage of test running and down time. At this time it is anticipated that active testing will be carried out using three work shifts five days a week.

4.0 THE CF-18 FATIGUE LIFE MANAGEMENT PROGRAMME

4.1 Background

Historically, it has been difficult to accurately determine the amount of fatigue damage an individual airframe has accumulated because of the lack of a loads monitoring system on fleet aircraft. Typically, a life limit for a given fleet of aircraft would be established based on a single fatigue test which was usually carried out during initial production, or sometimes a follow-on post production test. This life limit was often expressed in flight hours and did not account for differences in how each individual aircraft had been utilized.

The monitoring of individual aircraft loads history and the subsequent processing of this data to determine the amount of fatigue damage accumulated allows the fleet manager to track usage and fatigue damage of every aircraft in his fleet. The ultimate goal of a Fatigue Life Management Programme (FLMP) is to control aircraft usage and fatigue damage accumulation such that the economic life of the fleet is maximized while maintaining operational effectiveness. The fatigue management of an aircraft fleet is a complex process dependent upon many parameters such as the availability of accurate aircraft fatigue damage and usage data, understanding of the mission profiles and mission requirements, aircrew awareness and understanding of fatigue related problems, and the ability and desire of the operational community to implement fatigue damage control measures.

The fatigue life management process can be grouped into three basic activities; fatigue awareness, usage characterization, and guidance and control. Fatigue awareness is achieved through briefings, videos, training, and various fleet management documents. Usage characterization is achieved through the gathering and processing of aircraft usage data in a timely and accurate manner. Through the use of the processed fatigue data, fleet managers are able to monitor and control the future usage of the fleet to ensure the life of each aircraft is maximized.

The processing of the CF-18 in-flight recorded data started when the aircraft entered service in 1982. To address the problem of increasing fatigue damage accrual and to ensure that the maximum life of the entire fleet was achieved, the CF began addressing fatigue management issues in 1988 and formally introduced a Fatigue Life Management Programme in early 1989. The CF-18 FLMP allows aircraft managers at both fleet and squadron levels to better manage CF-18 usage.

4.2 Elements of Fatigue Management

The three elements of the CF-18 FLMP are fatigue awareness, usage characterization and guidance and control. All these elements are essential to a successful programme and each are dependent upon one another. The relationship between these elements is shown in Figure (6).

4.2.1 Fatigue Awareness

The objective is to provide operators, and to a lesser extent maintenance personnel, with a basic understanding of the causes and consequences of fatigue. Beginning in early 1988 the CF-18 operational community at all levels were given briefings on CF-18 fatigue related problems. To provide a more structured approach, fatigue awareness training is provided during primary and advanced flight training. A video entitled "Fatigue - What's the Difference" was produced and is shown regularly at each of the squadrons. Similarly various documents and articles addressing, in simple terms, fatigue related issues have been distributed to the operational community. Fatigue awareness is very important element to a successful FLMP because it is the aircraft operator who ultimately has direct control over the amount of fatigue damage that occurs during a given flight.

4.2.2 Usage Characterization

Usage characterization is the gathering, processing, interpretation and analysis of the in-flight recorded data. This element is fundamental to the FLMP because accurate and current aircraft usage data must be made available to the fleet and squadron managers. Furthermore, these data must be presented in a condensed, simple, easily understood (e.g. "user friendly") format or the data will not be fully utilized or may even be ignored. The gathering and processing of the recorded data forms part of the Aircraft Structural Integrity Program (ASIP) which defines the management of the aircraft's structural integrity using Individual Aircraft Tracking (IAT) activities for accurate fleet fatigue monitoring. Every CF-18 is equipped with a sophisticated usage tracking system called the Maintenance Signal Data Recording System (MSDRS). Figure (7) shows the various components of the tracking system and the types of data that are recorded. The data collected is processed through a computer code called the CF-18 Structural Life Monitoring Program (SLMP) located at the CF-18 System Engineering Support Contractor (SESC), Canadair.

In addition to a number of flight and engine parameters, the MSDRS monitors, at 10 Hz frequency, the sequential occurrences of the strain sensor readings and records the strain peaks and valleys for the seven locations shown in Figure(8). The strain sensors are specific to a material and are installed by the manufacturer during production. The locations of the sensors were selected by the manufacturer based on the criticality of the structure, its accessibility and the degree of protection from accidental damage. There is a primary and a backup sensor at each location although only one sensor is connected. Should the active sensor fail, the alternate one must be manually connected.

The direct measurement of strain allows an accurate determination of the stress at a given location. Hence, the fatigue calculation can be based on the actual stresses experienced during flight. This methodology is discussed in detail in the subsequent paragraphs. Use of direct strain measurement inherently accounts for airspeed, altitude, weight, stores configuration and c of g variation within a flight. However, the accuracy of the fatigue calculation is dependent upon the reliability and proper installation of the strain sensor system.

4.2.3 Guidance and Control

The objective of this FLMP element is to control the fatigue damage accumulation through the use of various measures, based on the information provided by the SLMP. For this essential element to be successful, it must be managed by the operational community. Damage control measures are discussed in detail in the subsequent paragraphs.

4.3 Data Handling and Report Generation

Every CF-18 is equipped with a cassette recorder where all the data collated by the MSDRS via the mission computer are stored. When the cassette is full, or when the data is required, it is removed from the aircraft and stripped at an Integrated Ground Data Station (IGDS). An Aircraft Data File (ADF) is created which contains all of the cassette data plus an information header. Figure 9 gives a graphic representation of the data flow from aircraft to base.

A limited fatigue analysis code and report generation capability, which is called the Fatigue Life Information and Evaluation System (FLIES), has been installed at each of the CF-18 Main Operating Bases (MOB). The FLIES can provide, on demand, relative fatigue damage severity information on individual squadron aircraft using a menu driven software package. The system enables the base to carry out a limited fatigue damage calculation which can be used for assessing the severity of individual flights. A statistical average for each

mission and for each squadron has been determined (and is constantly updated) and serves as a measure of the relative severity of a given flight or mission profile. The FLIES provides current processed fatigue information as rapidly as possible making it a valuable fatigue management tool at the operating base.

The Aircraft Data Files (ADF) are sent from each of the CF-18 MOBs to the System Engineering Support Contractor (SESC) for archiving and subsequent processing through the Structural Life Monitoring Programme (SLMP). Various fatigue reports are generated from the SLMP, including:

- a. Monthly Report: This operational level report provides cumulative fatigue damage information on individual squadron aircraft. The information includes mission severity figures, individual aircraft fatigue damage, strain sensor status tables and aircraft fatigue damage trends in terms of the monthly and cumulative historical damage rates. Most of the information is provided in a graphical format to simplify the interpretation;
- b. Quarterly Report: This command level report provides cumulative fatigue damage information on individual aircraft squadron, and fleet trends. In addition, the report provides VGH data, N_f exceedance curves, angle of attack (AOA) data and average weight statistics;
- c. Annual Report: This is an engineering level report which provides very comprehensive fatigue data, VGH, weight, N_f exceedances and AOA data. This report is reviewed by the engineering staff to ensure there are no unusual trends that may not be obvious on the more limited monthly and quarterly reports; and
- d. Special Reports: Canadair also produces specialized reports for specific requests and is capable of accessing any of the historical MSDRS data as required. For example, during our involvement in the Middle East each aircraft involved in the operations were independently tracked to monitor the severity of usage during the deployment period.

Figure (10) provides a graphic representation of the data handling process at SESC.

4.4 Data Processing Methodology - CF-18 SESC, Canadair

The SLMP is a complex computer program which uses MSDRS data to produce the various fatigue reports previously discussed. The SLMP expresses damage accumulation in terms of fatigue life expended (FLE) and the FLE rate, which is the first derivative of the FLE curve expressed in FLE per 1000 flying hours. The FLE represents the amount of fatigue damage an individual airframe has accumulated as a fraction of the total life of 6,000 flying hours representative of the design usage spectrum. This linear relationship was established using the information collected during the F/A-18 full scale fatigue test conducted by the manufacturer. However, because of differences between the design test spectrum and CF in-service usage, and differences between the test article configuration and CF-18 fleet configuration, the CF currently applies a scatter factor of 3 to the manufacturer test which results in a safe life corresponding to a FLE of 0.667 at aircraft retirement.

Reference [9] provides a detailed explanation of the methodology for the calculation of the FLE values. However, it is useful to briefly discuss the general principles involved to provide some insight into the significance of the FLE. For the purpose of the fatigue calculations, crack initiation was defined as formation of a crack of 0.01 inches. These cracks usually originate at locations of tensile stress concentrations where the material yield strength is exceeded when high load magnitudes are encountered in-service. From the in-flight MSDRS recorded strain peaks and valleys, a representative loading spectrum is generated, and by using the material stress-strain relationship, the corresponding stress spectrum is obtained.

To increase the accuracy of the damage calculations, there must be some method of accounting for local plastic deformation which will occur around areas of high stress concentration such as bolt or rivet holes. These areas are considered material notches for the purpose of analysis. At the root of the notches, where the material has deformed plastically because of some overload conditions, there is a large volume of unyielded material surrounding the yielded area. During unloading, this large volume of unyielded material induces residual compression stresses which retards the crack growth. However, these residual compressive stresses can be reduced, eliminated or even form a tensile residual stress if reversed loading takes place and causes local yielding in compression. To obtain the notch stress spectrum from the stress spectrum

Neuber's rule is applied. Figure (11) shows the concept of the determination of the crack initiation life once the notch stress spectrum is obtained. The hysteresis curve, which is unique to each material is used to find the equivalent strain. The use of the curve accounts for how an aircraft has flown in the past with respect to the application of high stresses which may have lead to local yielding. The equivalent strains are used to obtain the amount of damage per cycle. The linear sum of this damage per cycle (application of Miner's rule) gives the crack initiation life. The FLE is expressed as the total damage accumulation to date divided by the total structural fatigue damage required to initiate a 0.01 inch crack under the design loading spectrum.

After initiation, the remaining life of a crack is used up through crack growth. It is possible to analytically predict the crack growth phase by use of one of a number of crack growth models. Currently the SLMP does not calculate the crack growth based on the MSDRS data, however it is intended to incorporate a crack growth prediction model into the SLMP at some point in the future. Improvements to the FLE calculations are also expected to occur concurrent to the full scale test activities.

4.5 Fatigue Damage Control Measures

The CF-18 FLMP has received excellent support from the operational community. It was recognized by senior Air Force officers that unless some form of guidance and control was implemented, the CF-18 would not achieve its intended service life. Beginning in March 1989, formal guidance and control measures were put in place. Typical measures include a reduction of mission severity, control of the aircraft configuration, rotation of aircraft between squadrons, optimization of mission profiles to limit the fatigue damage incurred and various other measures. Some measures which have been implemented include:

- a. Mission Severity Reduction: Figure (12) shows the relative fatigue severity of specific missions within a squadron. Steps are taken to reduce the severity of certain mission by reducing high "G" manoeuvres which where not required by the nature of the operation. This figure is also used at squadron level to facilitate the aircraft utilization scheduling such that aircraft with high accumulated damage are used for less damaging missions;
- b. Aircraft Configuration Control: Figure (13) shows the effect that aircraft configuration has on the amount of fatigue damage that is being accumulated. It has highlighted the need to carefully manage the aircraft configuration during squadron training. For example, the CF-18 are no longer permitted to fly with a centreline fuel tank unless specifically required by operations. As indicated, this configuration is the most damaging and was used constantly prior to the implementation of the SLMP; and
- c. Individual Aircraft Fatigue Damage Control: Figure (14) shows the status of each individual squadron aircraft in terms of its wing root FLE and its current flying hours. The lines on the figure define a Fatigue Usage Management Envelope where aircraft should be contained in the middle "yellow" zone. Each aircraft is represented by a black square and the small lines give the FLE rate at which that aircraft flew during the previous month. This figure is updated monthly, and allows each squadron to manipulate individual aircraft by using them in either high or low severity missions depending on their cumulative FLE and current FLE rate.

The reduction of fatigue damage accumulation is a very high priority issue within the CF. Progress briefings are held periodically at very senior levels. The Vice Chief of Defence Staff (VCDS), the Deputy Chief of Defence Staff (DCDS), Commander Air Command and Commander Fighter Group are all briefed on a regular basis by the Air Operations staff (assisted by the Air Engineering staff). Each Squadron Commander is held responsible (and accountable) for the performance of his squadron in terms of fatigue damage accumulation. There are junior officers at each squadron appointed as FLMP representatives in case there are questions or problems.

4.6 CF-18 FLMP CURRENT STATUS

The CF-18 FLMP was introduced in January 1988 with the commencement of fatigue awareness briefings and information packages. Fatigue control measures were formally introduced in March 1989 to coincide with the upgraded fatigue tracking software (SLMP and FLIES). The fatigue damage rate for the fleet has been reduced by approximately 24% since the introduction of the fatigue damage control measures. Figure (15) shows the incremental FLE rate, which is an indication of the severity of flight for a specific period of time plotted against calendar date. The continuous downward trend can be seen. (There are,

as would be expected, some anomalies to the trend such as the rate for the second quarter of 1990. Detailed review of the data during that quarter revealed that the high FLE rate was primarily attributable to an annual air combat exercise held in April/May, and increased training in the operational training squadron. The third quarter data shows that the rate of fatigue damage accumulation is continuing to decrease.] Calendar fluctuation in the fatigue damage trends are inevitable but can be minimized by processing data on a regular basis (in the case of the CF, monthly and quarterly) and ensuring that the processed data is fed back to the operational community. Figure (16) gives the monthly data from a squadron involved in this annual April/May exercise and shows the effect that a single calendar event can have on fatigue damage accumulation.

Fatigue damage is the primary indicator for determining the life of an airplane. Hence, any reduction in the rate at which an airframe or fleet accumulates fatigue damage (which is the objective of a FLMP) will result in an extension in the life of that fleet. Figure (17) shows the percentage fleet size as a function of time (in years) and demonstrates the effect that damage trends have on fleet attrition. It shows that because of the implementation of the CF-18 FLMP, the life expectancy of the CF-18 fleet has been greatly increased. In fact, each airframe in the fleet has already gained two years of further operation. The resulting cost benefits are substantial and easily justify the expense involved in establishing the FLMP.

5.0 LESSONS LEARNED

5.1 CF-18 DADTT

From a fleet management viewpoint it is too early in the F/A-18 IFOSTP to derive any meaningful lessons learned at this time. Nevertheless, project planning and scheduling activities have highlighted the difficulties of project management, of joint international venture and the significant level of effort required for the coordination of the engineering activities between the project partners.

5.2 CF-18 FLMP

The CF-18 FLMP has evolved over the last several years as the users have become more familiar with the mechanisms of the programme. As with any complex, software based programme, there have been significant growing pains and several lessons learned. Many of the lessons learned are unique to the CF-18 FLMP and the fatigue tracking system it is based upon (the SLMP/MSDRS), however many are common to any fatigue management programme. The following paragraphs discuss some of the lessons learned through the implementation of the CF-18 FLMP.

a. Accuracy/Reliability of Strain Measurement

The use of measured strains in the fatigue damage calculation greatly improves accuracy because aircraft parameters such as weight, configuration, velocity, altitude and others are inherently taken into account. However, the accuracy of the strain measurement is very important. The strain sensors on the CF-18 were installed at time of manufacture were never calibrated because the manufacturer determined it was not required. It has recently been discovered that the wing root recorded strain measurement for a reference manoeuvre differs from aircraft to aircraft as a result of strain gauge drift. These differences in actual strain readings lead to inaccuracies in the fatigue damage calculation. The problem is being resolved by carrying-out in-flight calibration runs to provide a scaling factor for the measured strain. The historical data for all aircraft will be reprocessed to determine an accurate FLE. These are expensive and time consuming tasks, and highlight the requirement for ensuring the strain measuring device is accurate and calibrated on a regular basis. However, the calibration process can be reduced to the performance of a few reference manoeuvres and easily performed during maintenance test flights.

Furthermore, the reliability of the strain sensor must be high. Failure of a strain sensor introduces fill-in algorithms which give an estimate of the damage incurred during the period the strain sensor was not functioning. This damage fill-in is based on other recorded aircraft parameters and, in the case of the CF-18, is very conservative. Therefore, for an extended period of damage fill-in the FLE will be artificially inflated. This is very costly because an inflated FLE can mean an average aircraft will be removed from service prematurely. More accurate FLE fill-in algorithms are being devised as part of the SLMP accuracy improvement.

b. Air Operations Involvement

An effective FLMP must have support from the operational community at all levels. While the engineering community provides the fatigue damage information and strives to make it as accurate as possible, it is the operational community that must use this data to make the FLMP effective. In the case of the CF-18, the control and implementation of the FLMP rests with the Director of Air Operations and Training. They have initiated changes to standard aircraft configurations, to certain mission profiles and to the operational training syllabus in an effort to reduce the rate at which the squadrons and fleet aircraft accumulate damage.

c. Timely and Accurate Fatigue Damage Reporting

For a FLMP to function, accurate damage reports must be provided in a timely manner. It is of little use to provide accurate fatigue reports twelve months after the reporting period. To control the fatigue damage accumulation effectively, the fatigue reports for a given period must be produced and issued quickly to provide feedback on the effect of any fatigue reduction or of any role changes that may have occurred. The reporting period must be sufficiently frequent to identify calendar trends and allow the manipulation of individual aircraft if required to reduce the fatigue accumulation rate. In the case of the CF-18, monthly reports which detail individual aircraft and squadron trends are provided. They are distributed within 45 days of the end of the reporting period. Quarterly reports are provided which detail squadron and fleet trends. These reports are typically distributed within 60 days of the end of the reporting period. Finally, annual engineering reports which provide very detailed information on all aircraft are issued within 120 days of the end of the reporting period. To provide an even quicker feedback, a base level fatigue processing/reporting system has been installed at each base. This system is capable of providing relative fatigue damage trends for individual aircraft or squadron on demand. The data will typically be no more than 7 days old. This base level system provides very rapid feedback on the severity of individual missions. For example, the effect of a fatigue reduction measures, such a change in configuration, can be easily determined.

d. Fatigue Damage Data Presentation

For the fatigue damage reports to be of use, the data must be presented in a relatively simple manner. The CF-18 FLMP reports provide most data in a graphical format. It was found that squadron and fleet aircraft managers did not have enough time to sort through tables of data looking for trends or even individual aircraft fatigue damage information. The graphic presentation is extremely effective for highlighting problem aircraft or identifying squadrons that have not managed their fatigue damage accumulation properly.

e. Implementation of Fatigue Damage Control Measures

The implementation of formal fatigue damage control measures ensures that steps are taken to reduce the rate of fatigue damage accumulation. The CF-18 FLMP commenced with the introduction of the fatigue awareness element. Some success was initially achieved as personnel gained an appreciation for the causes and consequences of fatigue damage. However, it was found that as time progressed, the initial gains were lost and the severity of flying began to increase. This highlighted the need to have on-going fatigue awareness activities at all levels from primary flying training to senior management. It also demonstrated the requirement to introduce some formal fatigue reduction measures. The introduction of these formal fatigue control measures has resulted in a significant downward trend. When steady-state is achieved and if further reductions are still required, then it would be necessary to introduce more severe control measures such as budgeting the FLE consumption by squadron.

The CF-18 FLMP has been very effective in reducing the accumulation of fatigue damage in the fleet. The programme has been flexible enough to adapt to the various demands of the fleet and squadron aircraft managers for specific usage information. The feedback of data to the operational community has helped to ensure that steps are taken to properly manage the individual aircraft within a squadron.

6.0 CONCLUSIONS

The full-scale testing of the F/A-18 is a major programme in both Canada and Australia which utilizes extensive engineering and financial resources. However, this investment is more than justified in light of the returns offered by the continued airworthiness of the airframe and by the determination of its maximum economic life to support life extension activities.

The objective of the CF-18 FLMP is to ensure that the maximum life of the airframe is achieved by controlling the aircraft fatigue damage accumulation while maintaining operational effectiveness. To accomplish this objective the CF have implemented a programme which includes on-going fatigue awareness training, the gathering and processing of fatigue usage data, and the controlling of the fatigue accumulation rate for individual aircraft. The success of the programme is demonstrated by the 24 percent reduction in the FLE rate which has been achieved.

The success of the CF-18 FLMP can be, in part, attributed to:

- a. The participation and support of the CF-18 operational community at all levels. It is the responsibility of the engineers to provide accurate and current aircraft fatigue damage data, but final responsibility must rest with the operators;
- b. Individual Aircraft Tracking (IAT). The ability to track via the MSDRS the fatigue accumulation on every aircraft in the fleet has enabled the CF to manage the fatigue lives of each individual airframe;
- c. The Structural Life Monitoring Program (SLMP). This fatigue processing system provides accurate and timely reports. Reports are produced monthly, quarterly and annually and, because of the graphical format, are easily interpreted by all users; and
- d. The base level Fatigue Life Information and Evaluation System (FLIES). This fatigue data processing/reporting system provides very current (less than 7 days old) fatigue damage severity data on a flight-by-flight basis. This system provides a rapid feedback to squadron and Air Command on the effect of any changes to the aircraft usage or configuration.

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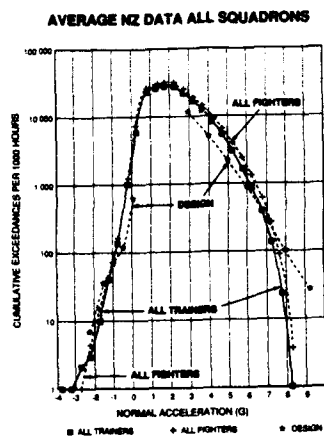


Figure 1 CF-18 Nz Exceedance Curves

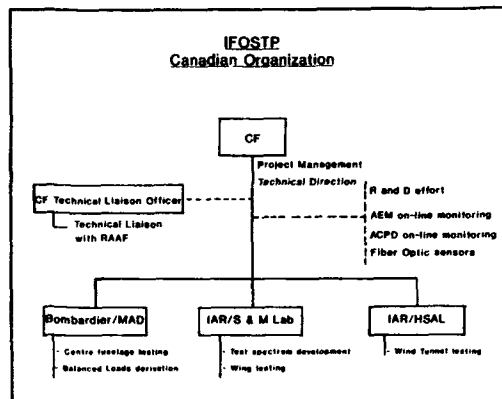


Figure 2 Organization Chart IFOSTP (CANADA)

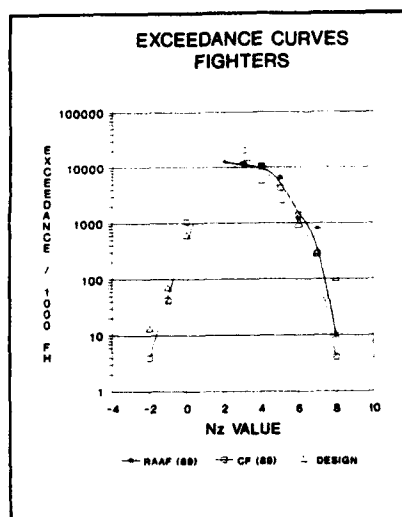


Figure 3 CF and RAAF Exceedance Curves F/A-18 Fighters

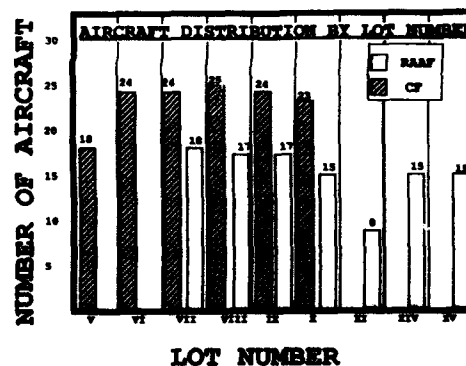


Figure 4 CF and RAAF Production Lot Effectivity

CF - TEST ARTICLE

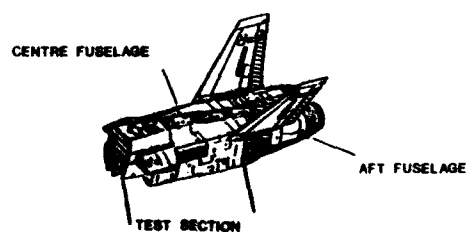


Figure 5 Centre Fuselage Test Section

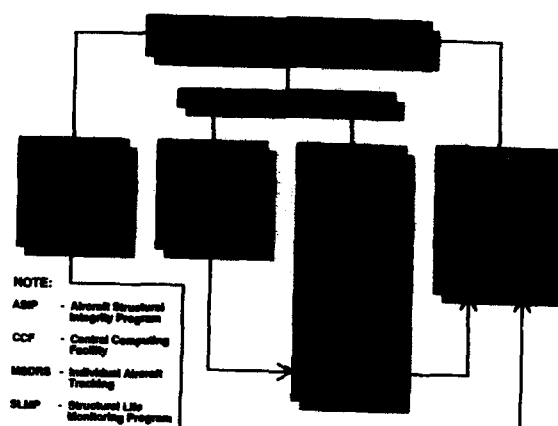


Figure 6 Elements of Fatigue Management

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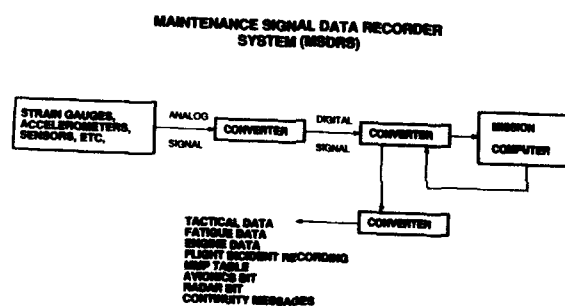


Figure 7 MSDRS System Components

STRAIN SENSOR LOCATIONS

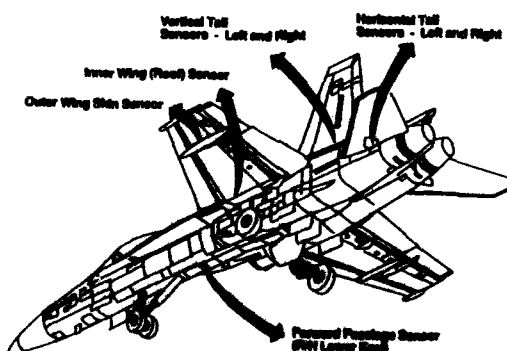


Figure 8 CF-18 Strain Sensor Locations

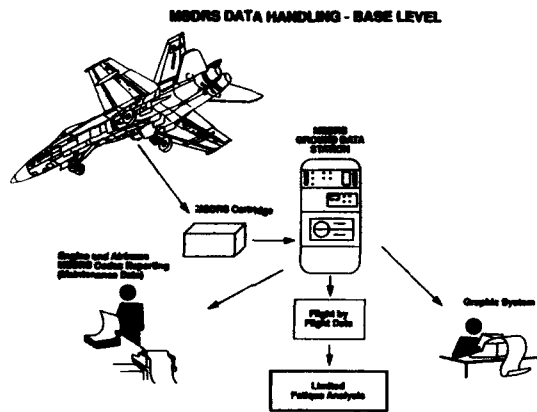


Figure 9 MSDRS Data Handling - Base Level

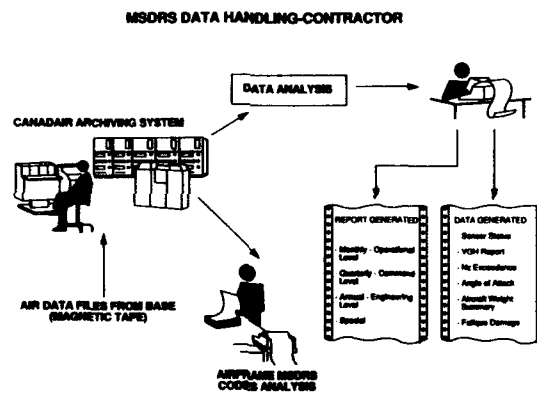


Figure 10 Data Reduction Process at SESC

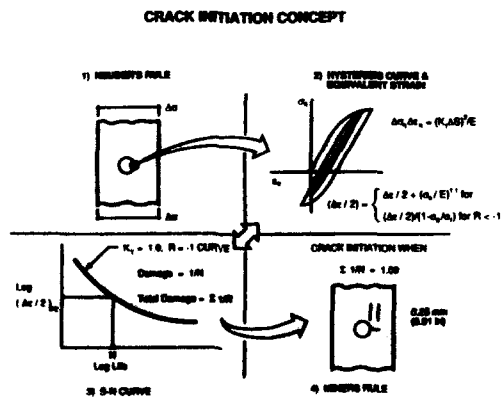


Figure 11 Crack Initiation Concept

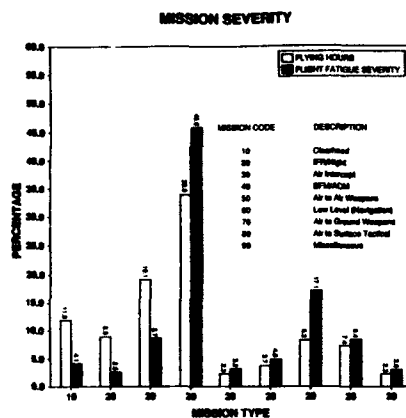


Figure 12 Relative Mission Severity

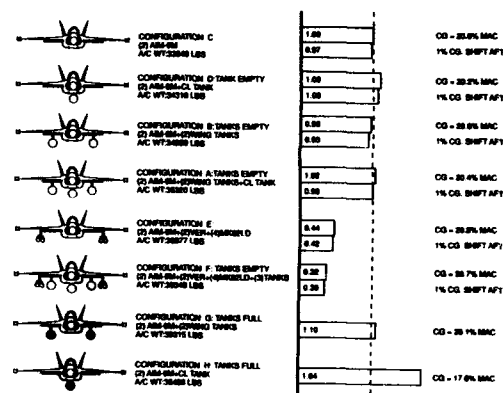


Figure 13 Configuration Effect on Relative Damage

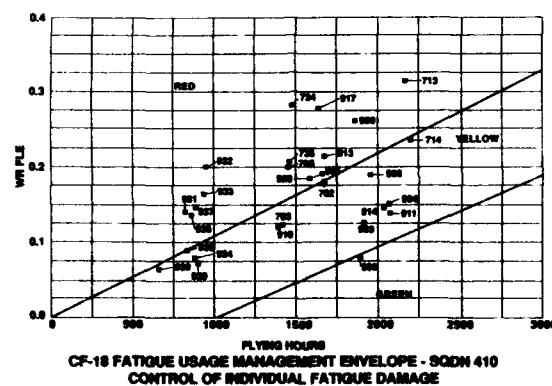


Figure 14 Control of Individual Fatigue Damage

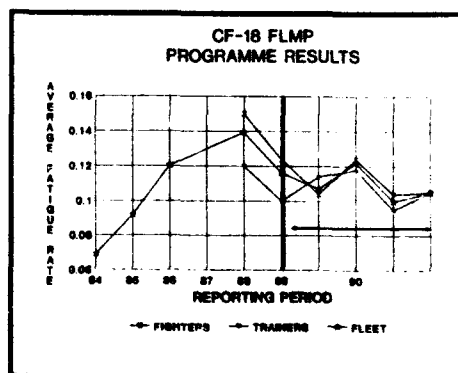


Figure 15 FLE Rate Trend

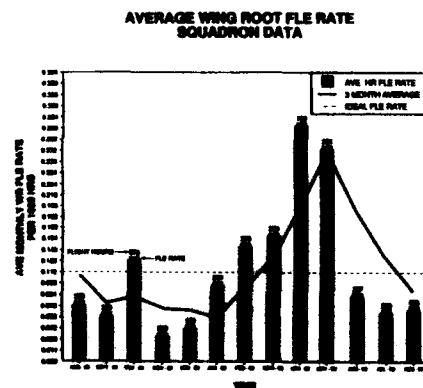


Figure 16 Monthly Squadron Data

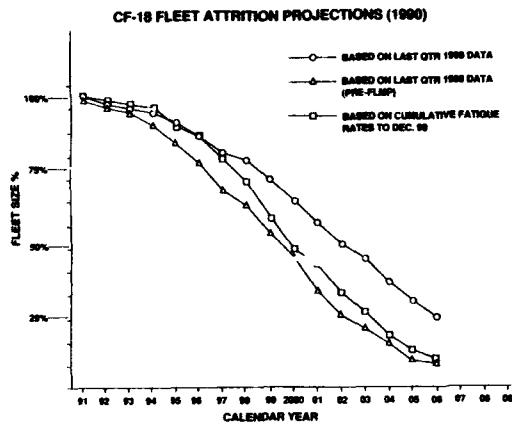


Figure 17 Fleet Attrition Trends

CF-18 USAGE SEVERITY

YEAR	AVERAGE FATIGUE RATE		
	FIGHTER	TRAINER	FLEET
1984	-	-	.069
1985	-	-	.092
1986	-	-	.120
1987	-	-	.130
1988	.150	.120	.139
1989	.124	.100	.116
1990	.105	.105	.105

Table 1 CF-18 Usage Severity to Test Spectrum

POINT IN THE SKY
CUMULATIVE OCCURRENCES ≥ 1.0 g

ALT (ft)	MACH RANGE					
	0-0.6	0.6-0.7	0.7-0.8	0.8-0.9	0.9-1.0	>1.0
0-10K	15.6 16.5 16.8	24.0 23.0 18.8	17.4 16.8 18.2	3.2 3.0 4.8	0.5 0.4 0.6	0 0 0
10K-20K	5.3 5.6 5.1	6.7 7.1 7.0	7.5 7.6 6.6	6.5 6.6 5.2	2.3 2.3 2.0	0.2 0.1 0.2
20K-30K	0.5 0.5 0.6	1.2 1.3 1.6	2.7 2.6 3.0	3.7 3.7 3.9	1.7 1.6 1.7	0.2 0.2 0.2
30K-50K	0 0 0	0 0 0	0 0.1 1.4	0.3 0.3 0.4	0.3 0.3 0.3	0 0 0

Note: Data based on following: 1989 = 112,761 Fatigue test points
cumulative flying hours: 1987 = 59,088 M.8 @ 5.K - 50%
1985 = 17,329 M.95 @ 15.K - 45%
VL @ 5.L. - 5%

Table 2 CF-18 PITS Distribution 3.0G Manoeuvres

POINT IN THE SKY
CUMULATIVE OCCURRENCES ≥ 7.5 g (DLL)

ALT (ft)	MACH RANGE					
	0-0.6	0.6-0.7	0.7-0.8	0.8-0.9	0.9-1.0	>1.0
0-10K	5.7 5.7 5.5 6.0 7.4 1.1	15.6 16.0 17.4 15.7 10.1 7.5	15.1 15.4 15.0 14.5 12.3 12.5	5.4 4.3 4.4 3.9 5.2 7.5	0.9 1.0 0.9 0.6 1.4 1.6	0 0 0 0 0.1 0
10K-20K	0.1 0.1 0.1 0.3 0	3.8 3.6 4.2 5.0 1.4	14.1 13.8 13.0 14.3 15.1 13.5	24.0 24.1 24.6 25.5 27.2 25.9	5.9 6.2 6.3 5.9 6.2 15.6	0.3 0.3 0.2 0.3 0.2 0.4
20K-30K	0 0 0 0 0	0 0 0 0 0	2.6 2.3 2.3 2.3 1.6 1.1	3.6 3.5 3.6 3.9 3.9 4.3	2.2 2.1 2.3 2.5 2.9 6.4	0.4 0.4 0.3 0.4 0.6 0.4
30K-50K				0.4 0 0 0		

Note: Data based on following: 1989 = 159,716 Fatigue test points
cumulative flying hours: 1988 = 125,375 M.8 @ 5.K - 50%
1987 = 83,943 M.95 @ 15.K - 45%
1986 = 53,223 VL @ 5.L. - 5%
1985 = 27,684
1984 = 10,355

Table 3 CF-18 PITS Distribution 7.5G Manoeuvres
References

THE G-222 AIRCRAFT INDIVIDUAL TRACKING PROGRAMME

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ABSTRACT

The G-222 is a transport aircraft designed about 20 years ago; from the fatigue point of view the design is of the conventional "Safe Life" kind, verified by means of a full scale test. The aircraft was equipped with a counting accelerometer, whose recordings have been used, together with the pilot-compiled forms, for the evaluation of life consumption.

In recent years, following the Damage Tolerance evaluation of the structure, a different approach has been developed for fatigue damage monitoring: the same input data are used for the prediction of crack growth. The main drawback of this approach is the lack of information about the sequence of load application, so that only non-interactive models can be properly used.

For this reason, AERITALIA decided to improve the quality of inflight recorded parameters in order to get more information about the actual usage, while also taking the sequence of load application into account.

The main lines of this activity, which is started now, are described and discussed.

1. INTRODUCTION

The G-222 is a medium transport aircraft designed by AERITALIA to meet the Italian Military Air Force (AMI) utilization requirements, in order to replace the C-119 and to integrate the C-130 in the AMI transport system.

Taking the expected usage, described in the AMI Technical Specification, into account, the fatigue strength of the G-222 airframe has been evaluated by theoretical analysis using the traditional Miner approach and by full scale fatigue testing. The low number of failures that took place during the full scale fatigue test, and their low relevance for the safety of the flight, prove the satisfactory fatigue strength of the G-222 structure.

To improve the Structural Maintenance Plan (SMP) and for possible future applications of this aircraft, AERITALIA decided to develop an Individual Aircraft Tracking (IAT) program, in order to verify the actual utilization and to update the SMP, if there is no agreement with the basic utilization. This activity is still in progress.

2. DESCRIPTION OF THE G-222

The G-222 is a transport aircraft, fig. 1, powered by two turboprop engines, optimized for medium payloads on medium ranges. It possesses a very high volume/payload ratio and is able to operate in a variety of roles; it is capable of operating from semi-prepared runways.

The first flight of the first prototype took place in July 1970. The second prototype made its first flight one year later. The results of the flight testing of the two airplanes contributed to the final assessment of the AMI Technical Specification, that was issued in 1973.

On December 1975 the first G-222 production model took off and began its flight test program; three other airplanes were extensively tested at the AMI Experimental Research Center to check systems reliability and to acquire operational and logistic experience. Special attention was paid to ground-flight tests to evaluate the loads due to taxi and take-off runs on semi-prepared runways. All tests were satisfactorily completed in the first half of 1978.

About 100 airplanes have been produced up to now. The fleet has now accumulated about 120000 flight hours (4000 FHs for the oldest aircraft).

Four typical missions were used in the design of the aircraft, fig. 2:

- Basic mission
Distance 1000 Km (2 flights, return without refuelling)
Cruise Altitude 300 m
Take Off Weight 22360 Kg
- Heavy mission
Distance 1800 Km
Cruise Altitude 4500 m
Take Off Weight 25560 Kg
- Max Payload mission
Distance 550 Km
Cruise Altitude 6000 m
Take Off Weight 26500 Kg
- Ferry mission
Distance 4500 Km
Cruise Altitude 7600 m
Take Off Weight 25100 Kg

The mixing of the fore-mentioned missions in the design spectrum is as follows:

- Basic Mission: 2022 missions (4044 flights) equivalent to 7000 FHs
- Heavy Mission: 1742 flights equivalent to 9000 FHs
- Max Payload Mission: 1915 flights equivalent to 3000 FHs
- Ferry Mission: 87 flights equivalent to 1000 FHs

Typical missions and versions are:

- Paratroop dropping mission (40 paratroopers)
- Troop transport mission (44 fully equipped troops)
- Materials transport mission (Max payload 9000 Kg)

- Materials airdropping mission (Max single load 5000 Kg)
- Fire-fighting version (Retardant max 6800 Kg)

About 50 G-222s were bought by AMI while the others were sold abroad.

3. FATIGUE DESIGN AND CERTIFICATION

The G-222 has been designed in accordance with the "Safe Life" design concept; the life goal was 20000 FHs, with a scatter factor of 2.5. Full scale testing was carried out to demonstrate a 50000 FHs fatigue life under AMI utilization, fig. 3. The test article was a complete airframe structure with the exception of:

- Landing gears
- Control surfaces
- Horizontal tail

which have been tested separately.

The fatigue load spectra for full scale fatigue testing were derived from the G-222 mission profiles, from ESDU 69023 (Vertical and Lateral Gust Spectrum) and from MIL Specifications: MIL-A-8866 (Manoeuvring Spectrum), MIL-A-8866 B (Taxi Spectrum and Landing Spectrum). The taxi spectrum was modified to take into account ground operations from semi-prepared runways. In order to reduce test time and costs, a truncation level and an omission level were applied to the load spectra:

- Truncation: the load levels occurring more rarely than 10 times in the life were not applied; this is "an arbitrary choice but it still seems to be a reasonable one", /1/.
- Omission: the load levels less than 22% of the mean load for vertical loads and less than 14% of the maximum alternate load for lateral loads were not applied. The level for the omission of vertical loads was the same as the TWIST sequence, /2/.

The four typical missions give rise to 13 different flight types when different gust loads are taken into account. As an example, fig. 4 shows the exceedance curves for the cruise segment of the 'Heavy' mission; in accordance with the TWIST philosophy, the same shape of the exceedance curves has been assumed for the different flights, which are characterized by different levels of severity. Flight A is the quietest, D is the most turbulence-disturbed. The continuous exceedance curve was discretized in five (or six in other cases) load steps. The taxi load spectrum is considerably less severe and therefore it has been decided to approximate it by means of only one step. Loads in each flight segment were organized in a Low-High-Low sequence; fig. 5 shows a typical flight. The 13 flight types were grouped in a block corresponding to 2000 FHs on the basis of the flight mixing previously described. This block, corresponding to 1/10 of the operative life of the aircraft, was applied 25 times in the full scale fatigue test, /3/.

No significant structural failures were observed during the inspections planned for the full scale fatigue test with the sole exception of the cargo door hooks, which required a small modification introduced in retrofit in all the aircraft.

In more recent years, following the effort made by many manufacturers concerning Dam-

age Tolerance verification of aircraft structures designed according to previous rules, AERITALIA decided to carry out such an evaluation for the G-222 aircraft. The main purpose of this activity was the qualification of the structure in order to guarantee higher safety margins against undetected or accidental structural damage; in some cases the Structural Maintenance Plan was reviewed and updated.

The activity, carried out using mainly analytical tools, demonstrated the high structural capability of the aircraft. The analytical Damage Tolerance verification is still in progress and experimental verification is planned to cover those areas in which this will be considered necessary due to narrow safety margins.

Fig 6 shows a typical result of Damage Tolerance assessment; the item is the link connecting the center wing section to the fuselage, which has been verified as a slow crack growth element.

4. SERVICE LOAD ASSESSMENT

As explained above, the G-222 was designed about 20 years ago and, in accordance with the monitoring philosophy of those years, it was equipped with a fatigue load meter, fig. 7. This instrument (counting accelerometer) is mainly used to analyze the utilization of the aircraft. The readings of the fatigue meter are collected in a "Flying Log and Fatigue Data Sheet", fig. 8, together with the main information regarding the flight such as:

- Weights of the aircraft (Total weight, fuel, stores carried, air-delivered stores)
- Number of landings
- Times (Duration of the flight, time below 1500 ft, time with flaps down)
- Average Height (Cruise, below 1500 ft)
- Presence of turbulence
- Flight type (Training, long range transport, short range transport, air delivery, etc.)

This information was used to evaluate the life consumption of the aircraft by a methodology based on the original Safe Life design of the aircraft (Miner rule). The accuracy of these evaluations is limited by the quality of the input data. The counting accelerometer can be used to evaluate the fatigue life consumption of those components of the aircraft whose loads are closely correlated to normal acceleration, mainly the wing and the fuselage. The G-222 is a rather stiff aircraft, so these evaluations can be carried out with sufficient accuracy by means of simple linear transfer functions. For those components in which life consumption does not depend on normal acceleration, the damage is evaluated on the basis of a "dummy" flight reconstructed by using the input data of the current flight.

Fig. 9 shows an example of fleet management based on a population of five airplanes; every 200 FHs the fatigue damage accumulated is updated and compared with the reference damage, which is evaluated in the full scale fatigue test sequence. Consequently, the aircraft are rotated in different roles in order to have a homogeneous life consumption.

Following Damage Tolerance verification,

the procedure for life consumption evaluation was improved: the information of the counting accelerometer and of the Flying Log is used to reconstitute the load sequence of each flight in the Structural Significant Items which are used as control points. Then crack propagation is evaluated for these Items by using a simple linear algorithm, as the counting accelerometer cannot give information on the load sequence.

The main drawback of this procedure is the low quality of the input data and the lack of information on load sequence. For these reasons, AERITALIA decided to improve the amount and the quality of the input data for the evaluation of life consumption. This objective will be reached by installing a digital flight recorder; log-forms and counting accelerometers will be maintained. The aim of this operation is the improvement of safety in aircraft usage and the reduction of maintenance costs.

Several parameters will be automatically recorded, such as:

- Height
- Indicated airspeed
- Vertical and lateral load factors
- Power setting
- Aircraft pitch angular speed and acceleration
- Elevators deflection
- Flaps deflection
- Dynamic pressure
- Mach number
- Fuel weight
- Incidence angle

while the following data will be manually recorded together with information on inspections, repairs, part replacement and component interchange which may affect the predicted structural damage status on a particular aircraft:

- Date of flight
- Aircraft Serial Number
- Base Code
- Accumulated flight hours
- Progressive mission number
- Mission code
- Take-off weight
- Stores carried
- Air-delivered stores
- Flight duration
- Number of landings

These data will be analyzed by a computer program to check their validity and consistency. For the valid set of parameters, time-history will be reconstituted and utilized for crack growth analysis according to an interactive model (generalized Willenborg). The results of this analysis will update the data bank used for the management and possible rotation of the airplanes of the fleet.

5. CONCLUSIONS

The availability of more modern and powerful instruments for flight data recordings makes it possible to manage a fleet by means of more complex and reliable methodologies.

The paper has shown, in the case of G-222 aircraft, how the Individual Aircraft Tracking methodology has evolved, starting from Miner rule calculations based on counting accelerometer readings, and going

on to linear crack growth analysis based on the same input data, and then to crack growth analysis which takes load sequence effects into account, based on a larger number of recorded parameters.

The development of this methodology, whose definition has not yet been concluded, has been considered necessary to improve safety in utilization of the aircraft, to reduce maintenance costs and to comply with modern airworthiness requirements, which put greater emphasis on the control of the crack growth process. Anyhow, the simple IAT methodology currently applied is justified by the fatigue soundness of the design, which has been confirmed by the results both of the full scale fatigue test and of subsequent damage tolerance analysis; beside, it must be borne in mind that the principal customer, AMI, utilizes the airplane in a quite homogeneous manner, fully consistent with the design spectrum.

6. REFERENCES

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- /2/ J.B. De Jonge, D. Schutz, H. Lowak, J. Schijve - A Standardized Load Sequence for Flight Simulation on Transport Aircraft Wing Structures- Amsterdam, NLR TR 73029, 1973.
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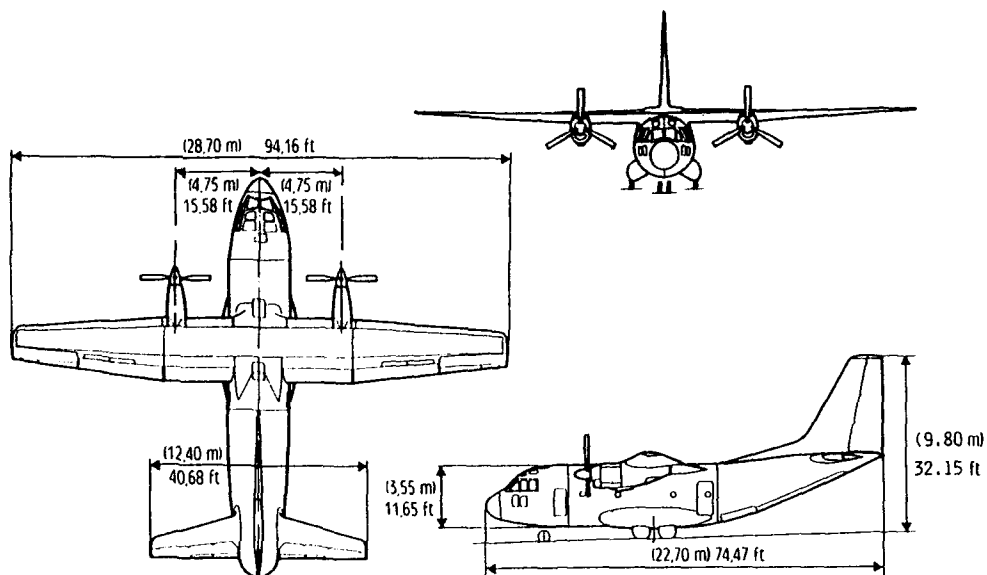


Fig. 1 - Three views of G 222 aircraft.

TYPICAL TRANSPORT MISSIONS

Take-off weight = 58 400 lbs.
(26 500 Kg)

Reserve fuel = 10% of initial fuel

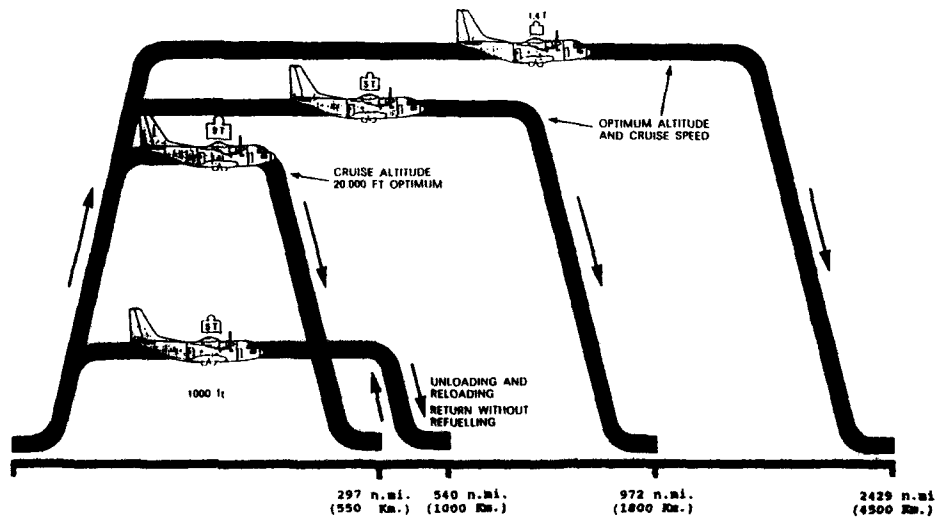


Fig. 2 - The four typical missions of G 222 aircraft.

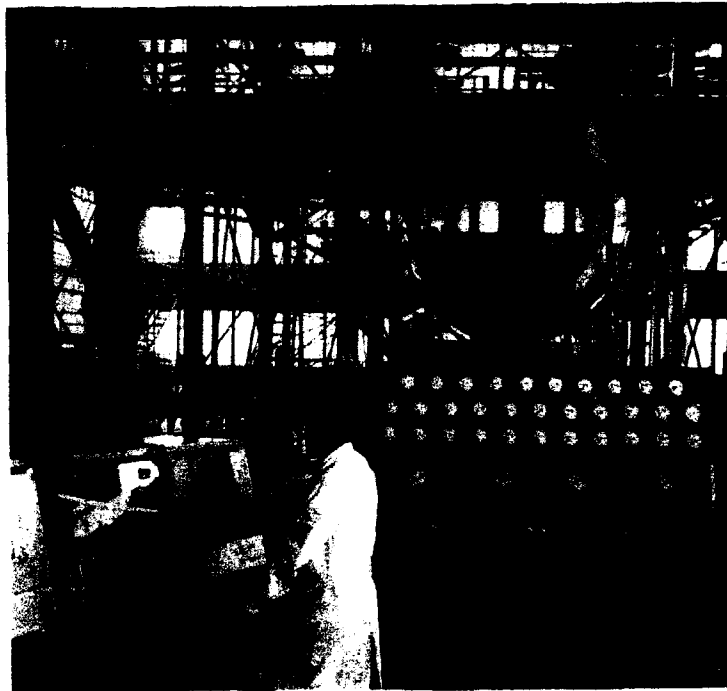


Fig. 3 - G 222 full scale fatigue testing.

FLIGHT TYPE	NO OF FLIGHTS IN ONE BLOCK OF 175 FLIGHTS	CYCLES PER FLIGHT TYPE					CYCLES PER FLIGHT
		I	II	III	IV	V	
D	1	1	1	3	3	7	15
C	5	-	1	2	2	6	11
B	9	-	-	1	2	5	8
A	160	-	-	-	1	5	6
TOTAL NO. CYCLES PER BLOCK OF 175 FLIGHTS		1	6	22	191	882	
CUMULATIVE NO. OF LOAD CYCLES PER BLOCK OF 175 FLIGHTS		1	7	29	220	1102	

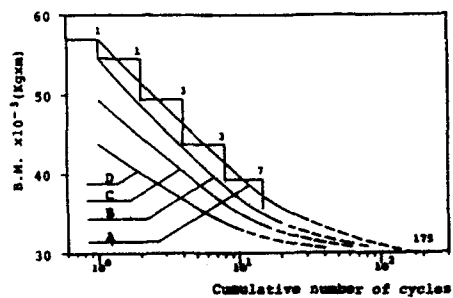


Fig. 4 - Example of different flight severities for the definition of the full scale fatigue load sequence (Cruise segment, "heavy mission").

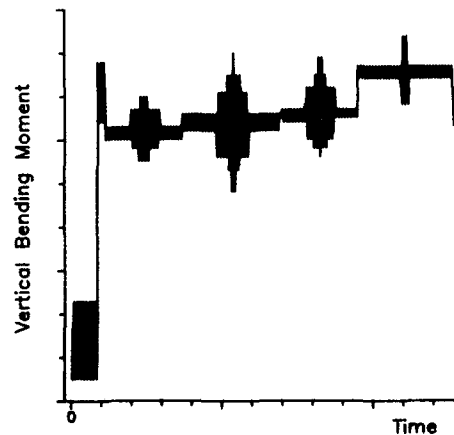


Fig. 5 - Arrangement of load cycles in a simulated flight.

G 222

STRUCTURAL D.T. ANALYSIS

G 222 - CENTER WING SECTION -
 (2.1C) WING-TO-FUSELAGE ATTACHMENT AREA
 (CONNECTING LINK EXTERNAL ATTACHMENT)
 (AT FRAMES 18,20,22 AND RIB 2)

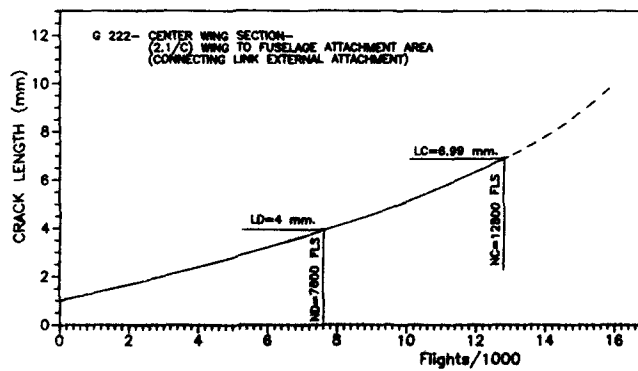
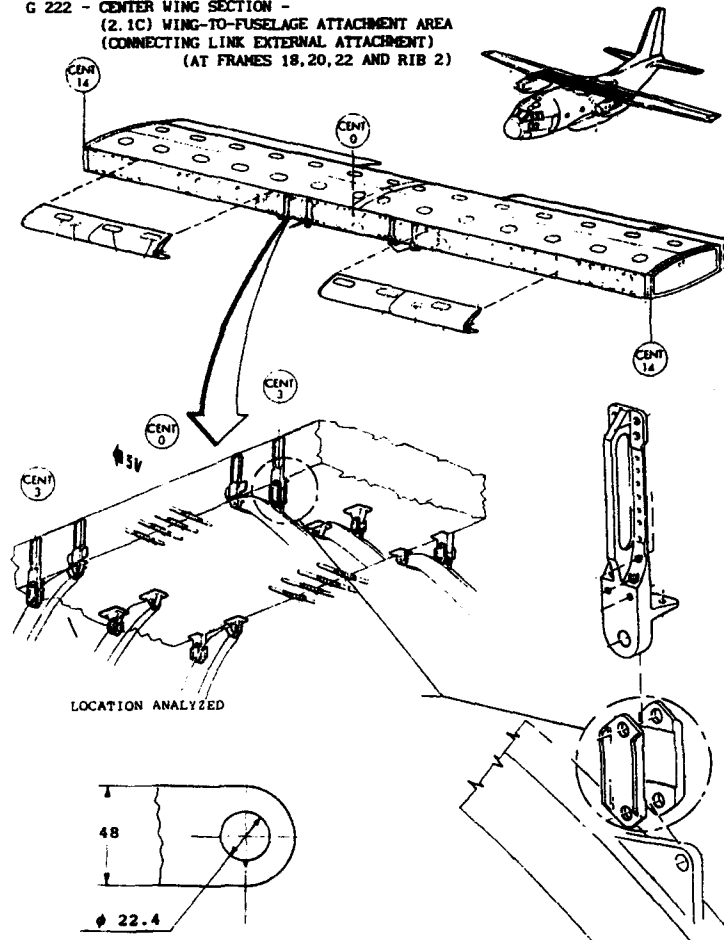


Fig. 6 - Example of Damage Tolerance analysis.

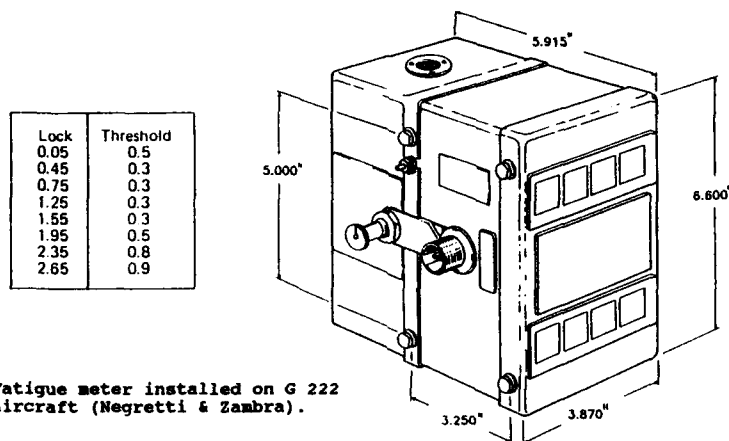


Fig. 7 - Fatigue meter installed on G 222 aircraft (Negretti & Zambra).

FLYING LOG DATA SHEET

AIRCRAFT SERIAL No.				STATION UNIT				TOTAL AIRCRAFT HOURS							
Date	Alt Up at take-off (ft)	Fuel at take-off (lb)	Fuel at landing (lb)	Stores Carried (lb)	Stores Air Delivered (lb)	Duration of flight (min)	Time Below 1500 ft AGL (min)	Time with flaps down (min)	Avg Altitude AGL (ft)	Average Cruise Speed (kts)	Pressure No	Turbulence No	Surf is Good	Name of Captain (print)	
1															
2															
3															
4															
5															
6															
7															
8															
9															
10															

Fatigue Meter Reading								Signature
0.05	0.45	0.75	1.25	1.55	1.95	2.35	2.55	
1								
2								
3								
4								
5								
6								
7								
8								
9								
10								

Turbulence Light, Moderate, Severe, Extreme	
Fatigue Meter	
Mark	Serial No
Fatigue Meter Cycling (Record each 10000 counts)	
Window	Date Cycle Completed
1st Cycle	
2nd Cycle	

Certified that the Form has been checked for obvious errors and anomalies and amended where necessary in red ink.			
Signature	Rank		
Name	Date		

Fig. 8 - Flying log data sheet.

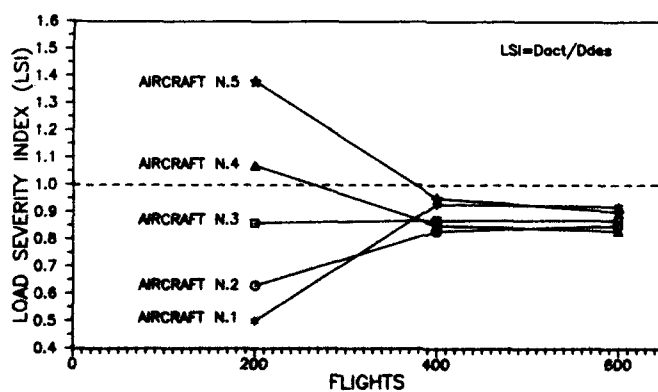


Fig. 9 - Example of fleet management.

LOAD MONITORING OF F-16 A/B AIRCRAFT OF THE RNLAF WITH A SMART ELECTRONIC DEVICE

by

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SUMMARY

Since the introduction of the F-16 weapon system in the RNLAF, load monitoring has been carried out with a Mechanical Strain Recorder (MSR) in each aircraft and a Flight Loads Recorder (FLR) in one sixth of the fleet.

From 1990 on, these recorders have been replaced by instrumentation which is capable of recording peaks and troughs in the signal of a strainingage at the location of the MSR. This new instrumentation has been installed in three aircraft per squadron. The result is a calculated severity per mission type per squadron. By using the mission type mixture, the severity per base, squadron or tail number can be established. Inspection schemes, based on the recently developed Fleet Structural Maintenance Plan for the RNLAF, can be changed accordingly.

Mission type and take off configuration for each flight is available from the debriefing form.

Besides the single channel measurements, supporting measurements with more channels have been carried out. In this way it was checked whether the load severity for the MSR location can be used for other structural locations.

1. INTRODUCTION

Since 1979 F-16 A/B aircraft are flying in the RNLAF, replacing the F-104 G and NF-5 aircraft. In 1991 the last squadron will be equipped with this aircraft. At the time of the introduction of the F-16 weapon system, inspection intervals were based on damage tolerance and durability calculations using the design spectrum. From the start on, it was foreseen to supply each operator with an updated Fleet Structural Maintenance Plan (FSMP). For this reason, a Flight Loads Recorder (FLR) was installed in each sixth aircraft. In this way, three years of actual usage and loading environment data of each operator became available for the update of the FSMP. For the RNLAF it was for the first time that aircraft were equipped with load monitoring instrumentation from the moment of introduction of the aircraft. To the USAF an updated version of the FSMP was presented in 1986, for the RNLAF the first part of the updated FSMP became available in 1989, reference 1. It is expected that in 1991 the last parts of the FSMP will be ready for publication. In the case of the RNLAF the update was made in the framework of the so-called KPAF-ASIP program which started around 1984. In this program for the European Participating Air Forces also use has been made of the results reached in the USAF program.

Also from the introduction of the aircraft in the RNLAF inventory, each F-16 was equipped with the Leigh Mechanical Strain Recorder (MSR) in order to carry out Individual Airplane Tracking (IAT). The MSR records by means of a scratch on a metallic tape a strain which is proportional to the wing root bending moment. In the present paper it will be discussed why the RNLAF has decided to build out the FLR and MSR instrumentation and replace it by a smart electronic device in a limited number of aircraft.

A description will be given of the set-up of the data collecting procedure and of the actions taken to ensure the quality of the accumulated data.

Attention will be given to the calculation of the "damage", in terms of crack growth, of the recorded load sequences. Examples of the data recorded and subsequent analysis results of the data base will be shown in the next chapter.

In the last chapter something will be told about supporting multi-channel measurements in which besides vertical acceleration and wing root bending strain also a strain at a different structural location has been recorded.

2. BACKGROUND OF RNLAF STRAIN MEASUREMENTS

As mentioned before, the load monitoring of the F-16 was originally planned to make use of FLR and MSR instrumentation. Processing and analysis of the recorded data were to be performed in the USA. After about 3 years of operation and recording the RNLAF still had not received back any results. In fact it appeared that at that time there was no actual contract to analyse the data.

However, at the same time the RNLAF did experience in service cracks at a total number of flight hours which suggested a more severe usage and loading environment than the USAF. Also there was some concern within the RNLAF about the severity of air display flying. It was decided to equip one aircraft with a simple strainingage instrumentation. The choice was made for a full strainingage bridge at the location of the MSR. In this way it was possible to compare the results of these measurements with the results of the MSR recordings of the RNLAF and other airforces (see Fig. 1).

At that time, MSR had already gained some experience with the so-called "Spectrapot-10". This is a one-channel recording device which takes care of the signal conditioning of the strainingage bridge and carries out a data reduction in flight by means of a micro processor. The signal is searched for peaks and troughs and only these are stored on a solid state memory. Also it is possible to record time made on the memory. In this way it is known in which part of a flight the loading cycles occur. Both, range filter and time mark can be set at different values. It was decided to use this Swiss made device for recording of the data. In 1984 the measurements started at the first RNLAF F-16 air base, namely Leeuwarden AB, with one aircraft.

In 1985 two more aircraft got provisions for installation of a Spectrapot device. The intention was to measure with two aircraft and have one aircraft for spare.

Also at the second F-16 air base, which became operational in 1985, three aircraft received provisions for a Spectrapot instrumentation. From earlier load measurements with F-104 aircraft a large difference in usage and loading experience had been observed between the two air bases. The RNLAF wanted to know if this was also the case for F-16 aircraft. Besides, at the second base a new version of the F-16 aircraft, namely block 15 instead of block 10, was introduced. One of the larger differences between these versions is a larger horizontal tail on the block 15 version.

In 1988 the photo reconnaissance squadron and a first squadron at a third air base converted to F-16 aircraft and again Spectrapot instrumentation was built in.

In the meantime, FLR recorders were already built out when the analysis of the batch of flights recorded till that moment started at General Dynamics in order to develop an updated version of the FSMP for the RNLAF. Based on earlier load monitoring programs, reference 2, and the results of the Spectrapot measurements so far, the RNLAF was of the opinion that the differences in usage and loading experience between aircraft in the same squadron were not large enough to make individual aircraft monitoring resulting in a different maintenance schedule for each aircraft a necessity.

In addition, the reliability of the MSR, its accuracy and the experiences with data processing and data analysis were such that the RNLAF decided in 1989 to use only Spectrapot instrumentation for the load monitoring of its F-16 fleet from January 1, 1990. The number of Spectrapot's was increased to three per squadron with five aircraft per squadron having provisions for the instrumentation. In this way the instrumentation can be put in another aircraft in case of long term maintenance or modification of the aircraft.

In total 8 squadrons are now equipped with the Spectrapot instrumentation. Since the start of the program in 1984 new versions of the Spectrapot instrumentation became available with larger memories, better resolution, real time clock etc. Such a new version of the Spectrapot, "Spectrapot-4C", is used for most if the instrumented aircraft, see figure 2.

Goal of the load monitoring program is to keep track of possible changes in usage and loading environment during operational flights of the RNLAF. Inspection intervals may be adapted if the change with respect to the reference usage and loading environment is too large. For the reference the operational flying till about 1985 has been taken. This is also the basis for the updated FSMP for the RNLAF which is in use from the beginning of this year.

1. OPERATIONAL LOAD MONITORING RNLAF

The load monitoring of F-16 aircraft of the RNLAF is carried out on a sample of the fleet, namely on three aircraft per squadron. Recorded are a sequence of peaks and troughs in the signal of a strainage bridge at the original location of the MSR. This location is on one of the major carry through bulkheads of the F-16 airframe. The signal is representative for the wing root bending moment, which is a good measure for the loading experience of wing and centre section of the fuselage.

The choice has been made for a full four gage bridge in order to have a large signal and to have temperature compensation. The gages have an electrical resistance of 350 Ω and a gage length of 6 mm. They have been manufactured by TML (TYPE FCL-6-350-23).

On the signal an analog filtering is performed by the Spectrapot instrumentation, using a -3 dB point at 20 Hz. In addition, a range filter of about four percent of the full measuring range is used for the Spectrapot-4C, (about 7 percent for the Spectrapot-1C).

Further, a time mark is set after each half minute interval, (1 min for the Spectrapot-1C).

Extensive EMI tests have been carried out on the Spectrapot instrumentation to check on emission and susceptibility. It is very important to make sure that the instrumentation is not susceptible for EMI. Because of the data reduction to peaks and troughs it is in general impossible to tell from the results whether a peak is a result of a real loading of the structure or of EMI.

1.1 Comparison strainage bridge output with MSR strain

In order to be able to compare the results of the Spectrapot measurements with those of the MSR, the calibration of the strainage bridge was carried out by comparing the recorded sequences of MSR and Spectrapot of the same flights, for examples see figures 3 and 4. In both signals the same, large, load cycles were taken and subsequently a "MSR μ strain" per Spectrapot-1C count was derived. It is planned to repeat this calibration for the Spectrapot-4C. With the larger resolution the accuracy of the calibration can be improved.

In addition, it was found in a recent investigation, reference 3, that the accuracy of the MSR is not too good. Three different MSR's were used in this test. The output of the MSR's were compared with the measured strains of strainages on the same rod on which two MSR's were installed. In a fatigue testing machine a few load cycles were applied. The result did show a large difference in the output of the MSR's if compared with the average strain of the strainages, namely 2, 7 and 14 percent smaller! So it is necessary to calibrate the individual MSR, which will be used for the calibration of the strainage bridge.

1.2 New location of strainage bridge

In figure 1 two locations for the strainage bridge are shown.

At the start of the measuring program a location half way between the attach points of the MSR was chosen. Partly this was based on information from General Dynamics that the gradient in the strain over the bulkhead was small. However, the strain measured with the strainage turned out to be approximately 25 percent larger than the one measured with the MSR.

It was decided to measure with strainages the actual strain distribution near the location of the MSR. In total 16 strainages were placed. The aircraft was set on three jacks. Loading was applied by jacks under the wingtips of the aircraft using existing hard points. The load was measured with a load

cell between the top of the jack and the wing. In this way loads up to 6500 lb could be applied at each wing tip giving about 23 percent of limit load in terms of wing root bending moment, see reference 3.

Figure 5 shows the strain distribution in the bulkhead. It was found that the strainage bridge used in the load monitoring program was located in an area with a large strain gradient as a result of the transverse flange. For practical reasons it is advisable to use an area with a more uniform strain distribution. In this way, small differences in the exact location of the strainage bridge between different aircraft still give the same output. As a result a new location for the strainage bridge was chosen. Next, the "old" and the "new" bridge were placed on the test aircraft and a factor between the output of both bridges was established.

3.3 Data handling

Management of the load monitoring program for the F-16 aircraft of the RNLAF is carried out by NLR. In each squadron an officer is responsible for changing the solid state memories at a 25 hours interval, (one week for Spectrapot-1C), and if necessary replace the Spectrapot to another aircraft with "provisions for".

Besides the load data on the solid state memory, for each recorded flight a debriefing form has to be completed by the crewchief and the pilot, see table 1. In this way, additional information about the flight becomes available, such as:

- Administrative data.
- Take off store configuration.
- Mission type, including some information about mission content.

Table 2 gives the mission type description as used in this program for the RNLAF. The definition of these mission types have been determined in close cooperation with pilots. It may be clear that completion of the debriefing form is essential for the program. This means that much attention has to be given to this aspect. Close cooperation between RNLAF personnel at the air base and NLR personnel is very important.

In figure 6 the data flow in the load monitoring program is given. As can be seen, the results are stored in the F-16 fatigue data base.

Dedicated software programs have been written to analyse the data stored in the data base. Selections of groups of flights can be easily made. For example: per squadron, tailno, time interval, mission type and combinations of these items. For each selection a table is made as given in table 3. In the next chapter more will be told about the quantities MCSI en SCSI as used in table 3.

Each half year, a review of the recorded data is presented in a standardized "half yearly survey report". For this report selections are made for "all RNLAF" and per squadron. In each case for three time intervals, namely: last half year, last year and last three years.

For the same selections an overview per mission type and aircraft type is given of the distribution of configuration classes.

Here, something has to be told about the "configuration class".

The F-16 is flown in a wide variety of store/fuel configurations. In fact a few hundred different configurations have been experienced so far. In order to present the configuration usage a set of ten different "configuration classes" have been defined on the basis of a calculated wing root bending moment per g at take off.

In the calculation the influence on the wing root bending moment of the additional masses of the stores and of the increase of the lift force on the wing as a result of the increase of the take off weight are taken into account.

In addition to the "half yearly survey reports" to the RNLAF, the F-16 fatigue data base is used for more detailed analysis in ad hoc projects.

4. CALCULATION OF "DAMAGE" PER FLIGHT

A few years ago NLR developed the so-called "Crack Severity Index", CSI, concept for quantification of the damage of recorded stress spectra in terms of crack growth potential, reference 4.

In earlier monitoring programs (F-104 G, NF-5 A/B) the "severity" of recorded load spectra used to be quantified in terms of a "Load Severity Index", LSI, which was calculated on the basis of a Miner type fatigue life calculation. It was felt that for the F-16, an aircraft which is designed according to the "Damage Tolerance" concept, a more modern method had to be used based on crack growth potential. The CSI concept was developed using recorded load sequences of the F-16 load monitoring program. It is important to realize that the calculated CSI value is a relative damage figure for a specific location and loading spectrum.

In the development of the CSI method use was made of the latest concepts with regard to crack growth prediction, including crack closure and associated crack growth retardation. An important quantity is the minimum crack opening stress. In the CSI concept this opening level is a function of the maximum and minimum stresses that are reached on the average of once in thirty flights.

For validation of the CSI concept fatigue tests on single side cracked hole specimens have been carried out, reference 5 and reference 6. In table 4 and figures 7 and 8 the load sequences and results are given. The "basic" sequence is a batch of 200 flights recorded at one air base. Derived from the "basic" sequence are the variations LOWI (flights appear in order of increasing MCSI per flight), HILO (decreasing MCSI), "Omission" (cycles below stress opening level deleted), "Truncated" (truncation of 13 peaks above 134 MPa to 134 MPa), "Low" (flights with MCSI > 1.73 deleted), "High" (flights with MCSI < 1.24 deleted).

In the "basic" sequence the mission mixture from Loosdrecht AFB, flown in 1985, has been used. The sequence VK basic (85) is made with recordings and mission mixture from a second F-16 air base, namely Volkel AFB. In table 4 the quantities SCSI and MCSI are used for presentation of the crack growth potential of a spectrum. As said before, use is made in the CSI calculation of an average stress

opening level which depends on the spectrum shape under consideration. Of course, the batch of flights for which the average stress opening level is calculated has to be sufficiently large. It was decided that at least a batch of 100 flights is needed for calculation of the average stress opening level. This result is called the SCSI value (for "spectrum" CSI). For calculation of the CSI value for a smaller batch of flights or for individual flights the average stress opening level of a reference batch of 488 F-16 flights, recorded in 1985 at two air bases, is used. This gives the MCSI value (for "nominal" CSI). In figure 9 a comparison is made between the results of the fatigue tests and the CSI calculation results for SCSI and MCSI. From this figure it is concluded that the CSI concept is a reasonably reliable tool for quantification of relative severities for this type of manoeuvre dominated spectra. As may be expected the results for the MCSI calculation are slightly less accurate than the results for the SCSI calculation. Interesting to observe is that the influence of extreme sequencing of flights in the batch of 200 flights is rather limited. Omission of load cycles below the stress opening level has no result on the fatigue test results, as was expected.

5. RESULTS OF THE LOAD MONITORING PROGRAM

In this chapter only a few examples are given of the results which can be made available from the F-16 fatigue data base.

In the half yearly report to the RNLAf more or less the raw data are presented. The most important question is: Is there a significant change in the CSI value per squadron as a result of changes in usage and/or loading environment?

If so, the half yearly report offers the possibility to a certain extent to find out what the reasons are. More detailed analysis can be done by NLR using the complete data base.

In figure 10 some typical examples of different mission types are given. Of course there is a large scatter in the mission content of flights having the same mission type. In figure 11 the resulting "average" spectra for two mission types are given. It is interesting to see that there is distinct difference between the same mission type flown at the two air bases. Differences in take off store configurations, figure 12, and in geographic location of the air bases explain these results.

In combination with a different mission mixture at those air bases, the total load spectrum is also different and consequently the overall CSI-value. Figure 8 shows a severity ratio of 1.32 per flight between the crack growth potential at Leeuwarden air base and the one at Volkel air base. By the way, the PSMP for the RNLAf takes these differences into account.

For a load monitoring program it is essential that new trends in the usage and loading environment are found. In recent years attention has been given to the flying by the RNLAf in Goose Bay (Canada) and to Air Display flying, figure 13. If aircraft fly these mission types for longer periods of time exclusively, a correction to the inspection scheme may be necessary.

In figure 14 variation over the years of the severity per mission type is given for 323 squadron at Leeuwarden air base. Also a comparison with the 311 sqn at Volkel air base is made.

In table 5 the overall result in SCSI value for 323 squadron is shown.

A substantial increase of the SCSI value has been experienced over the years. It may be noticed that the way of operation in 1985 has been taken as the reference basis for the PSMP of the RNLAf.

6. MULTI-CHANNEL RECORDINGS

Sofar a description of the standard load monitoring program, in which only one channel is recorded, has been given. However, most aircraft are equipped with a 4 channel version of the Spectrapot instrumentation. This opens interesting possibilities to carry out adhoc measurements with other parameters. In this way vertical acceleration, a strainage in the aft fuselage and the engine RPM have been or are being recorded in addition to the strainage at the MSR location.

The large advantage is that the data handling is already taken care of in a standardized procedure. Coupling with other data in the F-16 fatigue data base is relatively simple. In figure 15 for a large number of flights of all mission types the relation between strain at the MSR location and the vertical acceleration is given. This figure illustrates the fact that for this type of highly manoeuvrable aircraft measuring of strain in stead of vertical acceleration is a good choice if a location can be found where the strain history is representative for the loading of the most critical parts of the airframe. This is the case for the F-16 aircraft.

In another measuring program a strainage bridge in the aft fuselage at bulkhead FS 479.55 was recorded. Purpose of this investigation was to check how well the results for the MSR location could be used for prediction of the damage at another structural location. In the individual aircraft tracking (IAT) program it is assumed that the MSR data can be used as an indication for the overall load experience of the aircraft.

In the program it was found that this is only true if the mission mixture does not change significantly. In figure 16 it is shown that the loads at MSR location and aft fuselage location are a function of the usage of elevator and rudder and also dependant of airspeed. In other words this is mission dependant. In figure 17 the ratios of SCSI values are given for the most severe RNLAf mission types for MSR location and aft fuselage. For example the SCSI value for the ACT mission type is larger than the one for "all" flights for the MSR location. However, for the aft fuselage the damage is far more. In this case the SCSI for the MSR location underestimates the damage for the other location. For the mission type RANGE this is the other way around.

7. CONCLUDING REMARKS

In this paper the loadmonitoring of F-16 aircraft of the RNLAF using a simple electronic device has been described. The RNLAF contracted the National Aerospace Laboratory NLR in the Netherlands for definition and implementation of the instrumentation.

The data collecting on a routine basis is carried out by NLR and two times per year a survey data report is sent to the RNLAF.

Essential for this program is that only a sample of the aircraft is equipped with the instrumentation. On the other hand much additional information is collected from the debriefing forms.

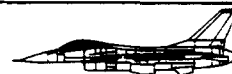
As a result, an adequate and flexible load monitoring system for the F-16 aircraft of the RNLAF is now in full operation as a replacement of older FLR and MSR equipment.

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TABLE 1
Debriefing form

F-16 WING STRAIN MONITORING



11-02-1981

GROUND PERSONNEL

SQUADRON	LOCATION	TAKE OFF	LANDING	TYPE OF AIRCRAFT
322 ☐ 306 ☐	LEEWARDEN	☐	☐	F16-A ☐
323 ☐ 311 ☐	VOLKEL	☐	☐	F16-B ☐
313 ☐ 312 ☐	TWENTHE	☐	☐	RF-16 ☐
315 ☐ 314 ☐	GILZE RIJEN	☐	☐	☐
316 ☐		☐	☐	☐

DATE AIRCRAFT NR TOTAL HOURS BEFORE FLIGHT MEMORY NR FLIGHT SEQUENCE NR ON MEMORY

EXTERNAL STORES CONFIGURATION BEFORE FLIGHT

WINGTIP	1 9	OUTBOARD	2 8	OUTBOARD	3 7	INBOARD	4 6	CENTERLINE	5
launcher	☐	empty	☐	empty	☐	empty	☐	empty	☐
aim 9	☐	u.w launcher	☐	pylon only	☐	pylon only	☐	rack/adaptor	☐
dummy	☐	aim 9	☐	ter	☐	dry 370	☐	dry 300	☐
.....	☐		☐	suu 20	☐	1/2 full 370	☐	full 300	☐
.....	☐		☐	ver	☐	full 370	☐	ecm pod	☐
.....	☐		☐		☐		☐	recce pod	☐
.....	☐		☐		☐		☐		☐

PILOT

FLIGHT DURATION T/O TIME MAX. HUD G POSITION IN FORMATION ☐ FLIGHT LEADER ☐ WINGMAN ☐

BASIC MISSION TYPE				**CROSS ONE**			SEE BACK SIDE		
RANGE	AGNAV	ACT	PI	CAP	DART	GF	IF	RECCE	
☐	☐	☐	☐	☐	☐	☐	☐	☐	☐

if RANGE AGNAV → GUNNERY BOMBING

if ACT PI → NUMBER OF SETUPS if CAP → NUMBER OF INTERCEPTS if DART → NUMBER OF PASSES

NUMBER OF PASSES			
LEVEL	DIVE ANGLE		LOFT
	≤ 15°	> 15°	
☐	☐	☐	☐

if ADDITIONS TO BASIC MISSION TYPE

ACT ☐ PI ☐ QRA ☐ NAV ☐ CAP ☐ ☐

MISSION EVENTS

REMARKS:

TABLE 2
Overview mission types

MISSION CODE	MISSION TYPE DESCRIPTION
AIR/GROUND	RANGE WEAPONS DELIVERY FOR TRAINING, UTILIZING TRAINING ORDNANCE (BOMBS, ROCKETS, GUNS), INCLUDING EVASIVE ACTIONS (AIR/AIR AND AIR/GROUND). PLEASE INDICATE UNDER REMARKS IF LIFE BOMBS WERE USED.
	AGNAV LOW LEVEL NAVIGATION TRAINING, INCLUDING ONE OR MORE SIMULATED GROUND ATTACKS, EVASIVE ACTIONS (AIR/AIR AND AIR/GROUND).
AIR/AIR	ACT AIR COMBAT MANOEUVRING, SELF DEFENSE, DOGFIGHT.
	PI INTERCEPTS FOLLOWED BY AIR COMBAT MANOEUVRING.
	CAP COMBAT AIR PATROL.
	DART AIR TO AIR FIRING ON A DART TARGET.
GENERAL	GF GENERAL FLYING (AIRCRAFT FAMILIARISATION IN SAFE HANDLING AND OPERATION). THIS CODE ALSO INCLUDES CONVERSION, TRANSITION, FCF AND ECF.
	IF INSTRUMENT FLYING IN REAL OR SIMULATED IFR CONDITIONS. THIS CODE ALSO INCLUDES LONG RANGE FERRY- AND NIGHT FLIGHTS.
RECCE VISUAL/PHOTO RECONNAISSANCE.	

TABLE 3
Review of recorded load data

Example table "Review of Recorded Load Data"

mission-/ aircraft- type	nr. of flights	ave. flight duration (hr)	TO WEIGHT		LOAD EXPERIENCE				crack op. level (MPa)
			ave. (lb)	st.dev. (lb)	NCSI		SCSI		
					per flight	st.dev.	per flight	st.dev.	
range act									
.									
.									
all									
F-16A F-16B									
BL-10 BL-15									

Table repeated:

- per squadron and all RNLAf
- last half year
- last year
- last 3 years

Table repeated:

- per squadron and all RNLAf
- last half year
- last year
- last 3 years

cr.o.l. = calculated crack opening level

NCSI = calculated with a fixed cr.o.l

SCSI = cr.o.l. calculated for batch

TABLE 4
Validation of CSI concept

loading sequence (block of 200 200 flights)	type of crack	average flight duration (hr)	average number of flights in test	relative severity in test	NCSI per flight	SCSI per flight
BASIC (LW 85)	through thickness	1.08	6407	1.00*	1.00*	1.00*
LOHI	"	1.08	6486	0.99	1.00	1.00
HILO	"	1.08	7048	0.91	1.00	1.00
OMISSION	"	1.08	6383	1.00	1.00	1.00
TRUNCATED	"	1.08	5161	1.24	0.99	1.08
LOW	"	1.08	10434	0.61	0.50	0.55
HIGH	"	1.02	3522	1.82	1.90	1.77
LW BASIC (85)	corner	1.08	9713	1.00*	1.00*	1.00*
VK BASIC (85)	"	1.12	12834	0.76	0.57	0.74

* relative severity with respect to "basic"

TABLE 5
Example of SCSI change in time
323 sqn. LW AFB

	SCSI		flight duration (hr)	crack opening level (MPa)
	per flight	per hour		
1985	1.25	1.12	1.12	41.36
1989	1.61	1.66	0.97	43.29

- 1985 is basis for FSMP RNLAf
- SCSI may be used for adaption of inspection intervals

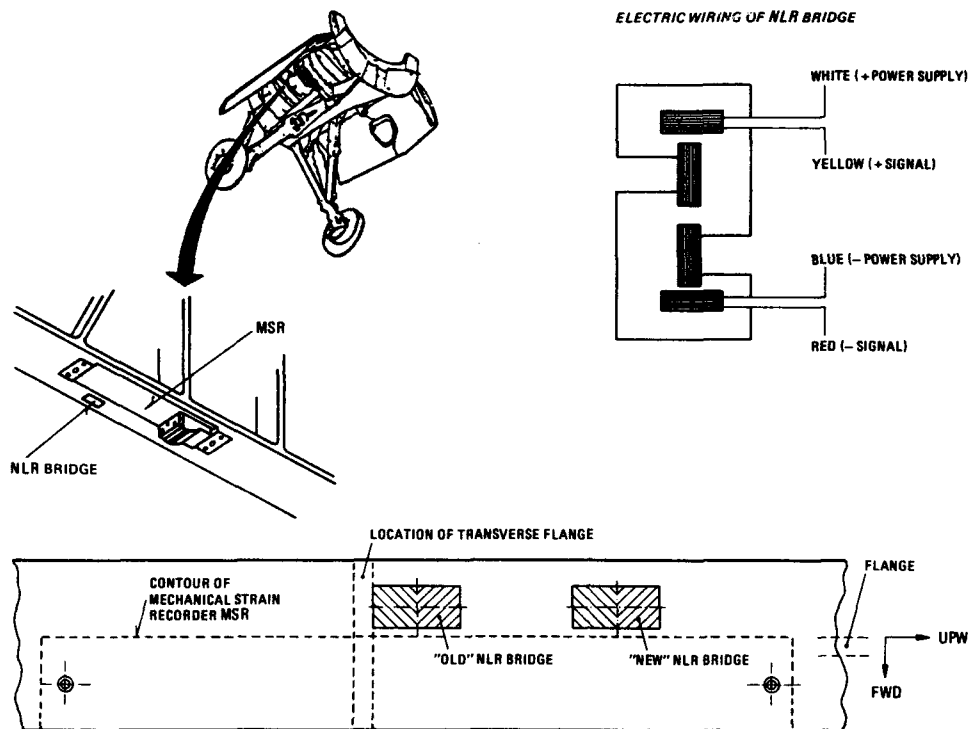


Fig. 1 Location of strainage bridges on FS 325.8 bulkhead

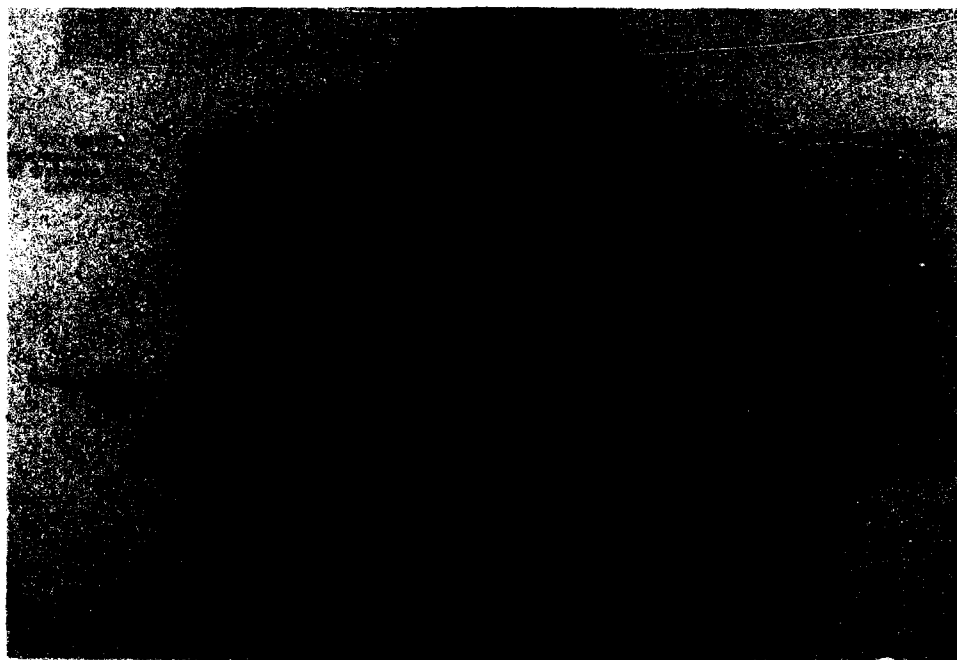


Fig. 2 Spectrapot-4C data collector

12-10

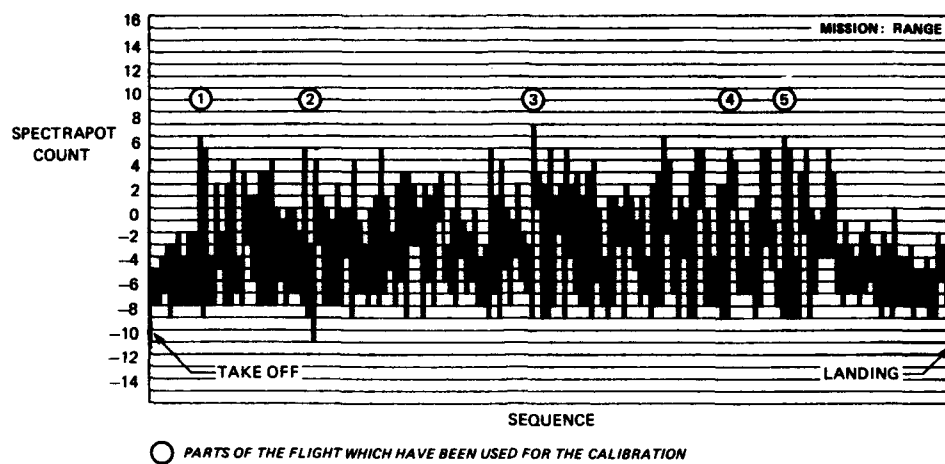


Fig. 3 Sequence of peaks and troughs for calibration

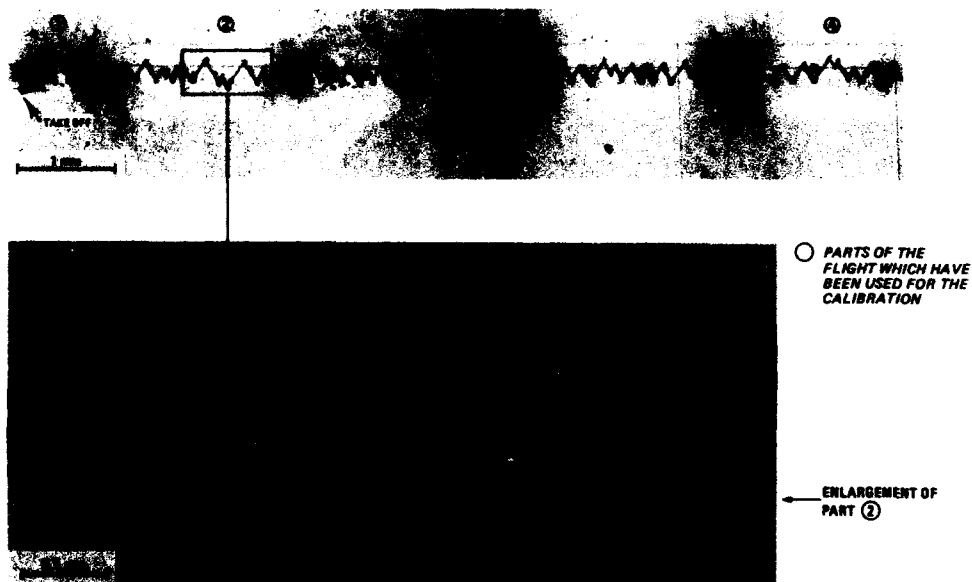


Fig. 4 MNR trace of the same flight as figure 3

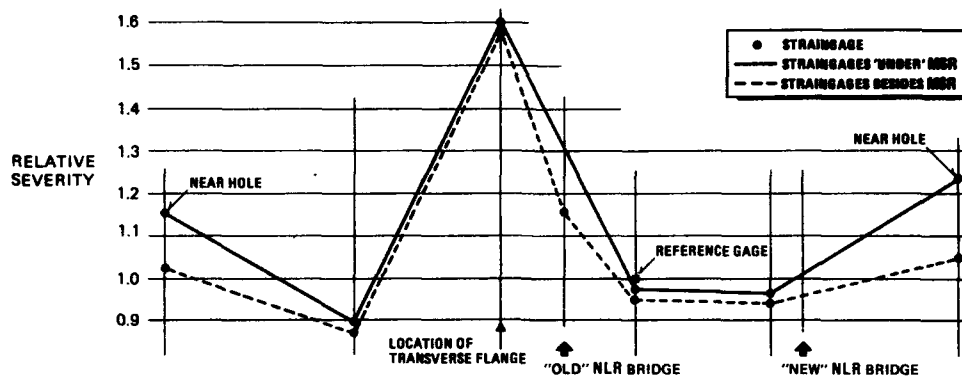


Fig. 5 Strain distribution at the MSR location on FS 325.8 bulkhead

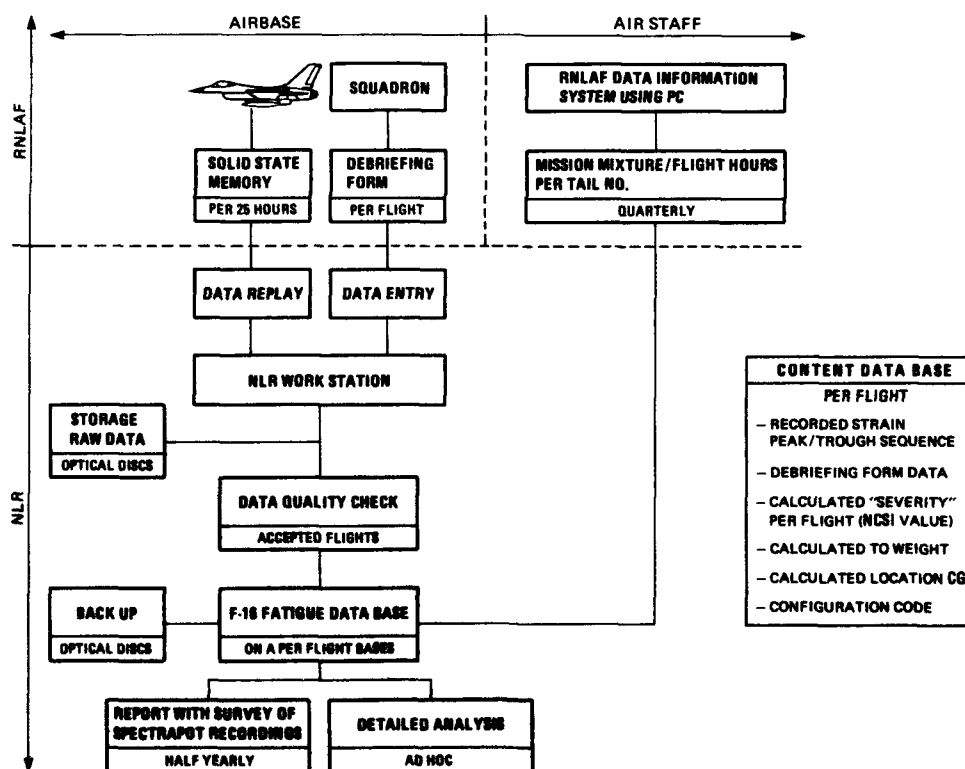


Fig. 6 Data flow in F-16 load monitoring program

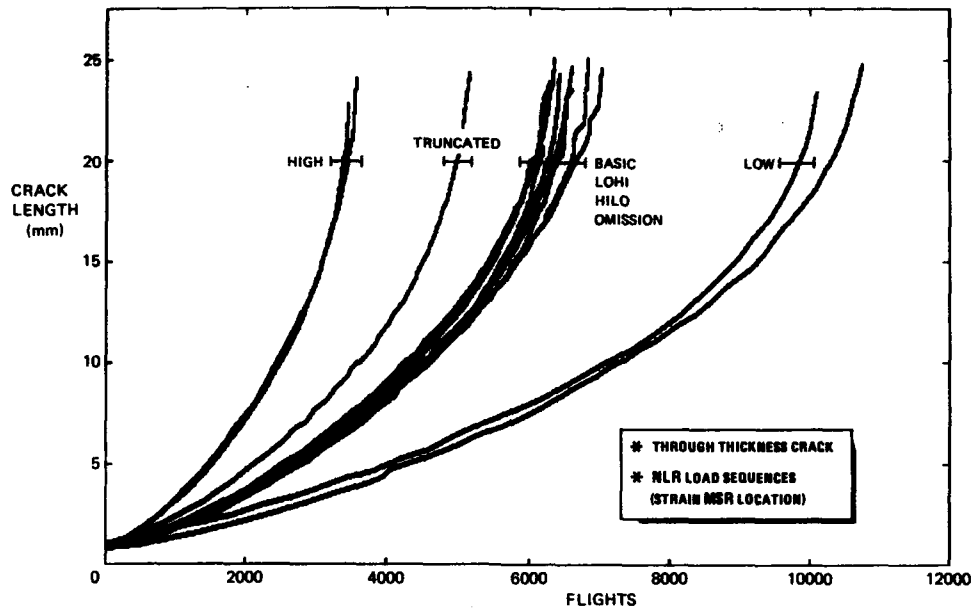


Fig. 7 Crack growth test results for CSI validation

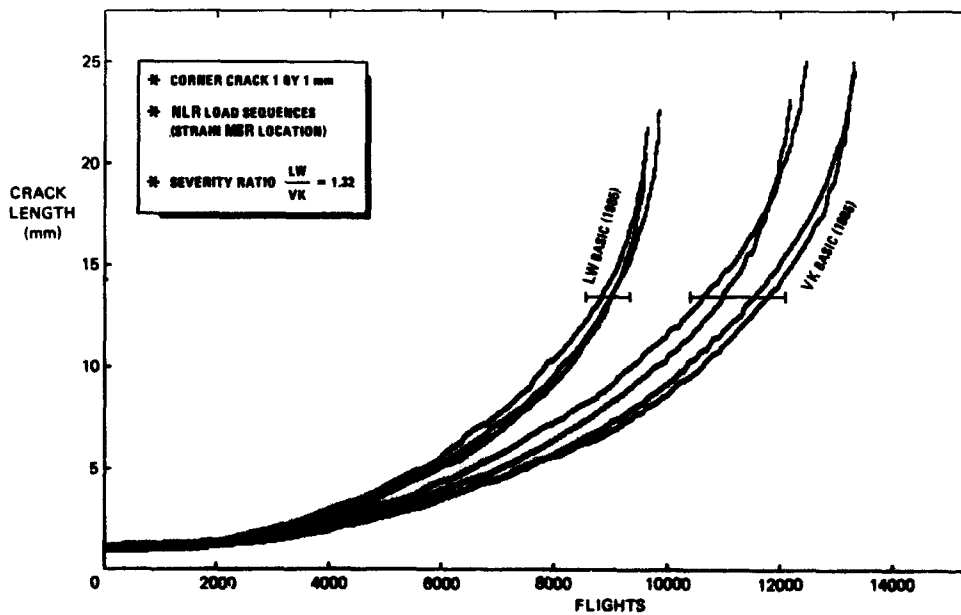


Fig. 8 Crack growth test results for CSI validation

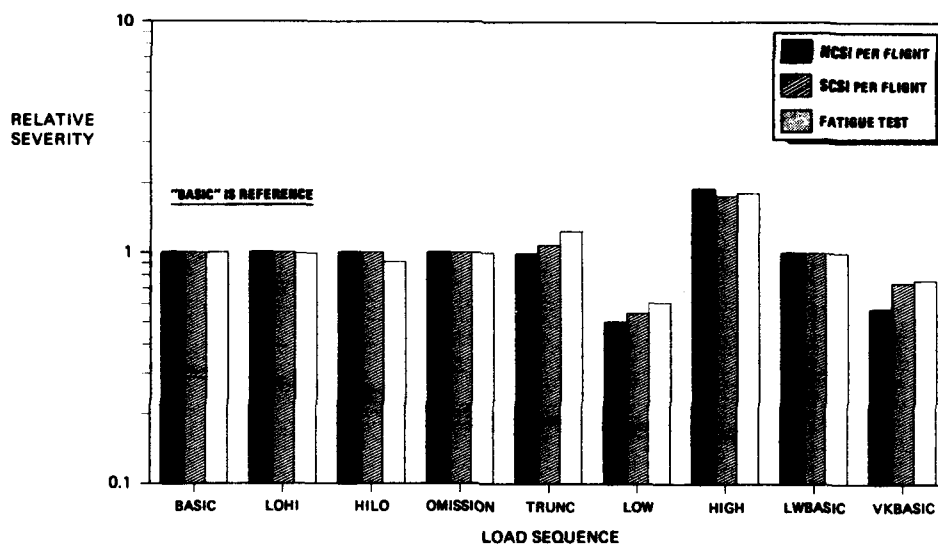


Fig. 9 Validation CSI concept

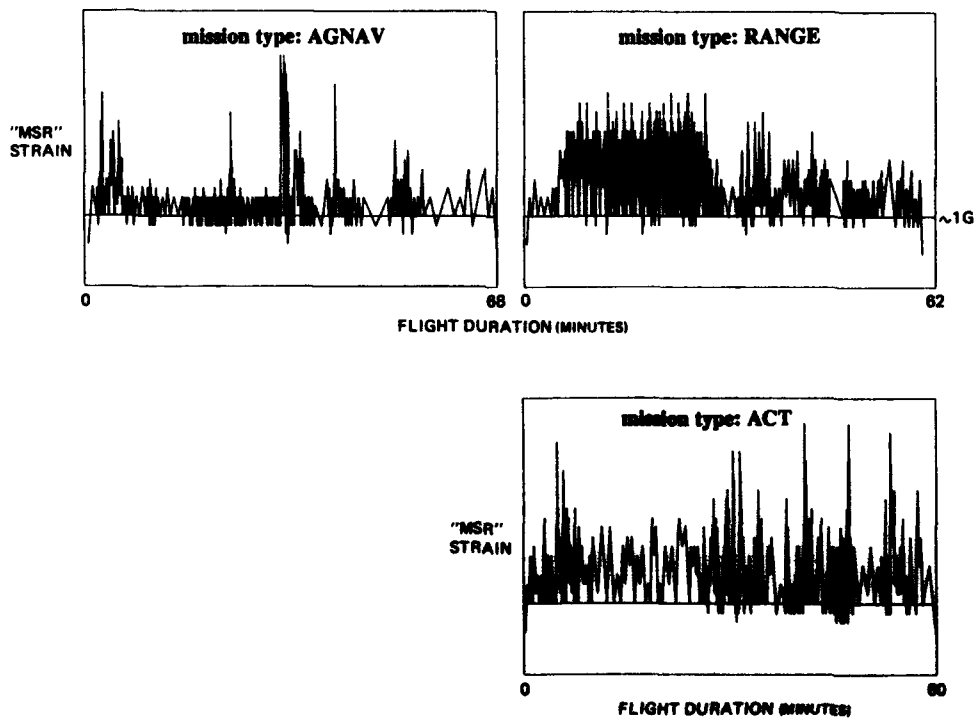


Fig. 10 Examples of different mission types

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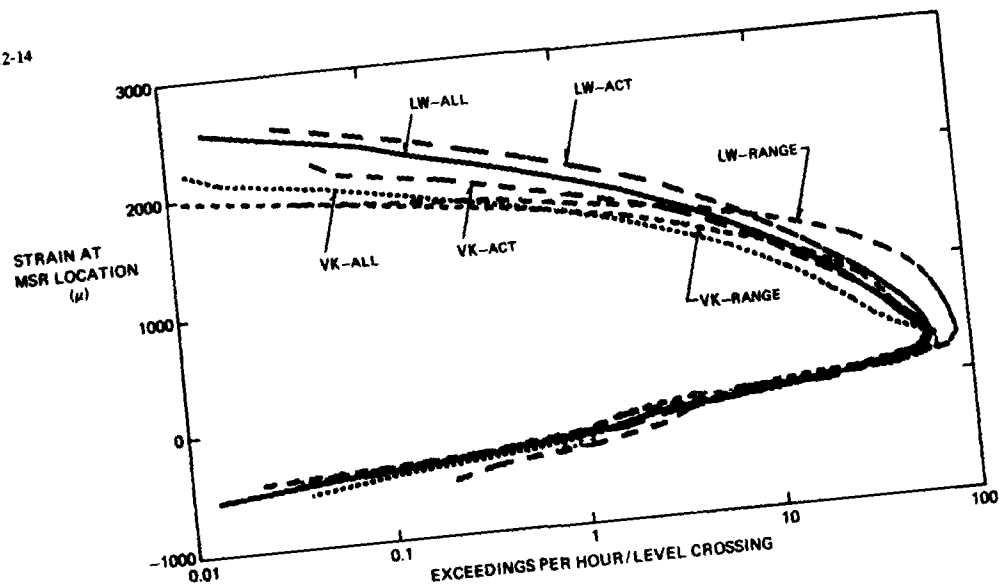


Fig. 11 Strain spectra per mission type for LW and VK air base

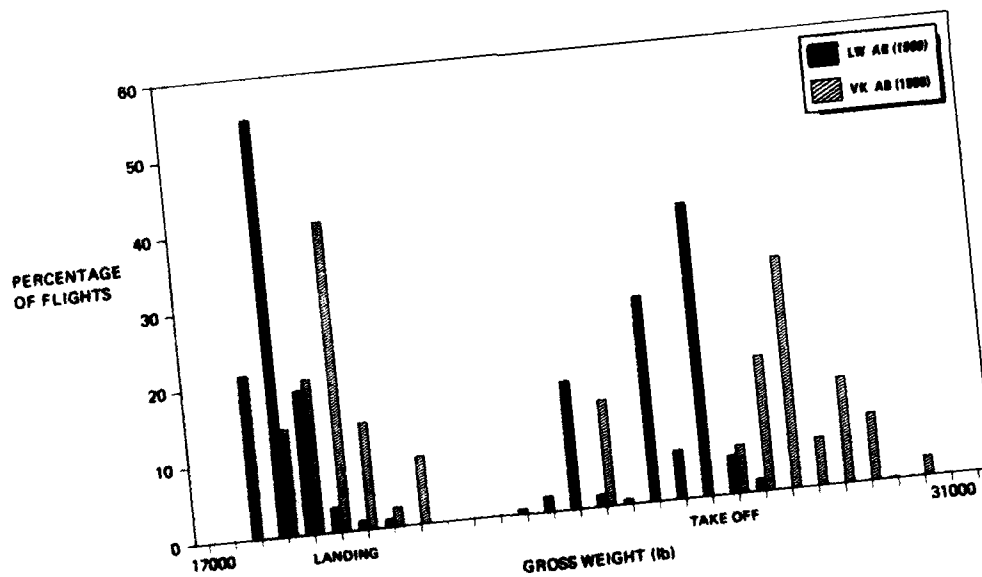


Fig. 12 Take off and landing weight distribution at two air bases

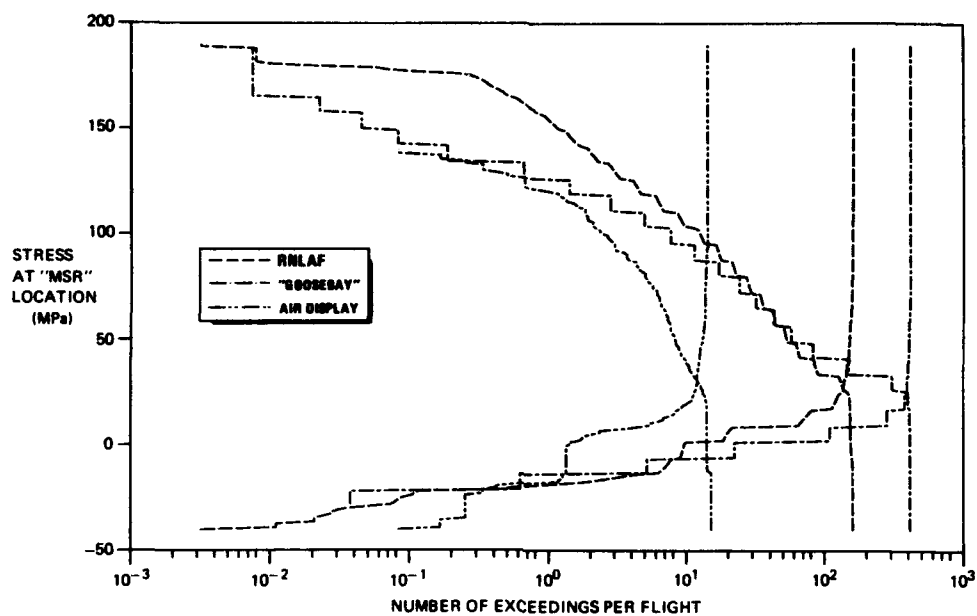


Fig. 13 MSR spectra of "extreme" mission types

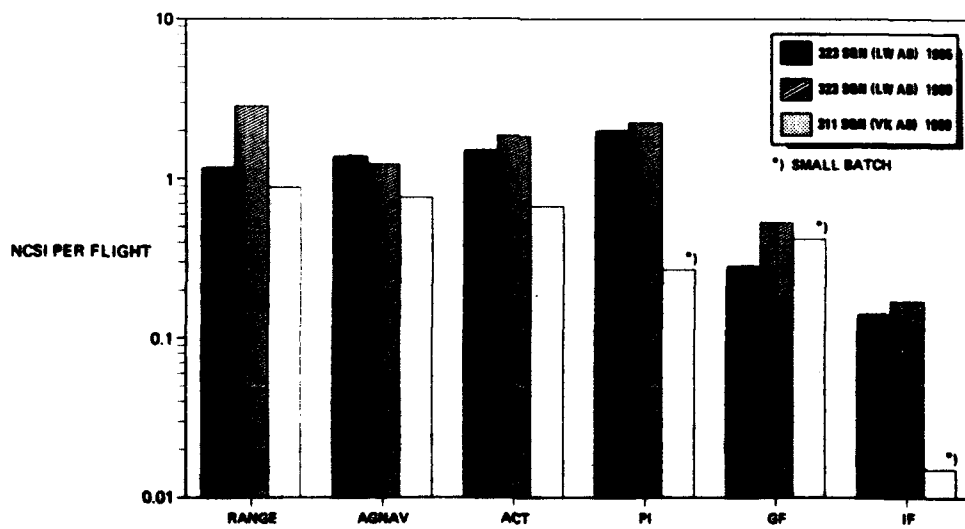


Fig. 14 Variation of NCSI per mission type

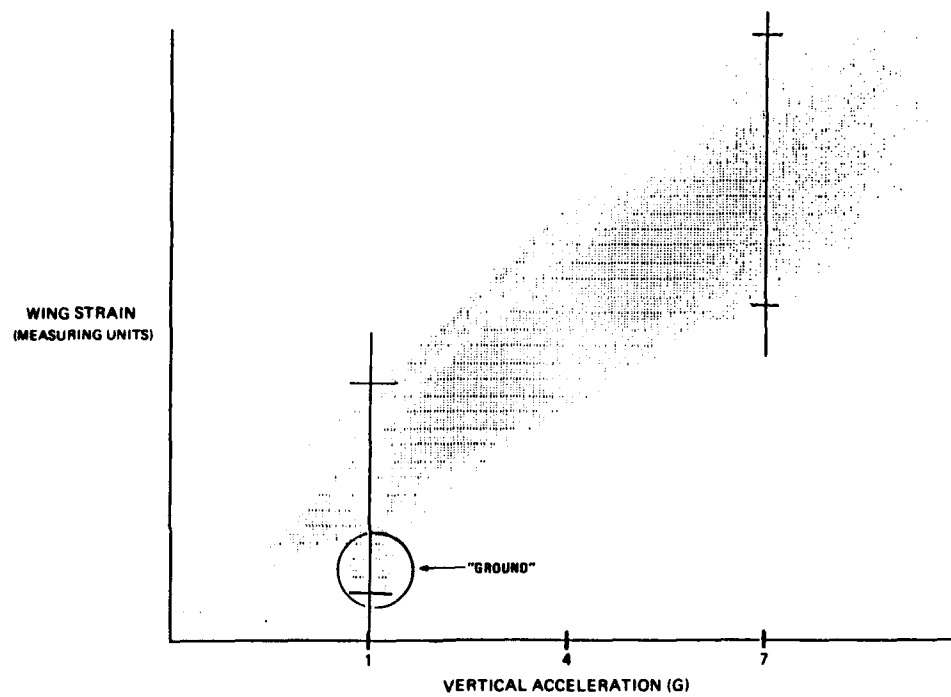


Fig. 15 Relation G-strain (based on about 250 flights)

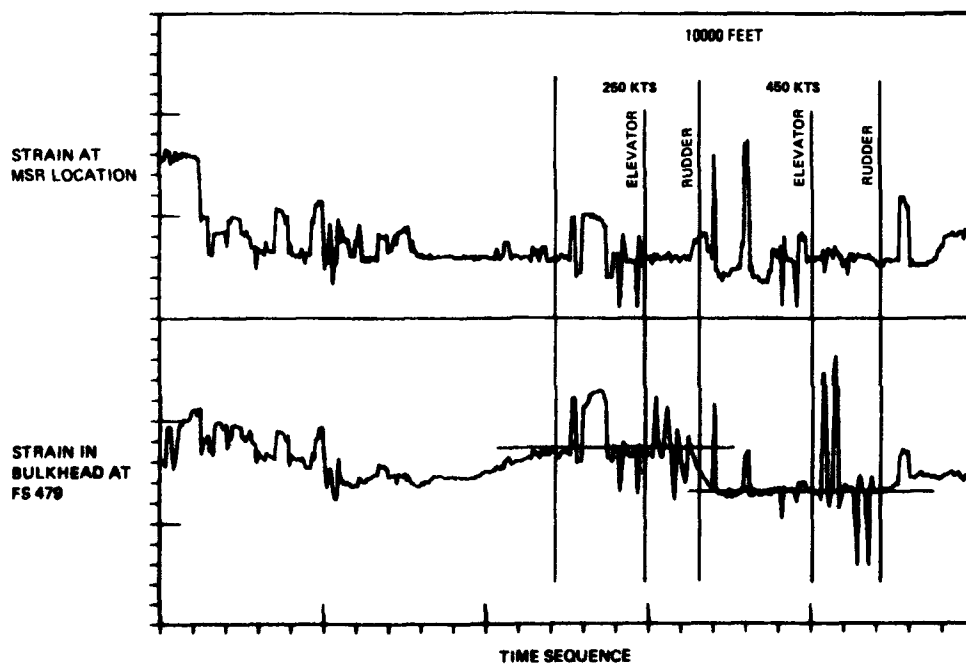


Fig. 16 Example of elevator/rudder input on measured strains

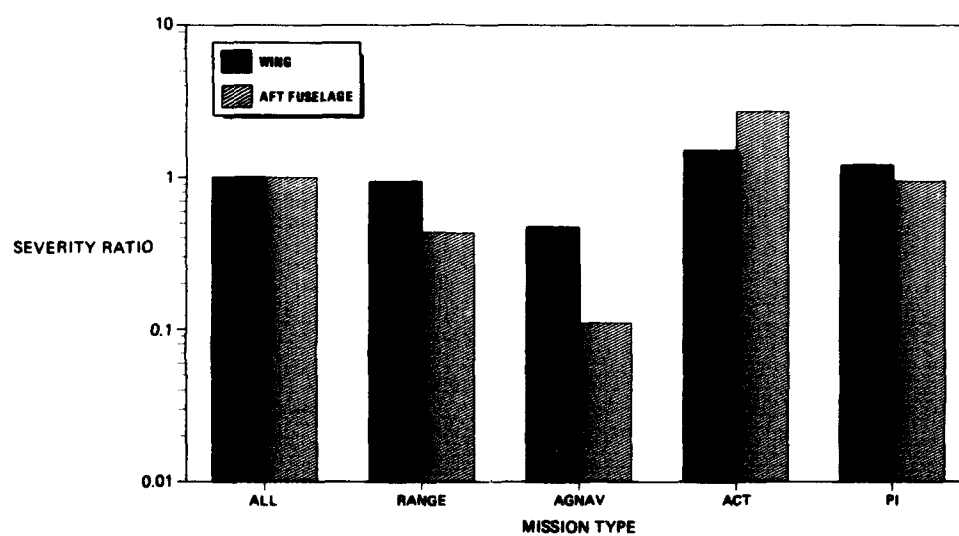


Fig. 17 Relative severities wing/aft fuselage

AIRCRAFT TRACKING FOR STRUCTURAL FATIGUE

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Summary

Aircraft tracking is carried out for flight safety reasons, but with an increasing tendency also for economic reasons. With the latter aspect the cost-to-performance ratio becomes more and more important. To take care of both, for the WS Tornado the parametric data acquisition of the crash recorder was extended to a multi-level tracking concept composed of the sectors

- Temporary Aircraft Tracking
- Selected Aircraft Tracking
- Individual Aircraft Tracking

The key elements of flight monitoring are the flight recorders that are distributed throughout all squadrons on a statistically representative basis and that register operating data for Selected Aircraft Tracking. In the Temporary Aircraft Tracking sector, the recorder parameter set also contains strain gauges in the various fatigue critical areas. Cyclical reading of these strain gauges ensures that any faults are revealed in the parametric algorithms.

Individual Aircraft Tracking is carried out on the basis of a reduced pilot parameter set. The data transfer from the aircraft to the evaluation centre for this task was converted from manual registration to electronic data processing, increasing the data processing capacity and at the same time significantly improving data quality.

List of Abbreviations

A/C	Aircraft
FCA	Fatigue Critical Area
FPS	Full Parameter Set
IAT	Individual Aircraft Tracking
OLMOS	Onboard Life Monitoring System
PFS	Pilot Parameter Set
RPS	Recorder Parameter Set
SAT	Selected Aircraft Tracking
TAT	Temporary Aircraft Tracking
WS	Weapon System

1. Introduction

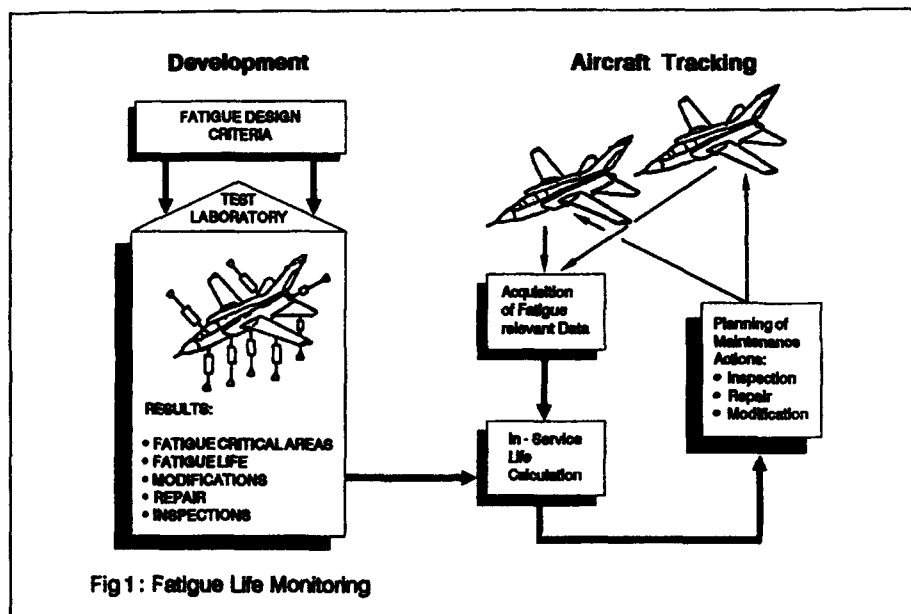
Development specifications for military A/C generally require a fatigue test of the entire airframe, or at least its principal components, as proof of structural integrity. Although test speeds allow the simulation of a large number of daily flights, several years are necessary for the completion of the fatigue test, allowing for interruptions for repair, modification and inspection. Thus, for reasons of time and money series production of A/C begins before all results of the fatigue tests on the various FCAs are known and the necessary modifications made. Individual batches therefore still have structural FCAs, albeit with decreasing frequency, whose inherent fatigue life in all probability will not last throughout the planned operational usage.

Hence necessity to retrofit and repair structural components is unquestionable. The question is, which A/C requires which repair, and when. There are several reasons asking this question, for example:

- restrictions on operational usage for unmodified structures and
- cost-effective modification

In order to answer these questions satisfactorily, it is necessary to know the fatigue life ratio of the operational stress spectrum on FCAs to the stress spectrum in the fatigue test, i.e. how many operation flight hours are equivalent to a simulated flight hour in the test. Depending on the structural point in question, such spectra comparisons range from

- either higher or lower operational stress spectrum compared to the test to
- the stress spectrum simulated in the test not being representative in detail for operational usage.



Thus, it is both useful and essential for the A/C user to have an instrument, which answers the complex questions mentioned above and enables necessary activities to be planned. This task is dealt with by the 'Aircraft Tracking for Structural Fatigue' (Fig. 1).

The following explanations will provide more details about the scope of this task, taking as an example the procedure utilized for the WS TOR-NADO.

2. Concept for Aircraft Tracking

Fatigue Life Monitoring is not used exclusively for events related to flight safety, but is used increasingly as a cost-effective means of material conservation. The value of this monitoring system thus varies by the ratio of its performance, including such aspects as

- number and importance of FCAs
- precision of damage calculation per FCA
- scope of application for all FCAs of the A/C structure
- its costs, which include investment costs for data acquisition and processing systems

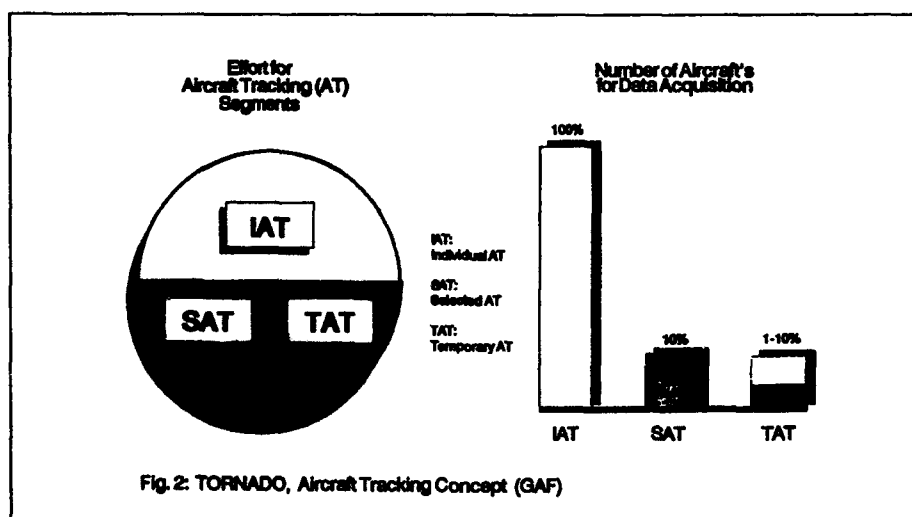
- running costs for data evaluation and report preparation

Taking the point of view that a number of FCAs must be monitored rather than the entire A/C, the question of optimisation of the procedure must be related to the individual FCAs themselves. The complexity of present-day weapon systems demands a wide-range tracking concept, which can also react as flexibly as possible to the problems of material preservation.

2.1 Tracking Elements (Fig. 2)

The tracking concept of the WS TOR-NADO is divided into the three sectors :

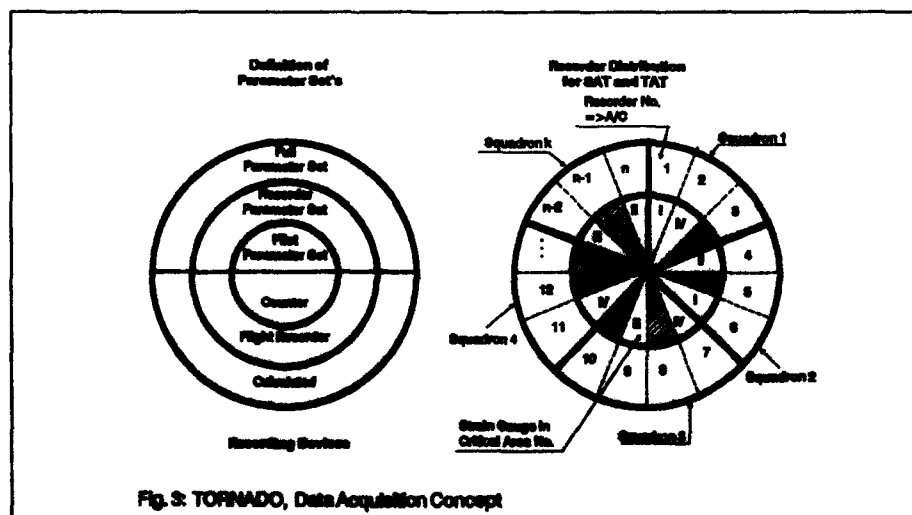
- Individual Aircraft Tracking
 - Selected Aircraft Tracking
 - Temporary Aircraft Tracking
- Monitoring is based essentially on flight parameters, which are available on the existing data acquisition unit of the crash recorder. Those parameters relevant to structural monitoring are collected in the RPS and registered by a flight recorder. The FPS is an extension of the RPS and is generated through differentiations and conversions of several parameters. It contains all essential influencing variables for mechanical strain (e.g.

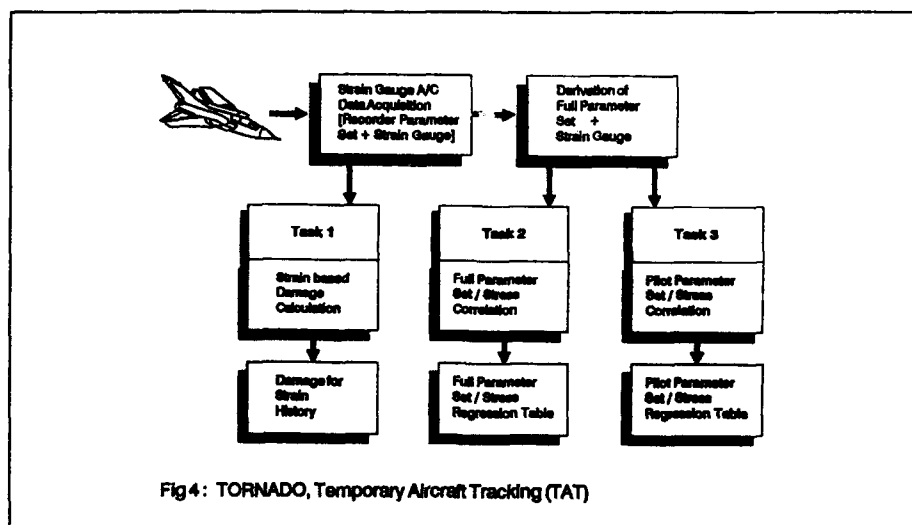


accelerations, weight ...). To reduce the expense of data acquisition for each individual A/C, an additional severely reduced PPS was defined, containing only the essential parameters for stress on the most vital FCAs. This IAT parameter set is registered by means of a counter. (Fig. 3).

The flight recorders are distributed on a statistically representative basis throughout the individual squadrons and register the flight parameter spectrum of selected aircraft. The key monitoring elements are, however, the strain gauges in the various FCAs. Due to overall

sampling rate restrictions only one FCA can be monitored in addition to the flight parameters for each aircraft. The strain gauges are evaluated by regression techniques to produce a realistic correlation between operational strain on the structure and the flight parameters that cause it. As the strain/parameter correlation is only partially deterministic because of the limited parameter sets, it is cyclically repeated within the TAT segment for the same FCA on several A/C. Hence, mismatches in the parameter algorithms, due to alterations in the A/C configurations and in the operational missions are covered simultaneously.





2.2 Temporary Aircraft Tracking (Fig. 4)

TAT draws on strain gauges, which are recorded together with the RPS in order to achieve simultaneous acquisition. The RPS is then extended to the FPS (Para. 2.1).

Task 1:

>Strain Based Damage Calculation<
Direct conversion of the measured strain history into a frequency of occurrence matrix and calculation of the accumulated damage. The validity is restricted to the A/C observed throughout the measuring period.

Task 2:

>FPS/Stress Correlation<
Correlation of the FPS with the stress at FCAs obtained from local strain measurement. The regression extends to a representative number of flights (>100) for a specific A/C configuration and is repeated cyclically.

Task 3:

>PPS/Stress Correlation<
Correlation of the Pilot Parameters utilized for IAT with stress at FCAs, calculated from the measured local strain. The regression extends, as in Task 2, over a representative sample.

An example of Task 1 of current interest is the acquisition of data on operational strain on the components of the aircraft, which so far have shown no defects in the fatigue test. In contrast to the

areas monitored, in this case it remains uncertain, after completion of the fatigue test, whether the loads simulated in the test are actually representative of operational loads. The analysis is based on the fatigue life of the complete subassembly and not that of a single, local FCA.

2.3 Selected Aircraft Tracking (Fig. 5)

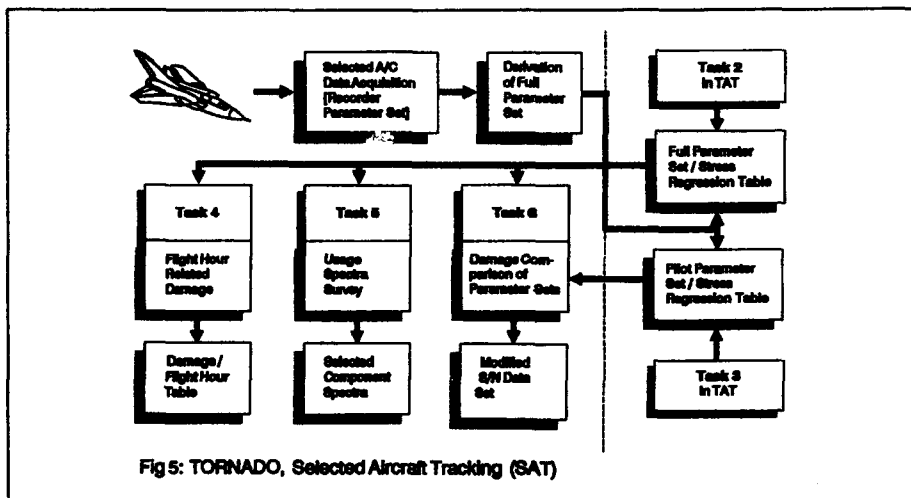
Derivation of the FPS takes place analog to TAT, although, unlike TAT, the recordings do not contain strain gauge parameters. The generation of the appropriate stress patterns for all FCAs is carried out via the FPS/Stress Regression Table (Task 2 in TAT).

Task 4:

>Flight Hour-Related Damage<
Conversion of the strain history calculated into a frequency of occurrence matrix and based on the calculation of accumulated damage. The fixation of the safe damage rate per flight hour is calculated on a statistical basis for each squadron.

- Task 5:

>Usage Spectra Survey<
The frequency of occurrence matrices obtained analog to Task 4 for selected parameters and stresses are stored as a basis for the evaluation of further correlated components and structural areas.



-Task 6:

>Damage Comparison of Parameter Sets<

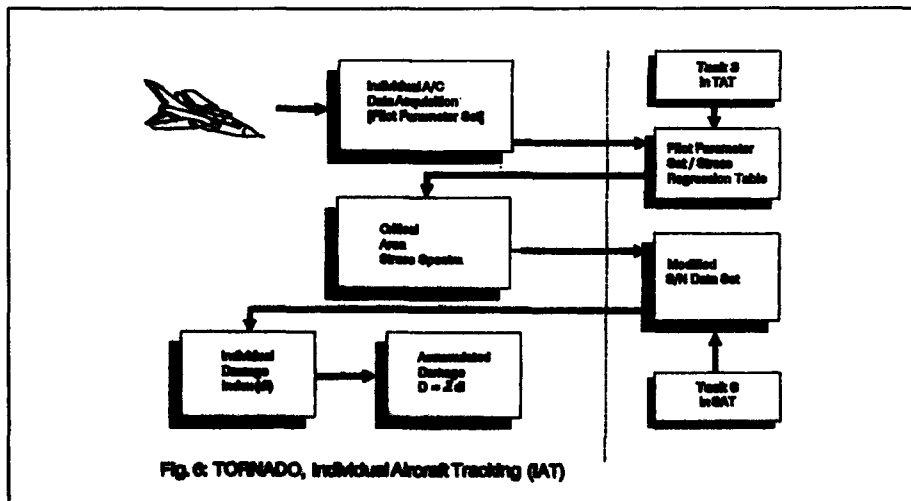
The first step simulates IAT, where a stress spectrum is generated with the FPS/Stress Regression Table. In the second step for the same data sequence a stress frequency of occurrence matrix is generated with the FPS/Stress Regression Table. In the third step the resulting damages are compared at individual stress levels. Out of that procedure the modified S/N data set is generated in order to make the FPS knowledge available for the individual aircraft.

Task 4 corresponds to the classic form of SAT, i.e. if the A/C cannot

be individually monitored with reference to a specific FCA. This necessarily occurs when the IAT pilot parameters are blind to the stress variation of the FCA. In this case the scatter of damage is lower, when related to the total flight hours compared to the damage calculated with the pilot parameters.

2.4 Individual Aircraft Tracking (Fig. 6)

From the conceptual point of view, the IAT permits optimum utilization of the inherent structural life. This naturally presupposes that



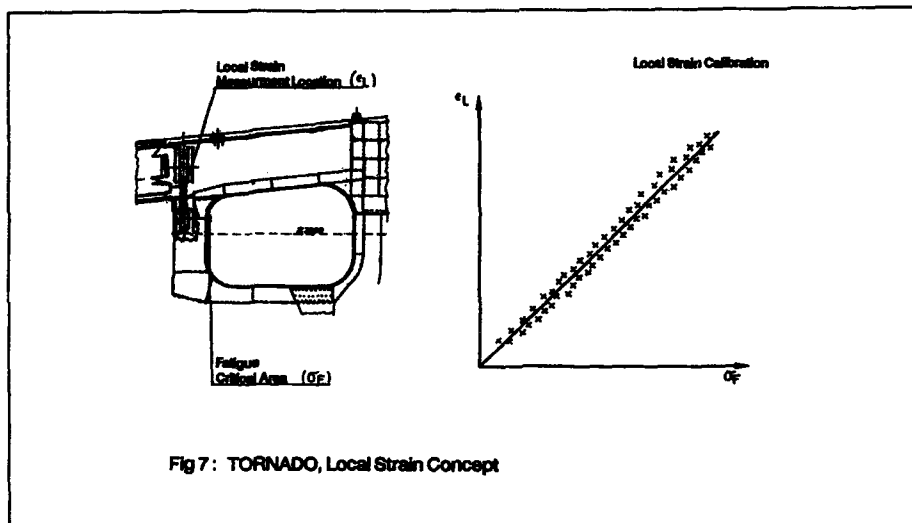


Fig 7: TORNADO, Local Strain Concept

appropriate sensors (pilot parameters) exist in the individual A/C for the acquisition of stress data. The pilot parameter signals acquired by IAT are converted via the appropriate stress regression table (Task 3 in TAT) into FCA stress spectra and transformed into the individual damage index via the modified S/N data set (Task 6 in SAT). The accumulated damage (D) is the sum of the individual damage indices (d_i).

3. Data Acquisition Concept

3.1 Transducer

The wide range of problems of material preservation in a modern weapon system requires transducers for the precise acquisition of stress progression at any structural point. In general, the two types of transducers, Strain Gauge and Flight Parameter, have been proved to be applicable.

In the WS TORNADO, both types of transducer are installed side by side.

- a) Strain gauges
 - Within a cost-effective overall concept the strain gauges are essentially only used for, regression to the flight parameters.
- b) Flight parameter
 - Corresponding to the monitoring concept the two data sets
 - RPS for SAT
 - and
 - PPS for IAT

are defined for the acquisition of flight parameters.

3.1.1 Strain Gauge for TAT

The process of application of strain gauges can be divided into the following two groups:

- a) Calibrated Loads
 - This evaluation method requires a series of calibration tests to produce the coefficients for the determination of the loads from strain gauge outputs. Because of the extent of the calibration, this procedure is only used with TORNADO in isolated cases, e.g. load measurements in control rods.
- b) Local Strain
 - Effective use of these transducers require prior in-depth knowledge of the strain distribution around the aircraft fatigue critical locations. This is necessary to define a suitable measurement location from which the strain can be scaled to represent the stress at the fatigue critical location.

Monitoring of the TORNADO's FCAs is based essentially on the local strain concept as key element. For this a suitable local strain measurement location was qualified for every FCA within the Major Aircraft Fatigue Test (MAFT). This means, the algorithm between stress in FCAs and local strain at the measurement location has been derived (Fig. 7).

No.	Parameter	Sampling Rate / s	No.	Parameter	Sampling Rate / s
1	Pressure Altitude	0.5	11	Inboard Spoiler STBD	1.0
2	Calibrated Airspeed	0.5	12	Rudder Position	2.0
3	Normal Acceleration	18.0	13	Wing Sweep Angle	0.5
4	True Angle Of Attack	2.0	14	Primary Strain Gauge	18.0
5	Roll Rate	8.0	15	Secondary Strain Gauge	4.0
6	Pitch Rate	4.0	16	Flap Position	1.0
7	Yaw Rate	2.0	17	Slat Position	1.0
8	Taileron Pos. PT	4.0	18	Fuel Remaining	1.0
9	Taileron Pos. STBD	4.0	19	Stores Configuration	4.0
10	Outboard Spoiler PT	1.0	20	Cleo Switch	0.5
			21	Identification Data	1.0

Fig. 8: TORNADO, Recorder Parameter Set and Strain Gauges

This algorithm enables as directly as possible a comparison of critical area stresses between the test article and the service aircraft. The local strain measurement location was selected with particular attention to its accessibility on the aircraft in operation.

In accordance with conceptual requirements, only selected A/C are equipped with strain gauges, in order to correlate the local strain with the flight parameters. Based on that, a strain gauged aircraft is dedicated to a specific FCA and due to the symmetric design of the aircraft the left and right hand FCA is equipped with a strain gauge. This symmetrical strain recording enables a scatter evaluation and on-line data checks. For the strain gauge data acquisition 2 strain channels are incorporated in the RPS. This allows for simultaneous recording of strain and parameter (Fig. 8).

a) Primary strain gauge

This signal is evaluated for the parameter/stress regression. It is sampled at a rate of 16 per second.

b) Secondary strain gauge

This signal is used for scatter and drift analyses and sampled at a rate of 4 per second.

For high frequency analyses in Task 1 an additional measurement computer may be fitted, which allows for sampling rates up to 500 samples per second on both channels.

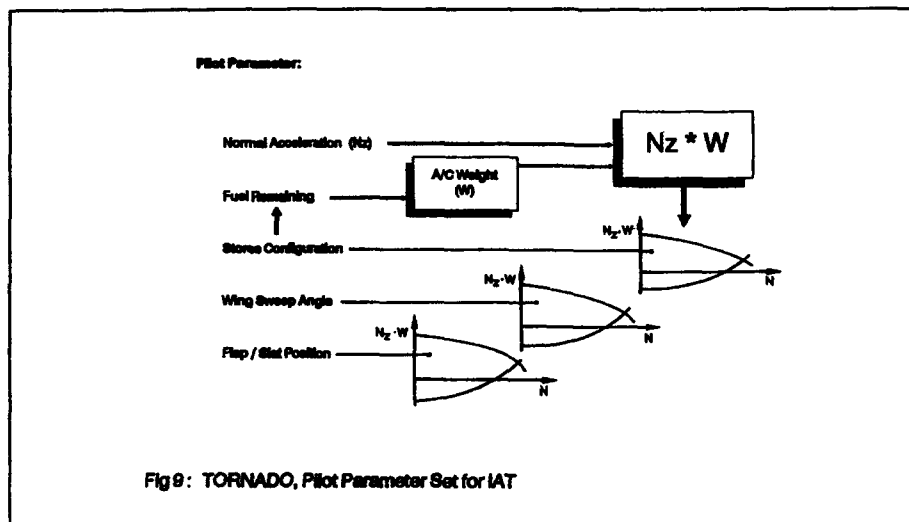
3.1.2 Recorder Parameter Set

The flight recorders are distributed according to statistical considerations throughout the entire fleet and therefore permit a representative overview of the A/Cs' operational usage. In calculating critical area stresses based on the recorder parameters particular care must be taken that:

- a) all parameters, which influence the stress of the FCA in question are included in the parameter set, and
- b) an adequate sampling rate has been provided
- c) a suitable concept for conversion of parameter signals to stress histories exists.

The latter demand is fulfilled within the FPS/Stress Correlation (Task 2 in TAT), which simultaneously indicates, whether any flight parameters essential in the calculation of stress are lacking. Should this be the case, only a strain-based damage calculation (Task 1 in TAT) is possible.

Since for cost reasons, not every aircraft will be equipped with a Flight Recorder, individual aircraft



monitoring takes place on the basis of a limited PPS.

3.1.3 Pilot Parameter Set for IAT (Fig. 9)

Because of the position on the center fuselage and inner wing of the main FCAs in the TORNADO (Ref. 1), $N_z \cdot W$ was defined as the pilot parameter for individual A/C.

From the flight parameters

- Stores Configuration and
- Fuel Remaining

the A/C Weight (W) is continuously calculated and multiplied by the

- Normal Acceleration (N_z)

to give a continuous $N_z \cdot W$ history. The range-filtered history is then counted for the number of level crossings of pre-defined $N_z \cdot W$ levels.

The counting procedure is that of Peak and Trough with the additional feature that at the point of level crossing, the flight parameter values

- Stores Configuration
- Wing Sweep Angle and
- Flap/Slat Position

are also acquired. The reading is then stored according to the constellation of these parameters.

The correlation of the pilot parameters with the FCA stress takes place within the PPS/stress correlation (Task 3 in TAT). If an inadequate correlation is present, the critical area is not monitored within IAT. In that case flight

hour-related monitoring takes place (Task 4 in SAT).

3.2 Recorder and Data Transfer (Fig. 10)

3.2.1 Flight Recorder for SAT and TAT

In SAT and TAT data registration a digital recorder with magnetic tape cassette is used. The parameter and strain gauge signals are recorded continuously, and only periods without any significant change in signal are condensed by reference to parameters. Thus, for following evaluations the complete signal content is available.

One cassette has adequate capacity for data from 15 to 20 flight hours. The recorded cassettes are sent directly by the squadron to the structural life evaluation centre.

3.2.2 Counter for IAT

The onboard data processing for IAT takes place within the Data Acquisition Unit (DAU). The $N_z \cdot W$ spectra collected in the DAU are transferred each working day to the OLMOS Ground Station (OGS) by means of a Hand Held Terminal (HHT) and stored temporarily related to individual aircraft.

At predetermined intervals, the data are transferred to the central computer establishment via direct data link. There the data are

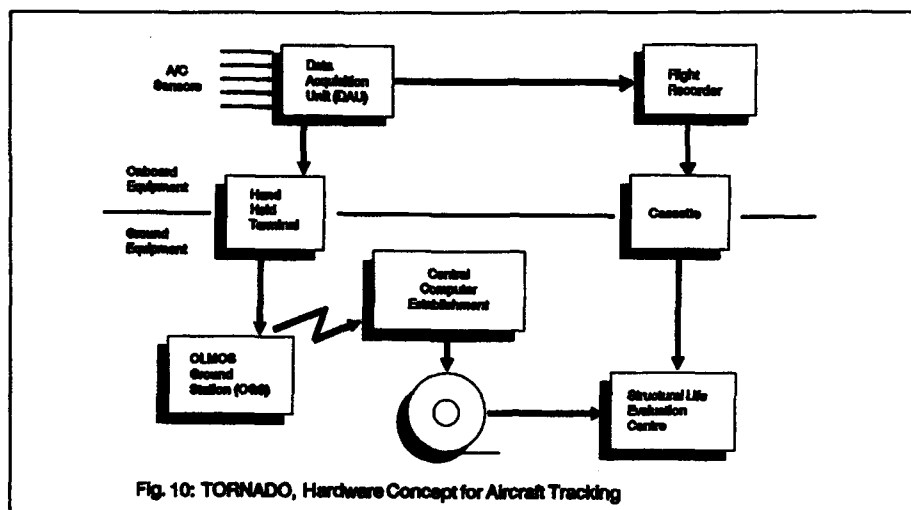


Fig. 10: TORNADO, Hardware Concept for Aircraft Tracking

assembled on the main frame computer and sent every month on magnetic tape to the structural life evaluation centre.

Because of the extensive data acquisition involved in IAT, the procedure described above was optimised for efficient data transfer. Moreover, high-quality data are obtained by means of the transfer's full adaption to electronic data processing, i.e. errors caused by manual interference are eliminated.

4. Conclusions

For the WS TORNADO, the multi-level process of aircraft tracking, consisting of the elements

- Temporary Aircraft Tracking
- Selected Aircraft Tracking
- Individual Aircraft Tracking

has proved its worth as a cost-effective method. Errors are revealed by means of cyclical repetition of correlations between strain and parameters in the routine operation.

Within the IAT system, the electronic data processing compatible data transfer from A/C to the structural life evaluation centre has also achieved good results. Apart from the capacity of processing a much larger quantity of data, major progress in data quality against that of manual data processing was achieved.

For future aircraft tracking the TORNADO experience has shown that

- a) a sampling rate of 16 per second is a minimum requirement for operational strain measurements at the center fuselage and the inner wing

- b) a sufficient measurement at the outer wing, fin, taileron, control surface, or landing gear requires a higher sampling rate depending on the location
- c) parametric equations for the individual monitoring (IAT) of structural areas listed under b) may be difficult to define
- d) the data acquisition for IAT can be further optimized by on-board data processing.

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AIRCRAFT TRACKING OPTIMIZATION OF PARAMETERS SELECTION

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SUMMARY

Any appropriate structural maintenance plan, based on a Safe-Life or a Damage Tolerance concept, is in close relation with the accuracy of in-service loads identification.

The development of multichannel devices with integrated capabilities for in-flight pre-processing allows to get in-service data and flight results about any critical part of the equipped aircraft, but with sometimes such an amount of computations that the "should-be simple" in flight processing has to be transferred toward a ground facility.

In order to reduce the volume of data for calculations (and their cost of acquisition), we studied the relative influence of the various flight parameters considered during the static and fatigue design of an aircraft.

The calculated stresses and loads have been compared to their very same counterparts measured in flight. Their effects on fatigue values have been quantified. This study has been performed for the MIRAGE 2000 aircraft tracking and in service loads identification.

1 - THE AIMS

The purpose of the study is to define a principle for the measurement of loadings acting on primary structural elements using a limited number of flight parameters ; from this it will be possible to :

- detect possible overstress incidents, with a view to carrying out post-flight checks,
- replan maintenance programmes to improve flight safety and ensure the longest possible structural life,
- build up statistics of actual operational profiles to confirm or up-date the database used for calculation.

Research into in-flight loads must therefore :

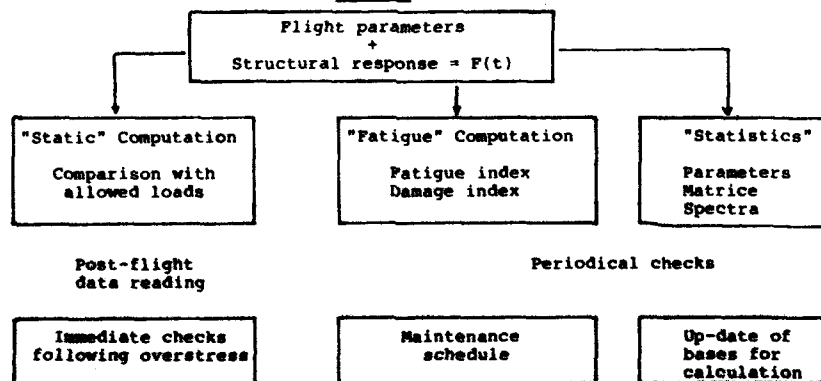
- reveal any local or general overshooting of theoretical limits ; great accuracy is not needed, but results must be reliable,
- record any subsequent load peaks (Fmin, Fmax) with the greatest accuracy achievable within reason, to enable a damage computation (crack initiation and propagation).

In-flight pre-processing of this data requires :

- continuous recording of flight parameters and derivatives for study and possible analysis before the next flight,
- storage of general statistical data (Markov Matrices, fatigue spectra) for regular recovery of information.

OBJECTIVES FOR IN-SERVICE USE

TABLE 1



2 - MODELLING OF STRUCTURAL RESPONSE

2.1 - The structural response computation model

For each manoeuvre, identified from aircraft control surface deflections and attitudes, flight mechanics equations can be used to balance aerodynamic effects against inertia forces.

Using a finite element modelling technique, local structural stress response is deduced for unitary aerodynamic and mass effects leading to the development of a response grid $\sigma/F/q$ for different Mach and altitude flight conditions and aircraft configurations (see Fig. 1).

The theoretical local response $F(t)$ in a given manoeuvre is obtained, with interpolation within the unitary grid, by linear combination using the parameters characterising the manoeuvre.

In-flight calculation of discrete results, for the updating of theoretical calculated aerodynamic effects, is achieved using linear regression techniques during manoeuvres specific to the area being studied.

The degree of correlation obtained in each area tested is confirmed by comparison with measurements taken in flight (see figure 2).

2.2 - Flight Control System Capabilities

The Mirage 2000 has electrically signalled controls whose transfer functions :

pilot inputs $\longleftrightarrow$ control deflections

ensure full control to the aircraft throughout the flight envelope.

The Electric Flight Control System functions have been specified to ensure that calculated structural limits are not exceeded (*), without any degradation of performance of handling qualities, by the definition of, in particular, maximum control surface deflections at all points of the envelope (see fig. 3).

(*) the system is designed for frequent excursions to the normal "g" limit (9g), but the pilot can exceed this (to 10.5g) by applying extra force to the control stick up over the control elastic limit stop.

2.3 - Structure sizing

The presented model permits calculation of structural response :

- for static strength, using unusual "step" type manoeuvres as required by design standards (see fig. 4) or higher loadings where demanded in the contract (in sideslip for example),
- for fatigue resistance, starting from planned operational use conditions : standard manoeuvres are repeated in different flight configurations.

The dominant criterium (static or fatigue) varies between different parts of the structure. This is illustrated in Fig. 5a for the main sizing elements.

Comparative results are shown hereunder in Table 2 for the main structural components in terms of structural margins.

TABLE 2

Stress/Load	STATIC		FATIGUE
	Input	Margin (1)	Margin (2)
Wing root			
- Main attachment	pull-up + roll	10 %	(**) 0 %
- Rear attachment	reversed pull-up	30 %	100 %
1/2 span			
- Undersurface skin	pull-up + roll	30 %	10 %
Fin root			
- Main attachment	reversed roll manoeuvre	50 %	100 %
- Rear fuselage	sideslip/lateral gust	30 %	100 %

Margins are established using comparison between :

- (1) Sizing stress and maximum stress expected in flight ($\times 1.5$ at extreme load)
- (2) Sizing stress allowed per g and in flight calculated stress per g

(**) Critical zone which can be inspected and repaired.

3 - ANALYSIS OF MANOEUVRES

All flight manoeuvres are considered to result from pitch, roll and yaw inputs, and their respective trim forces.

3.1 - The aircraft in balance

The principles used are described below :

- in pitch : normal acceleration n_z results from two effects :
 - stabilized pull-up (n_{α}) : this normal acceleration corresponds to the trimmed lift effect resulting from aircraft incidence (at a given Mach/altitude and loading/aerodynamic configuration)
 - dynamic pull-up ($n_{\dot{m}}$) : this normal acceleration results from the instantaneous deflection of the elevons, balanced by inertia forces, mainly q (pitch acceleration).
- in roll : asymmetric deflection of the elevons produces a rolling moment which is balanced :
 - at roll initiation ($p = 0$), by inertia forces proportional to p (roll acceleration),
 - at a stable roll rate ($\dot{p} = 0$) by aerodynamic forces proportional to p (roll rate).
- in yaw : lateral acceleration (n_y) results from sideslip effect balanced by a rudder deflection. Yaw equilibrium is achieved through the balance of two effects :

sideslip
rudder deflection

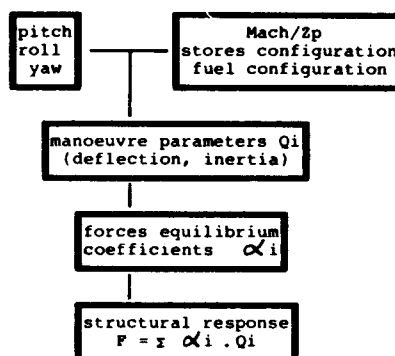
balanced by inertia n_y , $\dot{p}$, $\dot{r}$

Note : Asymmetric elevon deflection also produces aerodynamic effects on the fin whose intensity is significantly depending with aircraft incidence. These effects are compensated by a rudder deflection (Electric Flight Control System function).

3.2 - Linked effects

The use of this type of basic balanced manoeuvre has the advantage of integrating coupling effects (such as roll induced yaw) and the non-linearities of certain values (such as the shift of the wing centre of lift with incidence).

These basic inputs are defined by calculation of inertial and aerodynamic forces equilibrium coefficients. If one of them is known, the others can be deduced proportionately.



When applied to the structural model, the stress responses to each basic balanced input take account of the associated flight parameters described in the Table 3.

Notations : see at the bottom of figure 1.

TABLE 3

Basic Input	Aerodynamic effect (at given Mach/ $\dot{z}$)	Counteracted by : - Inertia - Aerodynamics
<u>Pull-up n_z</u> - stabilized pull-up (n_{α})	α (incidence)	$\dot{m}_m^{n_a}$ (pitch)
- dynamic pull-up ($n_{\dot{m}}$)	$\dot{m}_2 = \dot{m} - \dot{m}_1$	$\dot{q}$, $n_{\dot{m}}$
<u>Roll</u> - roll initiation	$\dot{m}_1$ (elevons)	$\dot{p}$ ($p = 0$) ($\dot{m}$ rudder deflection)
- stabilized roll	$\dot{m}_1$	p ($\dot{p} = 0$) ($\dot{m}$ rudder deflection)
<u>Yaw</u> - sideslip	s	$\dot{m}_n = 0$ n_y $s = 0$ $\dot{p}$ $\dot{r}$
- rudder deflection	$\dot{m}_n$	

3.3 - Structural Loads and Stresses

The study is carried out using stresses generated by loadings representative of the behaviour of major structural zones, with inputs in pitch, roll and/or yaw. These stresses, limited in number, (see fig. 5b) enable the analysis of :

- the vital wing/fuselage and fin/fuselage attachments,
- structural zones identified as "critical" using strength margin criteria.

The Table 4 reveals the influence of different starting conditions (α , δm , δl , δ , δn) and inputs on structural stress levels. The effect of an angle change 0.1° is estimated at determinant envelope analysis points as percentage of corresponding limit stress (see fig. 6).

4 - FLIGHT PARAMETERS FOR COMPUTATION

4.1 - Basic Parameters

These are those from which can be derived, at any given moment :

- the flight point (Mach number, altitude)
- aircraft mass (forces and moments of inertia).

They are obtained from :

- static pressure
- dynamic pressure
- fuel used and external stores.

4.2 - Principles adopted for the selection of manoeuvre parameters

Theoretical calculations give a grid of structural responses to unit pitch, roll and yaw inputs as a function of flight conditions and aircraft mass and CG.

Coefficients for inputs involved in the manoeuvre are defined starting with a root parameter (see Table 3) for each unit input and followed by interpolation using the structural response grid.

The choice of these main basic parameters depends on the ease of measurement : conditions imposed (usually through cost considerations) are as follows :

- no sensors dedicated to load measurement (is no additional incidence probe, yaw vane or roll rate gyro),
- no access to the information on the data bus (most data is obtained by derivation from the input to the crash recorder).

The choice of root parameter for the definition of the manoeuvre balance will depend on the relative accuracy (and cost) of recording the various parameters and will be either aerodynamic (control position...) or inertial (normal acceleration...) as shown in Table 4.

Validation of the results obtained, to meet the aims set out in paragraph 1, is carried out by correlation against values obtained from theoretical calculations or from measurement.

TABLE 4

Structural response to a 0.1° angle change

% of Limit Load	WING (M = 0.9/Z = 0)		FIN (M = 0.9/30000 ft)		
	Pitch		Roll	Yaw/Roll	
	α	δm	δl	δ	δn
Wing Root					
Bending moment	1.5 %	0.2 %	0.1 %	-	-
Rear attachment load	-	1.4 %	0.1 %		
1/2 Span (Stresses)					
LP1 x direction	-	0.7 %			
y direction	1.6 %	-	-	-	-
LP2 x direction	-	-	1.0 %		
y direction	1.8 %	1.3 %	1.0 %		
Fin bending			0.4 %	0.5 %	0.2 %
Manoeuvre balance					
Primary effect	n_z (0.15 g)	$\dot{q}$ (0.2 rad/s ²)	$\dot{p}$ (0.3 rad/s ²)	n_y	n_y
Secondary effect	δm	$n \delta m$ (= 0.01)	$p = 0$	$\dot{r}$	$\dot{r}$

4.3 - Parameters selected and correlations

Parameters selected, taking into account the notes above, are presented in Table 5. They involve some approximations whose influence on the results is quantified using ratio "C", here called the correlation ratio :

"C" = results obtained/results expected.

4.3.1 - "Static" results (Detection of limits being exceeded)

The study is carried out starting with the theoretical structural response to critical inputs for static strength sizing.

The correlation ratio is :

$C = \frac{\text{computed load}}{\text{forecast theoretical load}}$

TABLE 5

INPUT		CORRELATION "C"
Stabilized pull-up	nz	
- Wing root bending moment $\frac{1}{2}$ span (LP1) : y direction		1.0 <C< 1.05
		0.9 <C< 1.1
Dynamic pull-up	δm	(*)
Pull-up + roll	nz + δl	
Wing - 2 C31 attachment - $\frac{1}{2}$ span (LP2) : x direction		1.0 <C< 1.1
		1.0 <C< 1.25
Yaw + Roll	δl	
Fin root (**)	ny, δn	0.75 <C< 1.25

(*) When the transfer functions provided by the Electrically Flight Control System are taken into account, simulations of reversed pull-up manoeuvres show that the reverse control deflections are significantly less than was initially assumed when sizing. The limit case becomes normal "g" (with roll).

(**) As mentioned in para 3.2, effects on fin and rudder as significantly influenced by incidence. The 25 % correlation error results from a simplifying linearisation of these effects at average values. As the loads obtained in flight do not exceed 50 % of the calculated limit figures, the effects were not examined in great detail in flight.

Comparative results for the wing root, in proportion to the magnitude of the selected parameters, are presented in the diagram at Figure 6.

This diagram shows the main representative parameters concerning the studied areas through the loading response magnitude due to each type of manoeuvre.

4.3.2 - Fatigue results (Damage assessment)

The study is run using results from a series of about 50 flights with recording of all relevant parameters (see Table 3) and direct measurement of local stress levels, representative of structural loadings.

The correlation ratio is estimated in terms of calculated damage, using the Miner's law :

- A) starting with the local stress measured in flight
- B) starting with the theoretical calculated stress
- C) starting from the parameter selection (see Table 5)

Results obtained at design critical points (for the wing these are the root bending moment and skin stress at half span) are as follows :

TABLE 6

PARAMETERS	CALCULATION		
	(A)	(B)	(C)
Stress measured in flight	X	-	-
Mach/Zp Mass values + external loads		X	X
nz		X	X
Incidence		X	-
Pitch angle δm		X	X
Roll angle δl		X	X
Roll rate q, δ , δn , ny		X	-
Wing root (Mx)	0.92	1.0	0.94
$\frac{1}{2}$ span (σ_y)	-	1.0 (Ref.)	1.04
DAMAGE CORRELATION			

5 - CONCLUSIONS - DATA COLLECTION IN FLIGHT

Follow-up of structural damage on Mirage 2000 aircraft is carried out starting with the matching of loads recorded in flight to critical structural components.

Setting-up difficulties for data collection and flight parameter processing necessitated an in-depth preliminary study with the aim of reducing the number of useable parameters without losing sight of the aims of the exercise (detection of incidents where calculated stress limits were exceeded, assessment of local damage for the maintenance programme).

The techniques adopted, and their main results correlated in flight, have shown that these aims have been satisfactorily achieved, when taking in account the structural sizing margins, using the following five analogue and one digital parameters :

Mach Altitude Normal acceleration + 1 Pitch attitude Roll attitude	fuel consumption
--	---------------------

Recording of roll rate (p) is essential for the specific study of any structural zone where this parameter defines the sizing (roll rate is usually less constricting than roll acceleration).

Finally, it must be emphasised that any study of the fin needs two further parameters :

Lateral acceleration
 Rudder deflection

If no historic flight test results of non-linear effects on the fin are available, particular those, caused by aircraft incidence the study can be carried out with strain gauges.

While the importance of the structural sizing of the aircraft must be noted when parameter selection criteria are being considered, these principles have been applied, in their entirety, to another delta-winged aircraft without revealing any reason to question the validity of the fundamental results of this study.

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FLOW DIAGRAMM OF LOAD/STRESS CALCULATION

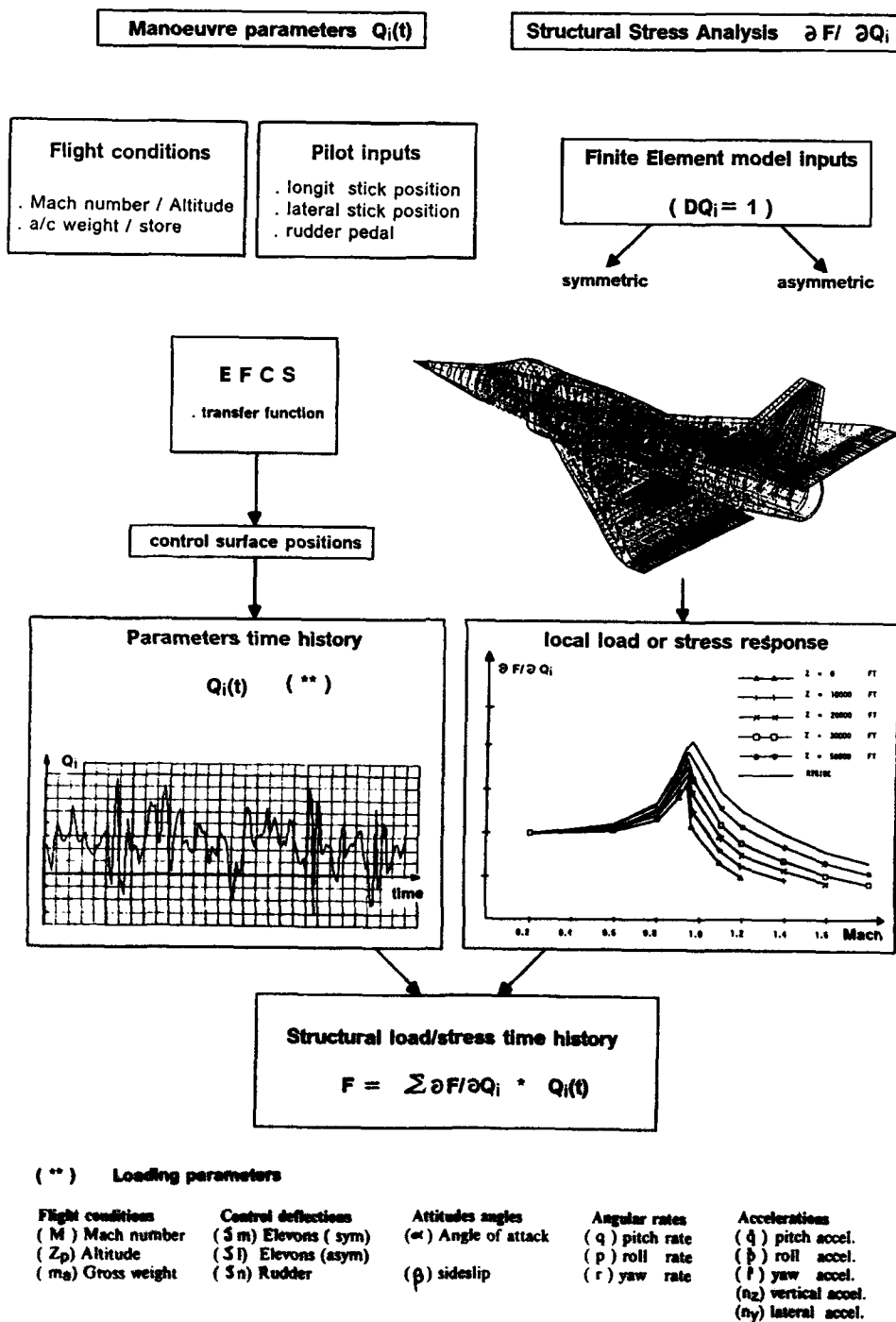


Figure 1

IN FLIGHT ACTUAL LOADS (a/c with external 2000l tank)

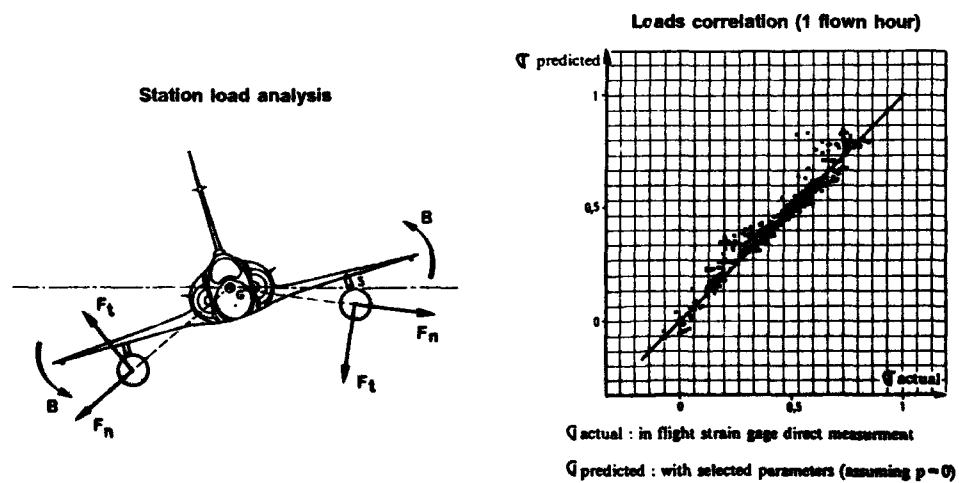
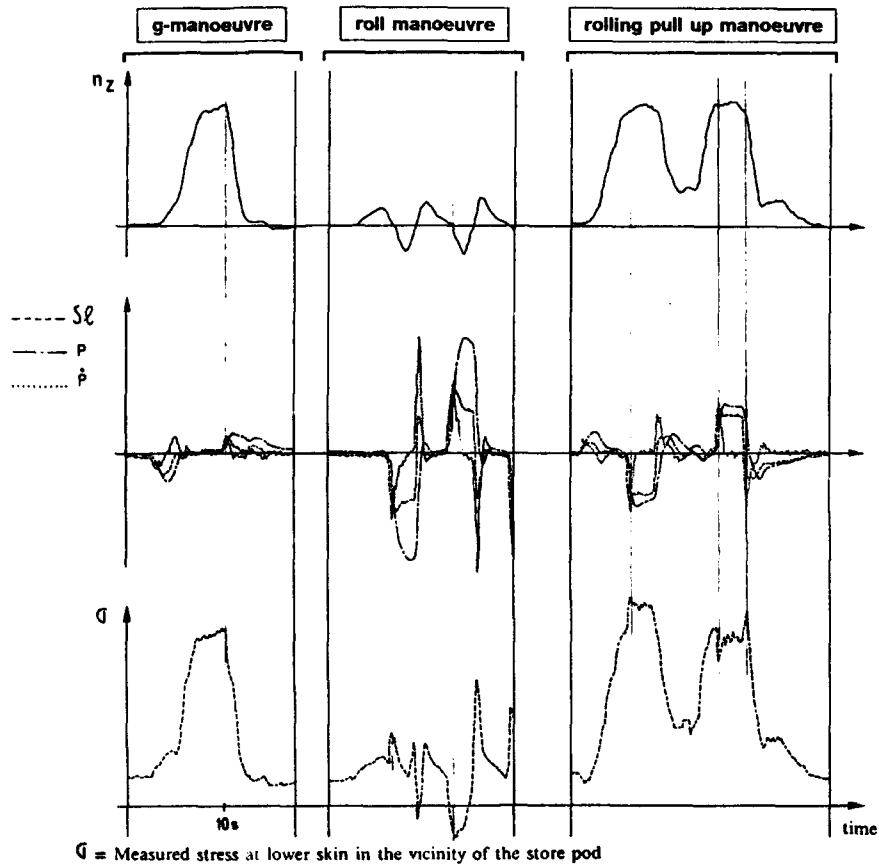
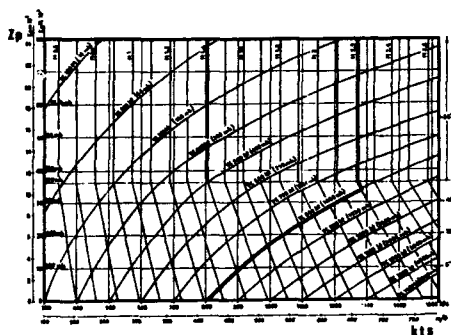


Figure 2

PIC DESIGN CONDITIONS

MACH NUMBER/AIRSPEED

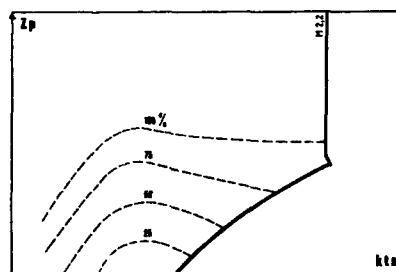
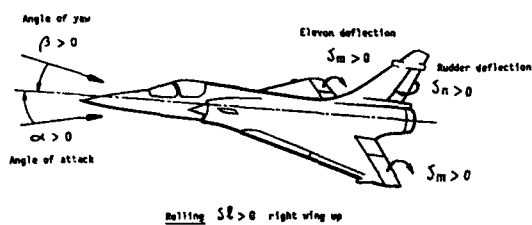
STRENGTH REQUIREMENTS



FRENCH 2004 E REGULATION
and
MIL-A-SPECIFICATIONS

LIMIT LOAD CONDITIONS

NORMALIZED ELEVONS DEFLECTION



NORMALIZED SIDESLIP ANGLE

NORMALIZED RUDDER DEFLECTION

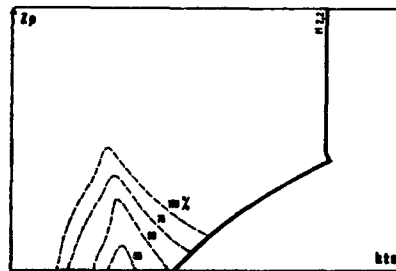
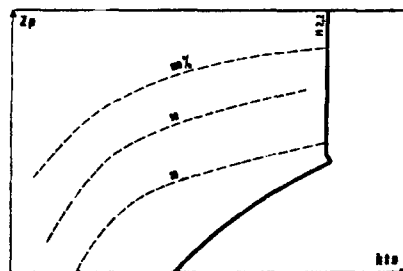
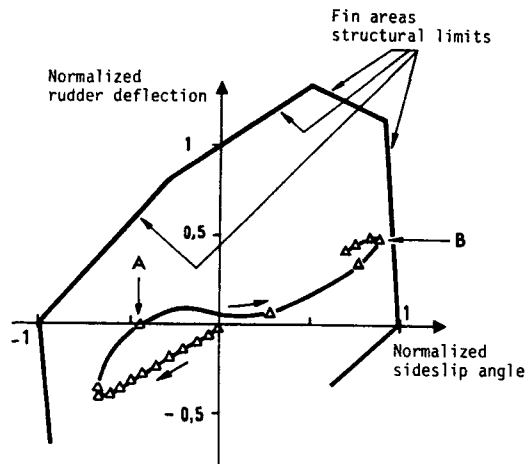
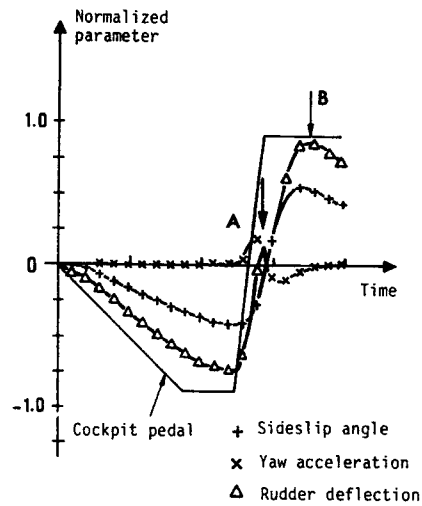
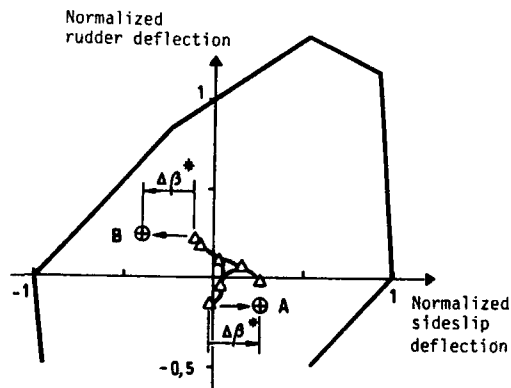
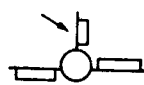


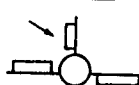
Figure 3

FIN LOAD ANALYSIS**YAW MANOEUVRE****FIN LOAD ANALYSIS****MANOEUVRE DEFLECTIONS**

(Mach 0,9/Z = 0)

**ROLL MANOEUVRE****FIN LOAD ANALYSIS**Instant A

Rear view

Instant B

Rear view

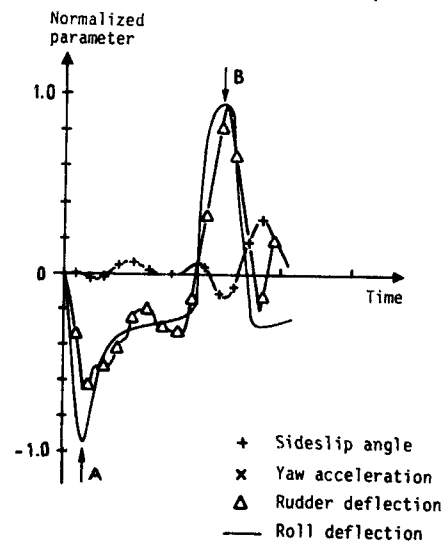
Manoeuvre normalized parameter
(Mach 0,9/Z = 25000 ft)* $\Delta\beta$ Equivalent fin load due to roll effect

Figure 4

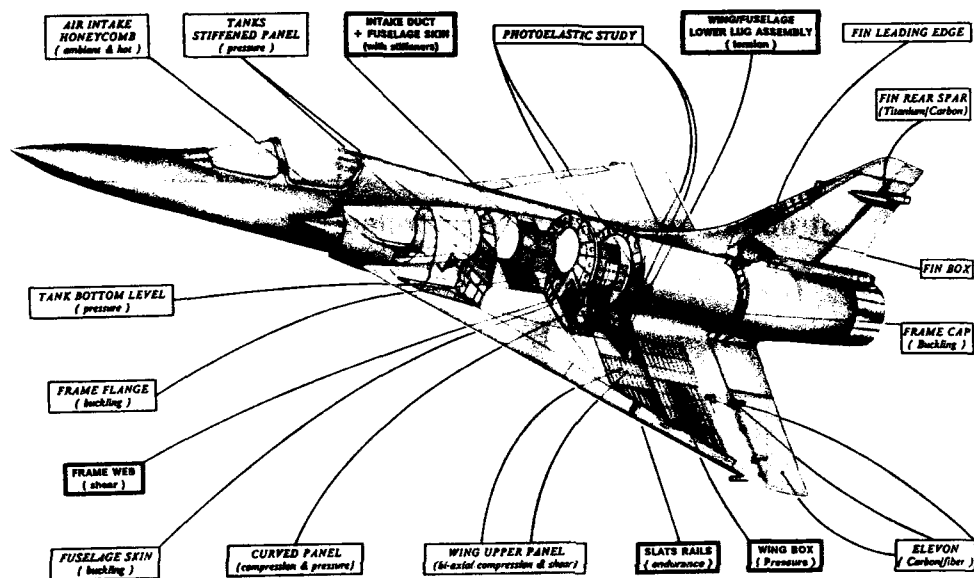
MIRAGE 2000 AIRFRAME

(A)

BASIC STRUCTURAL TESTS

STATIC

FATIGUE



(B)

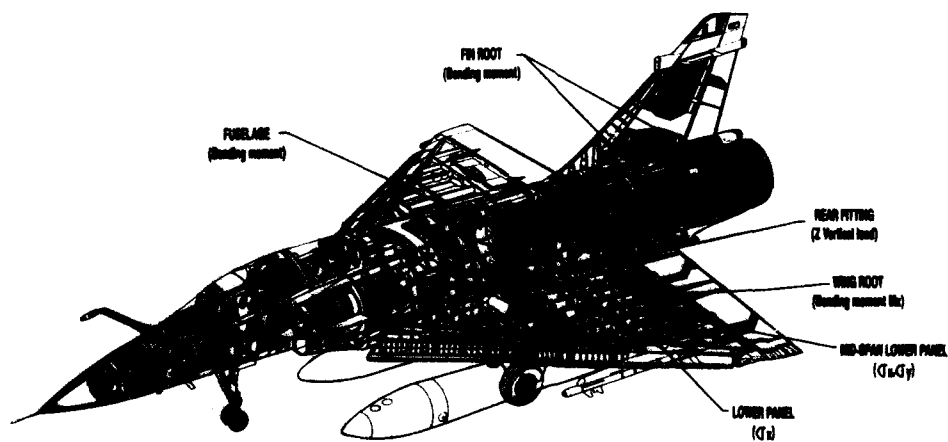
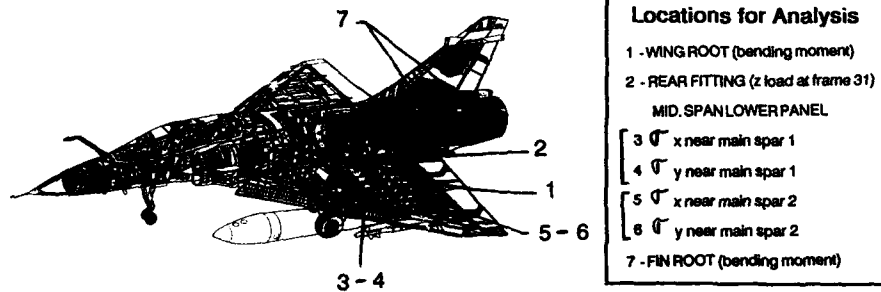
STATION OF LOADS ANALYSIS

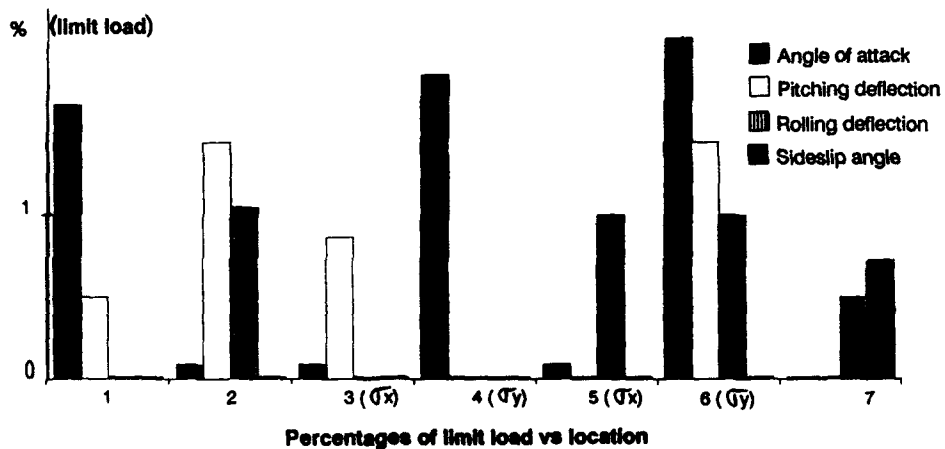
Figure 5

STATIC and FATIGUE ANALYSIS RESULTS

(in service Aircraft monitoring study)



Structural response for 0.1° deflection maneuver.



Damage analysis compared results (station 1)

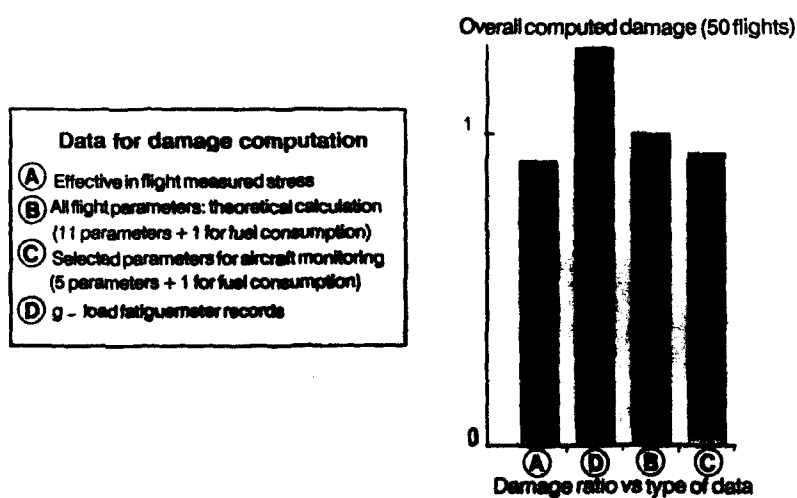


Figure 6

THE OPERATIONAL LOADS MONITORING SYSTEM, OLMS

by

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0. SUMMARY

The preservation of the damage tolerance qualities ensures the safe operation of aircraft lifelong. For this purpose and the assessment of fatigue life consumption different methods have been developed to monitor the service experiences. Nowadays mainly two kinds of monitoring systems which are different in the philosophy exists.

First, the parametric system which records and processes only aircraft mission parameters and second, the direct load and/or stress measurement system using strain gauges. The main advantages and disadvantages of both systems are discussed. A third possibility to monitor the service experiences of aircraft is to combine the advantages of both systems: this leads to the idea of OLMS.

In this contribution the advanced operational loads monitoring system OLMS, for a transport aircraft, is presented. Detailed descriptions are given concerning philosophy and realization.

OLMS represents the on-board equipment for the prespected Airframe Condition Monitoring Procedure (ACMP) which takes care of damage tolerance qualities and which will increase the efficiency of structural inspections. The OLMS as presented in this contribution is adaptable to all transport and combat aircraft with EFCS.

The verification has been performed on the test aircraft by means of strain gauge measurements. In this contribution a detailed description is given. Results of the OLMS-computer simulation program as well as results of the flight test verification are presented.

1. INTRODUCTION

Nowadays there exist mainly two kinds of systems to monitor the service experiences of aircraft for the preservation of the damage tolerance qualities and for the assessment of fatigue life consumptions.

First the parametric systems which only records aircraft mission parameters and second the direct load and/or stress measurement systems using strain gauges. The advantage of parametric systems is that aircraft mission parameters mostly are available from avionic and control systems and therefore no additional transducers are necessary.

The disadvantage is that the derivation of operational loads from aircraft mission data after its processing is difficult or impossible when the parameters are available only in form of cumulative frequency distributions and not in form of load time histories.

The advantage of direct load and/or stress measurement systems is that the load spectra on all components are available and the service experiences of the aircraft are implied direct. The disadvantages are the limited endurance of the strain gauges and the enormous effort in installation, calibration and maintenance during the aircraft service lifelong. A third possibility to monitor the service experiences is to combine the advantages of both systems: this leads to the idea of OLMS [1,2].

New civil transport aircraft see Fig. 1 equipped with Electronic Flight Control Systems (EFCS) allow access to the complete digital information about all its operating conditions. Since load relevant parameters like accelerations, control deflections etc. can be received from sophisticated avionic and control systems it is logical to use this information as input data for load calculations by means of appropriate mathematical models for an operational loads monitoring system. OLMS is such an advanced loads monitoring system and is under development for A320. It will become the basis for an Airframe Condition Monitoring Procedure (ACMP), with the main aim to perform structural inspections on condition, see Fig. 2.

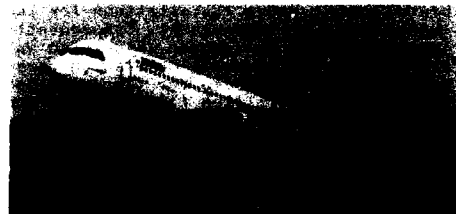


FIG. 1 TEST AIRCRAFT

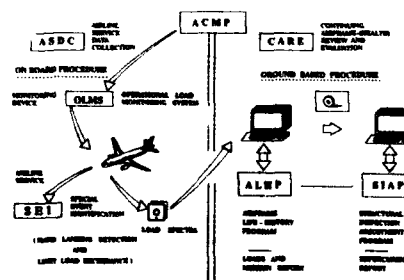


FIG. 2 AIRFRAME CONDITION MONITORING PROCEDURE, ACMP

An essential of every monitoring device to record the in-service experiences are the input data. From the airlines point of view it is not acceptable to have additional sensors and wiring because of extra weight and maintenance. These penalties can only be avoided when taking data already available from sophisticated avionic and control systems.

2. OBJECTIVES OF LOAD MONITORING SYSTEMS

The realisation of OLMS including ACMP will lead to the following advantages:

- An increased efficiency of structural inspections because the inspection thresholds and intervals can be related to the actual loads experienced in service.
- ACMP will settle the problem of ageing aircraft operated beyond the designed life time.
- All investigations required by the airworthiness authorities can be performed more efficiently with increased reliability and more economically
- ACMP will contribute important benefits to maintain the damage tolerance qualities during the whole service life time of the aircraft.

3. ARCHITECTURE AND WORKING PRINCIPLE

3.1 Description

Loads on different components and sections of the A/C see Fig. 3 will be calculated using load relevant parameters and appropriate mathematical models. An important item is that for these sections also calibrated strain gauge installations for load measurements were available. Consequently it was possible to compare calculated and measured loads. In addition mission parameters will be recorded for the description of the mission profiles flown. Fig. 4 shows the loads, parameters and control deflections to be calculated and recorded.

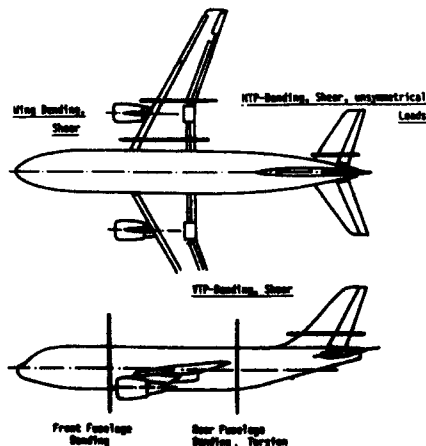


FIG. 3 MAIN COMPONENTS FOR FFI-STRAIN GAUGE MEASUREMENTS AND LOAD CALCULATIONS

No.	Parameter	Loading	Flightphase (Load Fact.)	Correction (Additional)	Remarks
1	Wing	Wing 1/2	Wing 1/2	Wing 1/2	Wing 1/2
2	Footings	Foot 1/2	Foot 1/2	Foot 1/2	Foot 1/2
3	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
4	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
5	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
6	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
7	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
8	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
9	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
10	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
11	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
12	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
13	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
14	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
15	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
16	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
17	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
18	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
19	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
20	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
21	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
22	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
23	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
24	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
25	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2
26	ETP	ETP 1/2	ETP 1/2	ETP 1/2	ETP 1/2

FIG. 4 OLMS CALCULATED LOADS*

3.2 Functions

Four major functions are integrated in the OLMS:

Data acquisition

The handling of the parameters necessary for the load calculation and the determination of mission profile data will be carried out in this function. The parameters are available on the Data Acquisition Module via ARINC 429 system bus with appropriate update rates, see Fig. 5. These data will be transmitted to the OLMS Data Processing Unit to perform load calculations and collection of mission profile data.

SOURCE	PARAMETER
ADRU Air Data Interface Reference Unit	ALT, IND, TAS, AOA, PTO, ROLA, VIB, LATS, VIBS
FACU Flight Management Guidance Computer	SW, CB
PRCSA CONT. Pressure Control System	PRC (loads different pressures)
DNC Data Management Computer	RALT
FQIS Fuel Quantity Indication System	FUEL QUANTITIES WING (inner and outer cell)
FQDC Flight Control Data Computer	STRA, AL, ELEV, AN SPL 1-5
SCDC System Data Control Computer	RQIN
SPCC Wing Flap Control Computer	SLAT/FLAP
FNC Flight Warning Computer	FLIGHT PHASE
FDU Flight Data Interface Unit	TIME, DATE, AC-TAIL NUMBER

FIG. 5 SOURCES OF THE LOAD RELEVANT PARAMETERS

Data processing

In this function loads will be calculated, mission profile data determined and collected. As an appropriate update rate for the load calculation 50 sps has been chosen. Equations of motion have been established containing load relevant parameters connected with aerodynamic coefficients, dynamic pressure etc., see the example for the calculation of the wing shear force on Fig. 6.

The aerodynamic coefficients depending on mach number (M_0) and dynamic pressure (P_0) taken first from wind tunnel tests have been verified by results of FFI-measurements. The coefficients are stored in its own M_0/P_0 -grid. Between the grid points a linear interpolation will be performed.

* A list of symbols appears on p15-14.

MASSES INFLUENCES

$$WSM_1 = - (WM_1 + FUEL_1)(VRTG \cdot G + YWPP \cdot ROLA + XWQP1 \cdot PTCA)$$

AERODYN. INFLUENCES

$$WSA_1 = S \cdot Pd \cdot [B(C8 + C9 \cdot AOA) + (C2 + C3 \cdot AOA + ROLR \frac{LA}{V} (C30)) + AILR(C32) + RHSPL3(C34) + RHSPL5(C36)]$$

$$WS_1 = WSM_1 + WSA_1 + TRANSFERFUNCTION$$

FIG. 6 LOAD EQUATION FOR WING SHEAR RIB 10/11

o Data reduction

The calculated load time histories of a total of 43 different quantities will be analysed in this function by means of the two-parametric statistical counting method "Range Pair Range" [3]. The principle of this counting method is shown on Fig. 7.

o Special functions

Since there are no specific parameters which indicate the "Health of the Structure" directly the following measures were chosen as an indicator for inspections on event:

- Overweight landing
- Hard landing
- Limit load exceedance

It is intended for the serial solution of OLMS to incorporate on request the possibility to trigger special reports after event or landing.

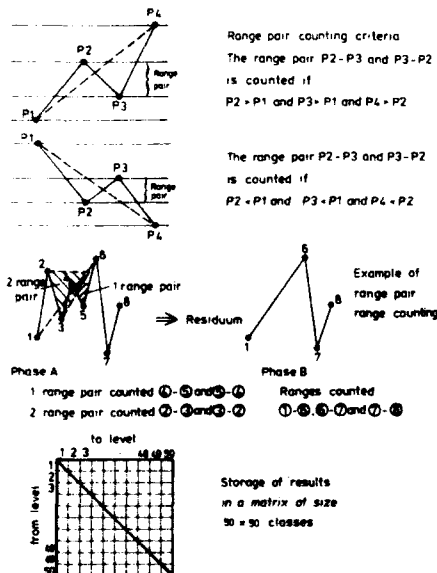


FIG. 7 DESCRIPTION OF COUNTING METHOD

The data acquisition, data processing as well as the data reduction will be performed on-line and in real time during flight. The working principle is shown on Fig. 8, the flow chart of the OLMS processing is represented on Fig. 9.

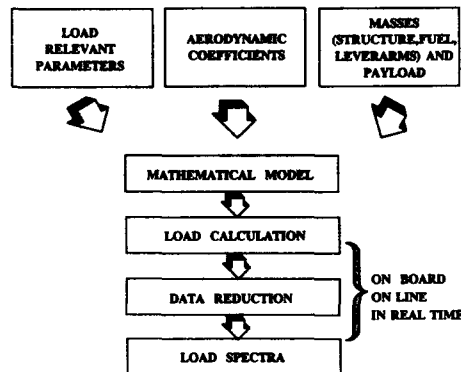


FIG. 8 OLMS WORKING PRINCIPLE

3.3 Hard- and software developments

Due to the fact that for the verification of OLMS during flight tests no extreme weight restriction was required a real time Micro Vax computer including an ARINC 429 Interface Unit has been chosen as hardware tool.

The software development was harmonized to this special computer. All OLMS functions have been programmed in F77 and later translated into PASCAL (necessary for ELN operating system). By means of the whole software tool complete recalculations of loads have been performed in the LAB.

3.4 OLMS - prototype

For development and verification purposes a special prototype has been designed and installed in the test aircraft No. 1, see Fig. 10. It includes a real time Micro Vax equipped with an ELN operating system which is combined with an interface unit for data acquisition. The data acquisition unit receives, identifies and accommodates the data from the different A/C systems for further processing in the Micro Vax computer. Details of the prototype contains Fig. 11. The prototype is the basic tool for the flight test verification of OLMS. To allow a flight phase related load calculation and storage the information from the Flight Warning Computer (FWC) has been used, see Fig. 12.

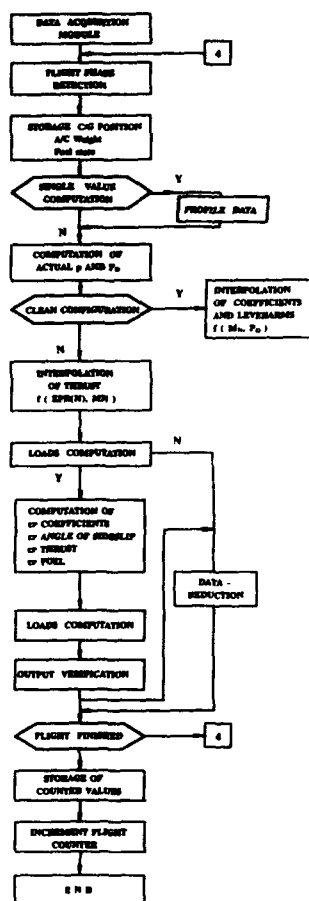


FIG. 9 FLOW CHART OF OLMS PROCESSING

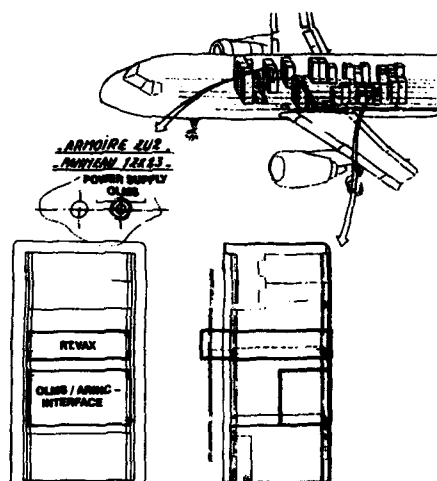


FIG.10 INSTALLATION OF THE OLMS DEVICE IN A/C 1

PROTOTYPE HARDWARE

- ARINC 429 ACQUISITION MODULE
- REAL TIME MICRO-VAX, ELN, OPERATING SYSTEM
- 1 MBYTE MEMORY
- 2 DISK, TAPE RECORDER (TK53)

PROTOTYPE SOFTWARE

- CONTROLLING: PASCAL; CALCULATION: FORTRAN 77
- REFRESHING RATE 64 SPS (MAX)
- LOAD CALCULATION 50 SPS
- 43 LOAD QUANTITIES
- MATRIX 50 BY 50 CLASSES / QUANTITY
- DATA FIELDS: AERO, MASS, THRUST, LEVERARMS

FIG.11 OLMS TECHNICAL DETAILS

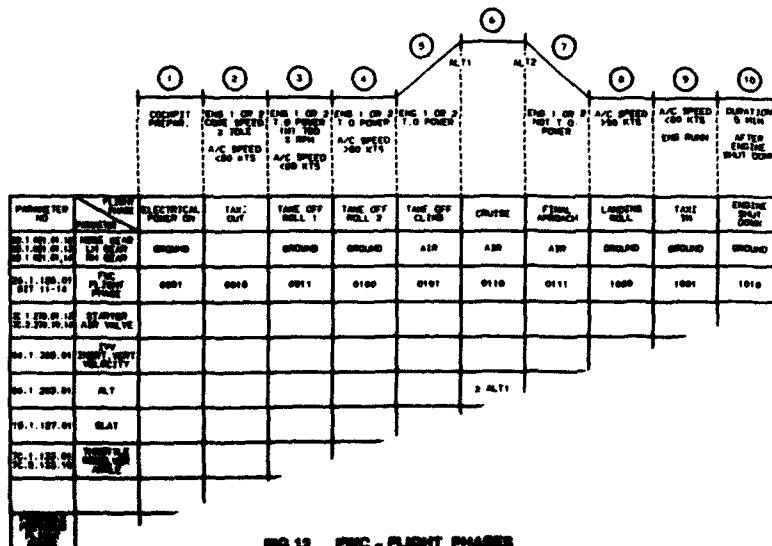


FIG.12 PWC - FLIGHT PHASES

4. VERIFICATION

The principle procedure of the verification is shown on Fig. 13 [2,4].

The following questions concerning the verification of OLMS have been investigated:

- System interference
- Real time behaviour
- Quality of recalculated loads
- Quality of reduced data
- Appropriate capacity of the memory

4.1 System interference

With the selected OLMS prototype the question concerning system interference can not be answered totally. The investigation was reduced to check that no impact is possible from the OLMS unit to other A/C systems and to meet the requirements for secondary systems under development conditions.

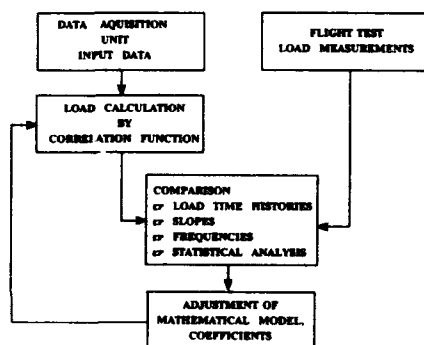


FIG.13 VERIFICATION OF OPERATIONAL LOAD MONITORING SYSTEM

4.2 Real time behaviour

The implementation of the OLMS hard- and software in the test aircraft revealed that the prototype was able to realize an update rate of 50 sps for all OLMS functions (data acquisition, data processing, data reduction) with the required number of quantities to be determined (loads and mission data).

It is necessary to remark that the first function: data acquisition has a constant computing time while the second step: data processing has a constant block time for any of the 50 time steps. For the third step: data reduction the elapsed time varies depending on the flight stages and the dynamic behaviour of the calculated load time histories.

Altogether the conclusion can be drawn that the computer power of a Micro Vax combined with a real time operating system (ELN) is sufficient for this task.

4.3 Quality of recalculated loads

The accuracy of the recalculated loads as required depends on the accuracy for fatigue assessments. It is obvious, that a total agreement is not possible since there is no absolute quality to compare. The desired quality to be achieved is given by the number of classes counted in the data reduction.

A total accuracy between 3 and 5% seems to be sufficient. The 50 classes chosen for the data reduction allow a $\pm 1\%$ discrimination (2% total) related to the total range (limit load or region of operation). Hence all further work concerning accuracy of the recalculation is valued on this fact.

The verification of the load recalculation includes the steps:

- Comparison of recalculated loads and measured quantities
- Check of the measurement quality
- Adaptation and improvement of the recalculation software

The main flight quantities such as altitude speed, machnumber, rates, acceleration were regarded as accurate and a comparison between received OLMS data and quantities of the data acquisition system has been performed. Anemometry data such as incidence and angle of sideslip have been checked and corrected during the flight test phase. The quality and repeatability can be regarded as sufficient. An important item is the verification of the load calculation algorithmen itself. Thorough checks and comparisons with measured loads were necessary.

The load measurements in the test aircraft were realized by calibrated strain gauge installations as it is state of the art [5]. During the flight test phase of the aircraft the principle errors of the measurement have been recognised as

- Long term drifts
- Residual thermal drifts
- Calibration effects
- Offsets
- Problems of repeatability

All those findings have been considered during the verification work. An other major item to solve was the treatment and correction of the data incorporated in the OLMS software.

These data were:

- Geometrical/configuration data
- Mass data
- Thrust models
- Atmospheric data
- Aerodynamic data

During the verification phase the geometrical and configuration data from the test A/C have been used. Prior to in-service operation possible changes and modifications have to be considered. As an example a modification of the slat/flap angle can be cited (depending on engines).

Concerning the mass data the components leading to the OWE are incorporated in the OLMS software. In order to cope with the different mass distributions the possible loadings of the A/C are incorporated using the loading plan number and fuel state data, respectively. The discrete loads from the landing gear (derived from the weight and c/g position of the A/C) also are included.

For the thrust model the data for the actual engine are considered. To cope with the different engine types the thrust models of all selected engines are included and will be activated by the A/C identification number.

Atmospheric data are calculated by the normal atmosphere using the normal temperature distribution under ISO-conditions.

The adaptation of the aerodynamic data which are stored as grid points representing different mach numbers and dynamic pressures is very complicated. These values have been taken from the aerodynamic data bank and are results of wind tunnel tests and calculations. It is well known that those numbers do not represent the real data related to the full scale A/C in all cases due to different reasons:

- Windtunnel errors
- Model accuracy and model laws
- Calculation problems

For the verification of the loads comprehensive investigations have been carried out during the development test phase of the A/C.

The method applied for this task was a so-called output error procedure using maximum likelihood algorithms with and without Kalman-Filter technique [6].

For these investigations maneuvers have been used to check the handling qualities, dynamic load maneuvers but also special stochastic inputs of the controls (multi step) [7].

A main problem to solve was created by the fuselage loads because the FTI-measurement loads were influenced by the cabin differential pressure.

The major item which has been investigated was to ensure the full repeatability during all flights. Therefore some modifications have been implemented into the OLMS software.

Another important item is the treatment of special load shares which cannot be calculated directly using the data from the different data buses. One of these is represented by the incremental loads due to the different structural modes. In many cases the load shares are below the threshold given by the counted classes but a very thorough investigation has been applied in order to qualify these influences. As an example the load increments due to the first wing bending mode during ground operation and during flight are not negligible.

Due to the lack of input data, no discrete algorithm is available, a transfer function has been used and the coefficients are adapted until the residual error is suppressed below the given threshold. During flight the results are satisfactory but for ground operation it is intended to incorporate the landing gear into the load monitoring activities when in connection with the A320 Weight and Balance System (WBS) proper landing gear sensors will be available.

4.4 Quality of reduced data

The data reduction that means the statistical analysis of the calculated load time histories by means of the Range Pair Range counting method - was the last step of the verification.

The quality has been checked by comparisons of counting results determined from OLMS calculation and flight load measurements. It is self evident that a good agreement in the load time histories also lead to a good agreement in the counting results. Incorporated in the check was also the correct function of the OLMS logic, the automatic switch over to specified memories when flight phases are changing.

4.5 Appropriate capacity of data memory

The decision which data should be stored in the OLMS memory has been made on the basis of the later usage for fatigue assessments. For this task a not so frequent retrieval of data seems to be sensible in order to reduce the workload.

On the other hand a more frequent readout will help to find and adjust trends of the A/C usage and can also indicate possible malfunctions of the system. As a good compromise a regular data retrieval at a time of 2000 flights was chosen. For a short haul transport this means a time interval between 6 and 12 months.

On this basis the memory has been designed. As mentioned above the test OLMS is equipped with a disk and tape system for data storage. The serial system will have a fixed memory (EEPROM) for final data storage. Nevertheless all respective investigations have been made to optimize a memory for the planned serial system. From the different loadings including the countings in different flight phases, see Fig.12, a memory of about 60 K-words is necessary.

For the other items which have to be stored for every flight a memory of 32 K-words is necessary. With a 4 byte word and a back up for modifications, additions and possible delayed data retrieval a 0.5 Mbyte memory is fully sufficient. For the task of fatigue investigation there is no need to store additional data.

4.6 Results

In this chapter results of the OLMS-verification will be presented. In total 10 FTI-flights (15 FH) have been investigated.

An example for the identification of measured and recalculated horizontal and vertical tailplane loads shows Fig.14. For this verification an output error identification procedure, the maximum likelihood method has been used. After identification the measured and recalculated tailplane loads are in a very good accordance.

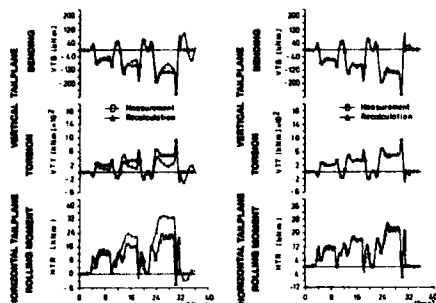


FIG.14 COMPARISON OF CALCULATED AND MEASURED LOADS BEFORE (LEFT) AND AFTER (RIGHT) IDENTIFICATION

4.6.1 Maneuvers

A comparison of FTI-measured and OLMS-calculated fin shear loads during a multi-step maneuver represents Fig.15. The shear loads have been calculated first using coefficients of the aero data bank and second FT validated data. The differences in the load time histories are very small.

Fig.16 shows comparisons of OLMS-calculated and FTI-measured loads during vertical and lateral maneuvers for load quantities of the wing and the horizontal as well as vertical tailplane. The accordance in the load time histories is of good quality.

A comparison of OLMS-calculated and FTI-measured aileron hingemoments during a multi-step maneuver is represented on Fig.17. The calculated aileron hingemoments reflect the measurement with best accordance.

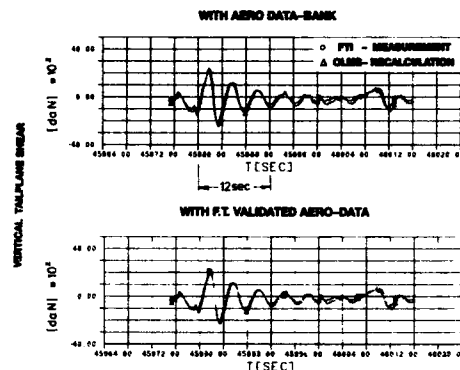


FIG.15 COMPARISON OF MEASURED AND RECALCULATED FIN SHEAR LOADS

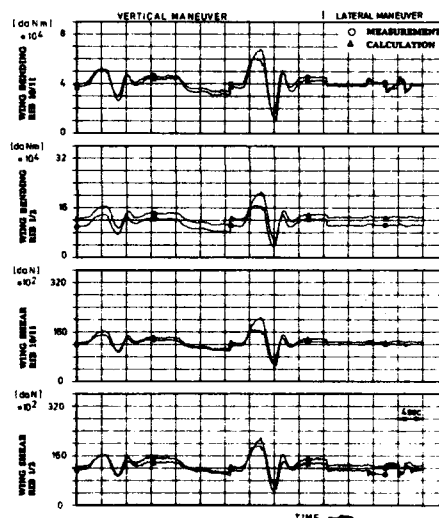


FIG.16 COMPARISON OF OLMS-CALCULATED AND FTI-MEASURED LOADS DURING VERTICAL AND LATERAL MANEUVERS

For not clean configuration some investigations have been performed to obtain the calculation in accordance with the measurement. Especially during flap/slat retraction to clean configuration and vice versa differences have occurred on the shear and bending wing loads in the sections rib 1/2 and rib 10/11. This difference was caused by unbalanced aero data fields. Hence the whole aerodynamic data field for not clean configuration has been harmonized and represented in dependence of the slat/flap angle. Examples of the final results shows Fig.18 for different take off configurations.

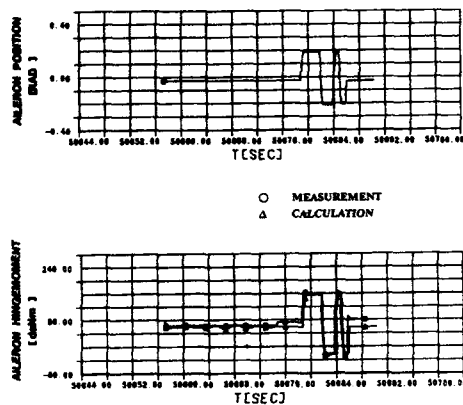


FIG.17 COMPARISON OF OLSM-CALCULATED AND FTI-MEASURED AILERON HINGEMOMENTS DURING A MULTISTEP MANEUVER

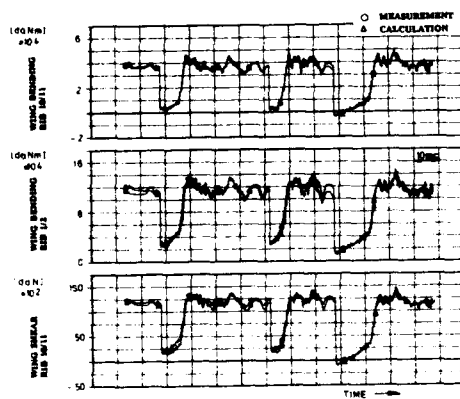


FIG.18 COMPARISON OF OLSM-CALCULATED AND FTI-MEASURED LOADS DURING DIFFERENT TAKE OFF CONFIGURATIONS

4.6.2 Whole flights

In the following results of OLSM-recalculated loads compared with FTI-measured loads of a whole flight will be presented for the main A/C components (wing, rear fuselage, horizontal- and vertical tailplane). The results are presented in form of load time histories as well as cumulative frequency distributions on Fig.19 to Fig.31. Altogether the results indicate a good agreement between load calculation and measurement.

Hence the feasibility is demonstrated for load calculations.

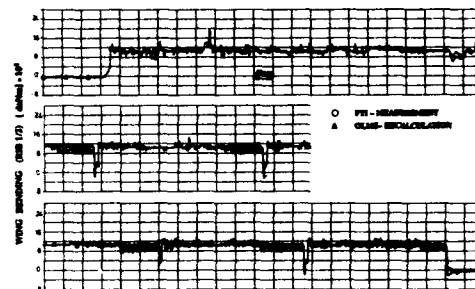


FIG.19 WING-ROOT BENDING DURING A WHOLE FLIGHT

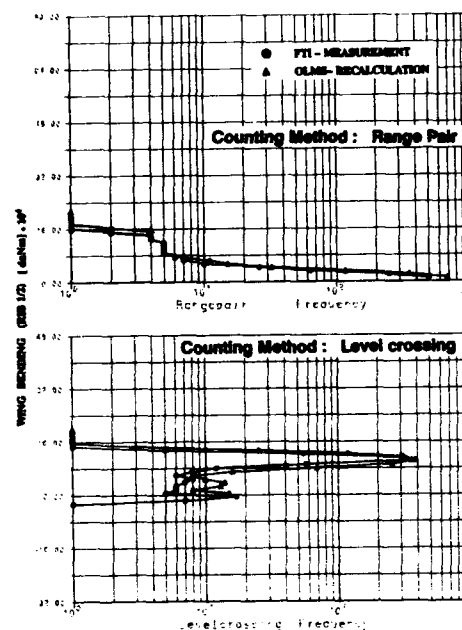


FIG.20 FREQUENCY DISTRIBUTIONS OF THE WING-ROOT BENDING DURING A WHOLE FLIGHT

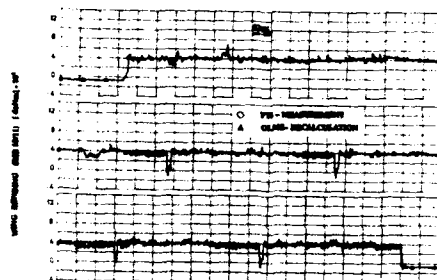


FIG.21 WING BENDING OUTSIDE ENGINE DURING A WHOLE FLIGHT

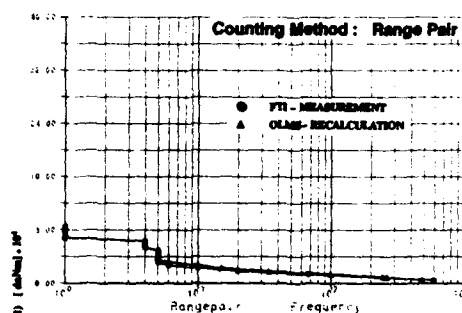


FIG.22 FREQUENCY DISTRIBUTIONS OF THE WING BENDING OUTSIDE ENGINE DURING A WHOLE FLIGHT

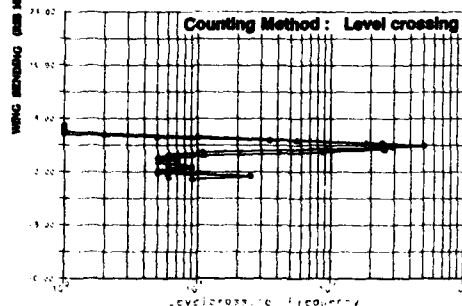


FIG.23 REAR FUSELAGE BENDING (MBZ) DURING A WHOLE FLIGHT

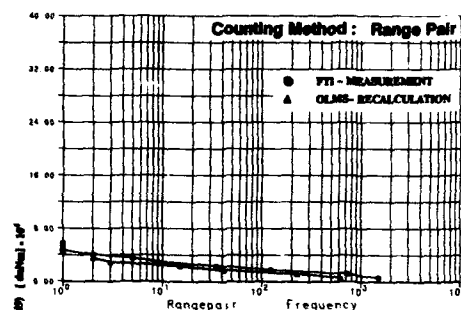


FIG.24 FREQUENCY DISTRIBUTIONS OF THE REAR FUSELAGE BENDING (MBZ) DURING A WHOLE FLIGHT

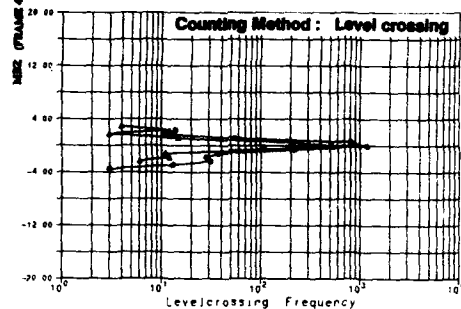
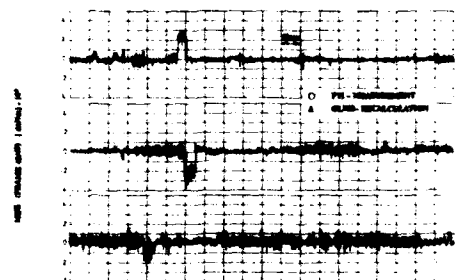


FIG.25 REAR FUSELAGE BENDING (MBY) DURING A WHOLE FLIGHT



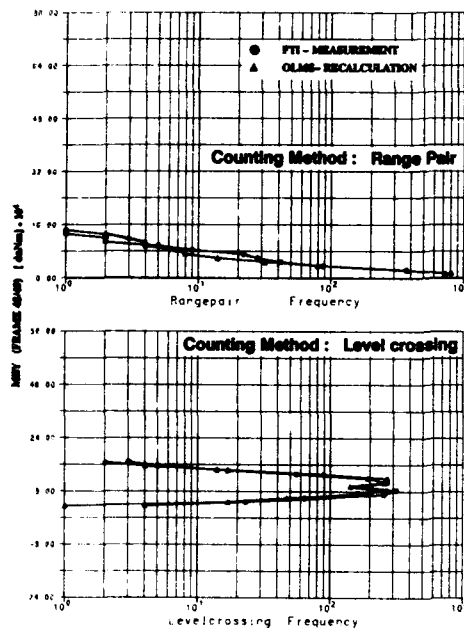


FIG.26 FREQUENCY DISTRIBUTIONS OF THE REAR FUSELAGE BENDING (MBY) DURING A WHOLE FLIGHT

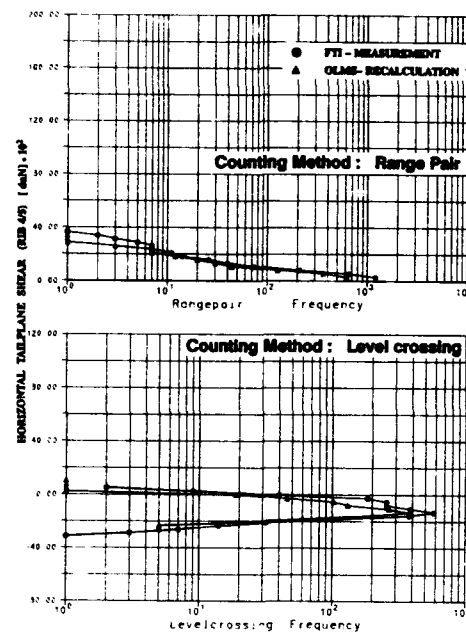


FIG.28 FREQUENCY DISTRIBUTIONS OF THE HORIZONTAL TAILPLANE SHEAR DURING A WHOLE FLIGHT

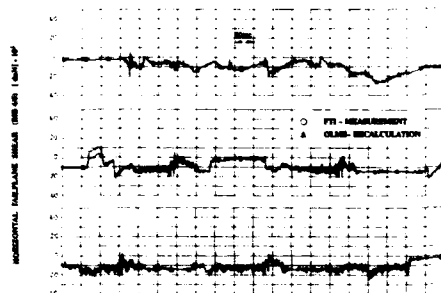


FIG.27 HORIZONTAL TAILPLANE SHEAR DURING A WHOLE FLIGHT

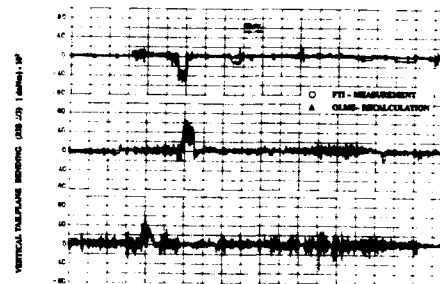


FIG.29 VERTICAL TAILPLANE BENDING DURING A WHOLE FLIGHT

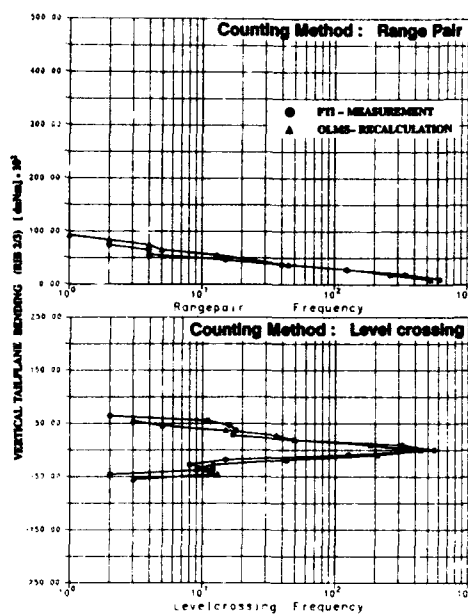


FIG.30 FREQUENCY DISTRIBUTIONS OF THE VERTICAL TAILPLANE BENDING DURING A WHOLE FLIGHT

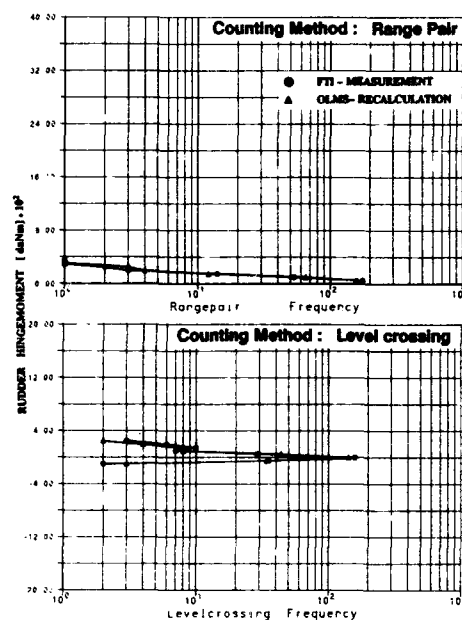


FIG.32 FREQUENCY DISTRIBUTIONS OF THE RUDDER HINGEMOMENT DURING A WHOLE FLIGHT

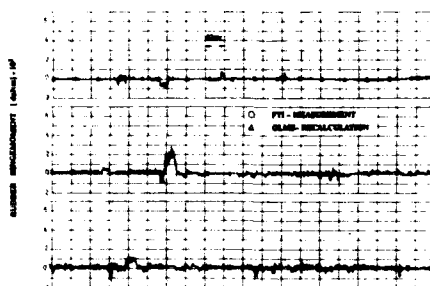


FIG.31 RUDDER HINGEMOMENT DURING A WHOLE FLIGHT

5. SERIAL SOLUTION

How the planned serial device of OLMS could be architected is shown on Fig.33. Important items concerning the specification of the OLMS-LRU are:

- Small dimensions
(not more than 3 MCU)
- Small weight
(less than 4 kg)
- Power consumption of 50 Watt
- Total memory of 0.5 Mbyte
- Data retrieval after 2000 flights
- Monitoring indication
(for example load monitoring running and load monitoring disturbed by LED's)
- It should be intended to incorporate the landing gear completely into the OLMS when in connection with the development of the A320 Weight and Balance System (WBS) proper landing gear sensors will be available.

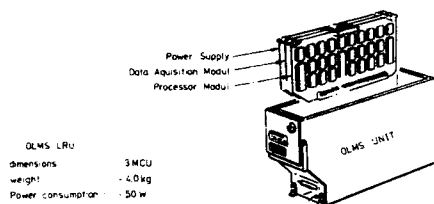


FIG.33 EXAMPLE FOR THE OLMS HARDWARE INTEGRATION (SERIAL DEVICE)

6. IMPACT OF OLMS / ACMP ON THE A/C Inspection Program

Up to now inspection threshold and intervals have to be fixed by fatigue calculations and test results under predicted loads for all aircraft of a fleet and independent of the actual load experiences during service. These results represent the essential part in the Maintenance Review Board (MRB). Experiences by a fleet lead program possible allow revisions of the inspection schedule.

The following part describes how the inspection procedure could be changed and improved by means of the ACMP, see Fig.2. The main aim of ACMP is to perform inspection on condition. That means that the inspection schedule will be influenced by the actual load experiences of individual aircraft. The loads data, prerequisite for this task will be provided by the OLMS-unit.

During the ground based procedure CARE fatigue data necessary for the adjustment of the structural inspection program, will be established. This will be done by two different programs, ALHP and SIAP.

In the first step the actual in service load spectra and mission data will be collected and analysed. The loads will be compared with the predicted loads and information about the confidence belt will be provided, the loads will finally be presented in a Loads- and Mission Report.

In the second step investigations aimed on the revision of the inspection program (MRB-revision) will be carried out. Starting from the load- and mission report, the initial data for the SIAP will be prepared i.e. derivation of stress spectra for SSI's, crack propagation calculation, fatigue life estimation and consumption. Based on these results actions have to be initiated concerning inspection threshold and -interval. The results will be presented in a structural repercussion report for the different A/C components. The primary results will be summarized in a synoptical report for the airlines.

7. CONCLUDING REMARKS

In this contribution the development and realisation of the OLMS-prototype, the methods of the verification and validation have been described in form of a feasibility investigation. The feasibility that loads can be calculated by means of mathematical models combined with load relevant parameters and aero data with a high measure of accuracy compared to FTI-load measurement data has been demonstrated.

In some cases the verification was very difficult since the FTI-measurement was affected by long term influences: for example:

- . Temperature drifts (fuselage strain gauge bridges).
- . Influence of cabin differential pressure (fuselage strain gauge bridges).
- . Influence of different fuel tank pressure conditions (wing strain gauge bridges).
- . Strain gauges failed (vertical tailplane)

In some cases it could be shown that the recalculation of loads leads to better results than the FTI-measurement (fuselage).

For future flight test investigations and parameter identifications some experiences have been gathered.

For example the need:

1. To take complete flights into consideration (long term effects)
2. To check the strain gauge installation in the wing area (influence by different fuel tank pressure conditions)

In relation to the progress of OLMS the following is worth to be proposed:

- . The verification results as presented in this report shall be regarded as basic data for the development of the OLMS serial device.
- . In the future the verification should be started together with the flight test investigation of the A/C concerned. This is very important because multiple work can be avoided and also different flight test and verification related tasks could be better timed and coordinated. Furthermore the disadvantages of strain gauge failures can be prevented.
- . Creation of a so called MINI-OLMS tailored for better realisation possibilities. This implies good chances for extent application. For example selected quantities to be monitored or a further choice of them:

1. Loads

- Wing root bending
- Horizontal tailplane bending
- Vertical tailplane bending
- Rear fuselage bending
- Rudder hinge moments

2. Load parameters

- Load factors n_x , n_y , n_z
- Gust velocities

3. Mission data and parameters

- Weight, altitude, speed, c/g position, payload, fuel, flight duration and number of flights.
- Flap/slat extension vers. speed
- Position of rudder, aileron, elevator and stabilizer.

4. Special events

- Hard landing detection (sinking speed)
- Overweight landing detection
- Limit load exceedances

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LIST OF SYMBOLS

AILR	Aileron position
ALT	Altitude
AOA	Incidence
B	Side slip angle
CG	c/g Position
C8...C36	Aerodynamic coefficients
ELEV	Elevator position
G	Earth gravity
GW	Gross weight
LATG	Lateral acceleration
Mb	Bending moment
M_H	Mach number
M_t	Torsion moment
P_d	Dynamic pressure
P_y	Side load
PTCA	Pitch acceleration
RALT	Radio altitude
RHSPL	Right hand spoiler position
ROLA	Roll acceleration
ROLR	Roll rate
RUDD	Rudder position
S_2	Shear force
S	Wing area
STAB	Stabilizer position
TAS	True airspeed
VRTG	Vertical acceleration
WM1	Related wing mass
WSA1	Wing shear force due to aerodynamic influences
WSM1	Wing shear force due to masses influences
XWQP1	Leverarm due to pitch acceleration
YAW	Yaw acceleration
YWPP	Leverarm due to roll acceleration

LIFE MANAGEMENT APPROACH FOR USAF AIRCRAFT

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SUMMARY

The USAF Aircraft Structural Integrity Program, that traces its origin to B-47 failures in 1958, was established based on the recognition that repeated loads were a threat to the safety of operational aircraft. Later, it was recognized that manufacturing and in-service damage also had the potential to degrade their safety. This threat has been successfully controlled through the adoption of the damage tolerance approach in 1975. This approach, also referred to as "retirement for cause," is used as a basis for an inspection/modification program to maintain safety throughout the life of the aircraft. However, when the aircraft structure has degraded to the point that multiple site damage has occurred, then the inspection program that was developed for the pristine structure needs to be changed. It is the purpose of this paper to review the occurrences of multiple site damage on USAF aircraft and how this has influenced their lives. This will be done through the experiences derived from the KC-135, C-5 and C-141 aircraft.

1. INTRODUCTION

The Aircraft Structural Integrity Program (ASIP) was initiated in 1958 by the United States Air Force in response to fatigue failures on B-47 aircraft which occurred at that time [1]. The original version of the ASIP was based on the fact that repeated loads were a threat to the safety of aircraft. Prior to this time, aircraft were generally designed based on static strength considerations only, and the factor of safety was supposed to account for deterioration from usage and quality problems in addition to loading uncertainties and material strength. The original ASIP used the "safe life" approach, which was a reliability based method, for design to preclude failure from repeated loadings. It became evident in the sixties that the original version of the ASIP was inadequate since this approach did not preclude fatigue cracking problems and that aircraft designed and qualified by this approach suffered premature failures in service aircraft. There were many aircraft in this category, but the one that was used to highlight the shortcomings of the safe life approach was the F-111. One of these aircraft in December of 1969 suffered a catastrophic failure with only 100 flight hours after the fleet had been qualified for 4000 hours. The F-111 and other service aircraft failures resulted in costly redesign and modification programs. The failure of the safe life approach was partially attributed to the fact that it did not adequately account for damage in the structure derived from the manufacturing process or from in-service maintenance of the aircraft. This deficiency motivated the USAF to adopt the damage tolerance approach. After this approach was applied to the design of the B-1 bomber and to the modification of the C-5 and F-4 aircraft, it was formally established in MIL-STD-1530A dated December of 1975. Since that time, all of the major weapons in the USAF have had their Force Structural Maintenance Plans (i.e., the how, when and where to inspect or modify the aircraft to maintain their safe and economical operation) updated by the damage tolerance approach. The applications of the damage tolerance approach are documented in Reference 2.

When the USAF damage tolerance requirements were drafted in the early seventies and released in MIL-A-83444, dated 2 July 1974, they included the application of damage tolerance to monolithic structures through the slow crack growth requirements. In addition, they addressed fail safe structure through the fail safe and crack arrest requirements. Through experience from observation of the results of teardown inspections of in-service aircraft, the drafters of MIL-A-83444 recognized the need to consider the preexisting flaws in secondary members after the failure of a primary member. Also, from examination of teardown inspections they saw the need to establish the size of these flaws based on whether or not there was a significant likelihood of damage in the secondary structure from a machining operation in the primary structure. When this likelihood was high the structure was called dependent. When this likelihood was low the structure was called independent. Therefore, the fail safe structure requirements in MIL-A-83444 recognized that there was a potential for the loss of fail safety due to multiple site damage and the application of these requirements in a new aircraft should mitigate this problem. The basic concepts for damage tolerance design that would alleviate the potential for multiple site damage are now incorporated in the guidance in AFGS-87221A which replaced MIL-A-83444.

There are, however, many aircraft in the USAF inventory that were not originally designed to the USAF damage tolerance requirements. These aircraft have been assessed against the slow crack provisions of the damage tolerance requirements and for each of them a "Force Structural Maintenance Plan" has been developed that defines how, when and where inspections and modifications must be made to maintain flight safety. The slow crack growth approach for achieving damage tolerance is shown in Figure 1. The time from either the initial flaw or an NDI detectable flaw to critical crack size is called the safety limit. Inspections are recommended at one-half of the safety limit.

The fail safe approach for achieving damage tolerance is shown in Figure 2. It is noted that the process of inspections for the broken member continues until the onset of multiple site damage. At this point the structure must be modified to eliminate this

condition. Although no USAF aircraft has been qualified to the fail safe guidance in AFGS-87221A, the USAF wants to maintain the fail safety that was initially incorporated in the designs of large aircraft such as the KC-135, C-5 and C-141. To accomplish this, steps have had to be taken to recognize the potential for the degradation of fail safety because of the occurrence of multiple site damage. It is the purpose of this paper to review the structural programs on these aircraft to determine if they can be used to provide guidance on future activities to prevent in-service failures.

2. THE KC-135 AIRCRAFT

The KC-135, a tanker aircraft that followed the KC-97, was designed to refuel the B-52 fleet. The prototype designated as the 367-80 was developed by the Boeing Company with their own funds and the USAF ordered limited production in 1954. The KC-135 was derived from the -80 design. The Boeing 707 and 720 were also derived from the -80 design as shown in Figure 3. The first flight of a KC-135 occurred 31 August 1956. Five other configurations of this aircraft were delivered before production was terminated in 1965 after 820 aircraft had been manufactured. A total of 37 different designations of the -135 aircraft now exist. Based on the strength to weight ratios for various aluminum alloys, as shown in Figure 4, a decision was made to use 7178-T6 on the lower wing skin. A fatal defect of this material is shown in Figure 5. This figure shows how poorly this alloy compares with 2024-T3 with respect to limit load critical crack length divided by weight. The decision by the USAF to procure a large number of KC-135 aircraft was excellent. The decision to use 7178-T6 for the lower wing skins resulted in a very costly modification program to remove this material from this part of the aircraft.

The KC-135 was designed with a flight hour goal of 10,000 hours. Later, in 1962, the USAF made a decision to perform a fatigue test to better quantify the expected life. This test was conducted to failure for fifty-five thousand test hours of a 5.1 hour tanker mission. Based on this test, it was believed that a service life of 13,000 hours could be achieved if certain modifications were performed. These modifications consisted of reworking approximately 2000 fastener holes.

Contradictory to the 1962 fatigue test results, the KC-135 aircraft experienced service problems early in its life. The 7178-T6 lower wing skins were designed with stresses approximately fifty percent higher than the 707 aircraft which had the higher toughness 2024-T3 alloy for the lower wing skins. The comparison of the one g stresses for the KC-135 and the 707 are shown in Figure 6. Consequently, aircraft operating between 1,800 and 5,000 flight hours had experienced fourteen cases of unstable cracking in the lower wing skins. In all there have been approximately thirty cases on unstable cracking in the range of 1,800 to 17,000 flight hours. The longest of these cracks was approximately 1.1 meters.

In addition to these problems, by 1968, it had become evident that a service life of 13,000 hours would not be adequate for this aircraft. Therefore, in 1972, another fatigue test was performed to determine the actions required to extend the life beyond 13,000 flight hours. This test was significantly more sophisticated than the earlier 1962 test and was more representative of actual force usage. One of the main differences was in the application of high loads. The 1962 test included an application of ninety percent of limit load every 200 flights. The 1972 test included the application of sixty-two percent of limit load every 200 flights. Because of the retardation effect of the ninety percent limit load application, the 1972 test exhibited earlier and more widespread cracking than exhibited by the 1962 test. In fact, 367 cracks were found in the 1962 test article and 1060 cracks were found in the 1972 test article. However, as with the 1962 test, there was poor correlation in time when the cracks occurred on the fatigue test article and when they occurred in service. The 1972 test article failed catastrophically at 55,500 cyclic test hours. Based on the result of the 1972 test, the USAF determined that the KC-135 wing lower surface would need to be replaced at 13,000 equivalent tanker hours. The results of this test also alerted them to the possibility that the fail safety of the wing structure could be degraded by the presence of multiple site damage.

Since some aircraft were near or already over the threshold of the 13,000 hour limitation, a decision was made to replace the lower wing surface on these aircraft. Further, since the only available design at that time was the original 7178-T6 wing, it was used for the replacement. This was done for twenty-nine wings. This replacement was questionable because it was accomplished with the same brittle material. However, it is likely that it did enhance the safety of the aircraft and also provided wings for teardown inspections.

The teardown inspections of six wings removed for a wing skin replacement served as the basis for an assessment of the influence of crack pairs in the structure. A crack pair was defined as a primary and secondary crack located such that an unstable primary crack could cause the secondary crack to go unstable and therefore precipitate catastrophic failure of the wing. In other words, the assessment was made for the purpose of determining the degradation of the fail safety of the wing because of multiple site damage. Finite element analyses of the wing have shown that approximately twenty fastener holes are subjected to significantly higher stresses in the event of failed skin element. If there was a crack of one millimeter in one of these fastener holes then the residual strength would have been reduced to a level considerably below limit load. A risk analysis for the wing was conducted by R. Meadows from the Oklahoma Air Logistics Center located at Tinker Air Force Base. The

data base included 245 cracks, each of which were 1.27 millimeters or longer in length. Also, 29 crack pairs were found which for the purposes of this study were defined as 1.27 millimeters in length for the primary crack and 0.254 millimeters in length for the secondary crack. Meadows used the most critical crack pair from each aircraft for his evaluation. The results of this examination showed that the mean time for a crack pair to develop was 10,709 to 15,441 flight hours with ninety five percent confidence. Based on these results, Meadows performed a risk assessment and found that by the time the fleet of aircraft had reached a life of 13,000 flight hours, he expected, at best, one loss and, at worst, fourteen losses. This is shown in Figure 7. He also concluded that the degradation of fail safety started at about 11,000 flight hours.

Another teardown inspection of an aircraft with 11,558 flight hours indicated multiple crack alignment at the Wing Station 360 splice. Based on this result, this splice, which is a known area of high stress, would have failed catastrophically with the application of seventy percent of limit load. In December of 1976, a VC-135B was in the depot for reskinning when it was found that the rear spar chord in the center section was severed and there were adjacent wing skin cracks. This aircraft had 12,400 flight hours. Further teardown inspections of wings that had been in service 8,000 to 10,000 hours revealed that there were aircraft in the fleet that were exhibiting more multiple site damage than that found in the population examined by Meadows. Consequently, the recommendation was made that the reskinning should take place between 8,000 and 9,000 flight hours and restrictions should be placed on aircraft that were operating above 8,500 flight hours. The reskin was successfully performed without loss of an aircraft.

3. THE C-5A AIRCRAFT

The USAF signed a contract with Lockheed for the C-5A aircraft in December of 1965. The first delivery was made in December of 1969 and the last of eighty one aircraft was delivered in May of 1973. The wing construction was based on the C-141 design with multiple panels connected by spanwise splices to achieve fail safety. However, the C-5 was not planned to be certified fail safe by the FAA as was done for the C-141. The panels were manufactured from 7075-T6511 extrusions with integral stiffeners and beam caps. Design service life goals for the aircraft were 30,000 flight hours, 12,000 landings and a total of 5950 pressure cycles. In mid 1966 an over specification weight was identified by the USAF and as a result a comprehensive weight reduction program was undertaken. A significant portion of the weight reduction was achieved by removal of wing structural weight.

By the end of 1971, it was evident that the wing in the major fatigue test article (X998), was in a state of general cracking, and would not meet its objective of 120,000 cyclic test hours to demonstrate a safe life for the aircraft of 30,000 flight hours. Cyclic testing of the wing was terminated after 24,000 simulated flight hours because it was in a state of general cracking. Fuselage testing on that specimen continued to 45,000 hours and 18,960 pressure cycles. In 1972, a damage tolerance assessment of the wing structure was made and from these results a safety limit of 8,000 Representative Mission Profile (RMP) was established on 15 January 1975. The safety limit, which is defined as the time in flight hours from an initial flaw to critical, was the life of the wing structure since the small critical crack size precluded an inspection to extend its life. This was a significant event in that damage tolerance criteria was used to establish the life of a noninspectable structure. Prior to this time the safe life approach had been used for this purpose.

In 1973, there was a concern about the loss of fail safety because of cracking in the wing. This concern was partially based on the observation of the nature of the cracking in the full-scale fatigue test article. It was found that cracking that occurred in the spanwise splices often developed in a hole common to a fastener. This observation was the motivation for the "dependent damage" requirement written in MIL-A-83444. Another reason for the concern is that nonlinear finite element analyses performed for the case of a broken panel showed that there were many fastener holes in the adjacent panel subjected to a stress that approached ultimate for the material. Approximately fifty to sixty fastener holes were in this category. Consequently, a small crack of the order of one millimeter in one of these holes could cause catastrophic failure. However, the crack data base for determination of the statistics of the crack population did not exist at that time and the subject was not pursued.

Another damage tolerance assessment of the C-5 wing was initiated in September of 1977 to reevaluate all actions necessary for the protection of structural safety until the wing could be modified. As a part of the effort of the 1977 assessment, a wing was torn down that had been on an operational aircraft (Lockheed number 68-0214). This aircraft had accumulated approximately 6700 RMP flight hours. The teardown inspection was performed on the left wing including the entire center section. A total of 44,641 fastener holes were examined in detail. This inspection revealed 1361 cracks of which 931 were considered significant. There was extensive mechanical damage in approximately twelve percent of the holes and in approximately thirteen percent of the holes the Taper-Lok fastener interference was below minimum. This teardown inspection showed that the existence of a "rogue" flaw in the C-5 fleet was likely. The results from the teardown inspection were extensive enough such that they could be used to assess the risk of failure of the intact structure from fatigue cracking and also could be used to assess the degradation of fail safety from this cracking.

A risk assessment was made to determine the failure probabilities for two cases. The

first case was for the situation where the aircraft was assumed to have a failed panel. This assessment was made to evaluate the loss of fail safety in the structure due to cracking in the adjacent panels. The risk of catastrophic failure for this situation is shown in Figure 8. The second case was for the situation where the structure was assumed to be intact. The assessment for the latter case was aimed at providing an upper bound on the number of hours that the aircraft could be flown before the wing modification would be required. The associated risk for the intact structure is shown in Figure 9. One input to the risk assessment is the probability distribution function for stresses, at a given wing location, associated with a single flight. This distribution function is known quite accurately from the many hours of recorded flight data. Another input to the crack length distribution function. This was derived from the cracks found in the teardown inspection of aircraft number 68-0214. The observed cracks constituted a sample from which an estimate of the parent population was derived. A sample of this is shown in Figure 10. The Weibull, log normal and Gumbel's extreme value distributions were used for the parent populations. There was no significant differences in the calculated risk observed among these distributions. An estimate of the equivalent initial flaw population for the C-5A spanwise splices is shown in Figure 11. The studies performed during the 1977 damage tolerance assessment provided the crack growth rate function and the critical crack size dependency on stress. The analysis determines the single flight probability of failure. For the case relating to fail safety (i.e. the broken panel), the United States Air Force Scientific Advisory Board (SAB) in September of 1977 endorsed the following statement regarding fail safety: "If a structural member fails for whatever reason, then the risk of catastrophic structural failure on a single flight of no more than one in ten thousand is acceptable." The calculations showed that at 7100 RNP flight hours the single flight probability of failure was in fact one in 500. The failure probability of one in ten thousand was reached at approximately 4300 RNP flight hours. After a deliberation, the SAB judged this to be acceptable. The usage of the C-5A operational aircraft was carefully monitored until the wing box change was accomplished. Although wing cracks were found by the inspection program that was imposed by the 1977 damage tolerance assessment, there were no aircraft lost because of this problem.

The new C-5 wing box was designed with a stringent one g stress and fatigue requirement in addition to a damage tolerance requirement that precluded the need for inspections in its 30,000 life. The damage tolerance requirement meant the safety limit on the wing was 60,000 hours based on initial flaws from MIL-A-83444. The wing was fatigue tested for 105,000 hours with a representative flight by flight spectrum with little evidence of degradation. It is not likely that this wing will suffer from multiple site damage during its design life.

4. The C-141 Aircraft

A recent assessment of the C-141 utilized some of the knowledge gained from the work done on the KC-135 and the C-5. The C-141 became operational in the USAF in 1964. It was originally designed and qualified for a life of 30,000 flight hours. In 1974 a damage tolerance assessment was initiated to determine if the life of the structure was adequate to justify stretching the fuselage to obtain an increase in cargo volume capacity. This assessment evaluated the entire structure and examined the fatigue test program that was still being accomplished on a full-scale fatigue test article that was a complete fuselage with wings, pylons and a vertical tail. This fatigue test article, called Specimen A, had accumulated at that time 90,000 cyclic test hours with a block loading sequence of a spectrum that had since been shown to be not representative of operational aircraft usage. Crack growth tests from small flaws were used to develop ratios of lives from the Specimen A testing and from current usage with a flight-by-flight spectrum. From these ratios, which were developed for various locations on the aircraft, it was determined that the C-141 should be able to achieve a life of 45,000 hours. On this basis, a decision was made to stretch the aircraft.

The question now is: "could the life of the C-141 be extended beyond 45,000 hours?" At the time of the original damage tolerance assessment, it was judged that Specimen A was not at its economic life. In fact, this article was tested to a total of 118,468 cyclic test hours with the spectrum changed to the then current usage and the sequence was changed to flight-by-flight. There was a partial teardown inspection program on the fatigue test article, complemented by an extensive nondestructive inspections. The crack size distribution from these inspections is shown in Figure 9. It was found that the crack growth from a 0.127 millimeter corner flaw matches the 41 millimeter crack found in Specimen A. From an examination of the C-5A initial crack distribution in Figure 9, the expected crack size for a probability of 1/6000 is just slightly larger than 0.127 millimeters. This indicates that the C-5A initial crack distribution is reasonable for the C-141. As with the C-5A, a loss of a panel will cause a large number of fastener holes (approximately fifty) in the adjacent panels to have high stresses. These stresses are high enough such that a crack of approximately one millimeter located in this high stress field could propagate rapidly and cause catastrophic failure of the aircraft. This condition is estimated to exist with a one in ten thousand probability of occurrence at approximately 45,000 flight hours. Therefore, flight beyond 45,000 hours could not be accomplished with the desired level of safety.

5. CONCLUSIONS

The three aircraft discussed above have several features in common. First, it was found that fail safety was jeopardized with a relatively small population of cracks.

This may not be true in all cases. For example, an Avro 748 operating in Argentina had a failure due to multiple site damage on 14 April 1976 [3]. In this instance the aircraft was able to sustain damage in numerous fastener holes at a rib location before the wing separated in flight. Also, since the stress is significantly elevated subsequent to a failure of a structural member, the sizes of the cracks in this population are such that it is unlikely that they would be found unless a specialized nondestructive inspection procedure was used. Therefore, it is prudent for the structural analyst to make a judgement during the design process on the extent of cracking that would degrade safety. The second feature is that a combination of fatigue test article and operational aircraft teardown inspections were adequate to identify this problem and permit remedial actions to take place before an aircraft was lost. It is not likely that nondestructive inspections of operational aircraft would have been useful to determine the actions needed to solve these problems because of the requirement for the detection of small flaws. Another feature is that the assessment of risk of failure was a useful tool in the assessment of the severity of the situation. In the case of the KC-135, the risk assessment focused management attention on the problem. This precipitated a series of actions that ultimately lead to the decision on the timing for reskinning the wings. In the case of the C-5A, the potential for multiple site damage degrading the fail safety of the wing did not change the timing for the wing box replacement, although it did make the pilots aware that the wings on these aircraft in late life were not the same as they were when new. For the C-141, the flight hours for the threshold for multiple site damage has been established. This result and other issues relative to the structural integrity of this aircraft will be the basis for USAF management to make their decision on the future of this system.

In summary, it has been shown that multiple site damage can play a major role in the life management program for wing structures in addition to other components of an aircraft. For an aircraft where the maintenance of fail safety is vital to the operational safety of the aircraft then it is essential that there be an assessment of the timing of the loss of fail safety from multiple site damage.

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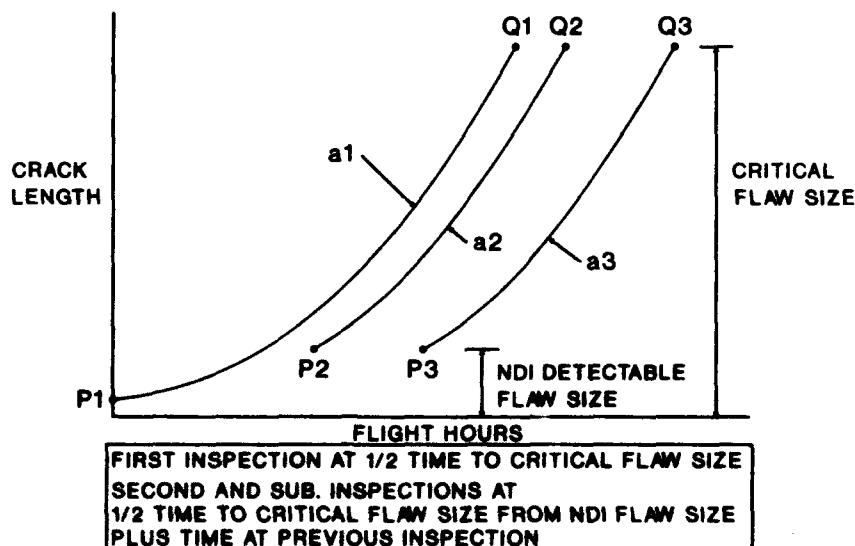


Figure 1. Damage Tolerance Concept - Slow Crack Growth

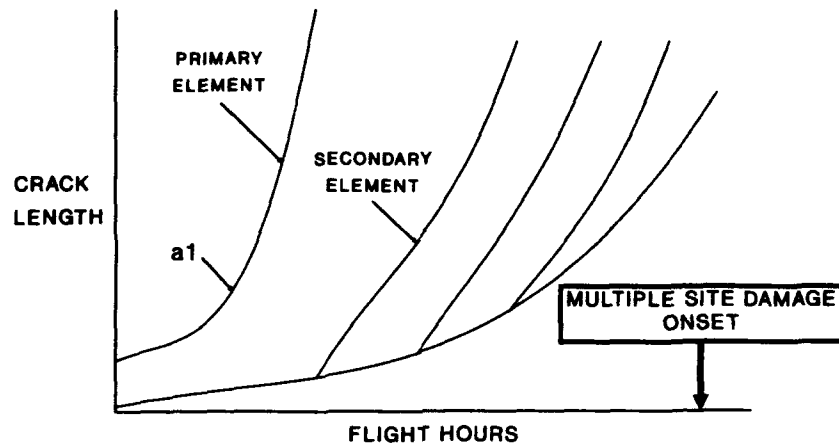


Figure 2. Damage Tolerance Concept - Fail Safe

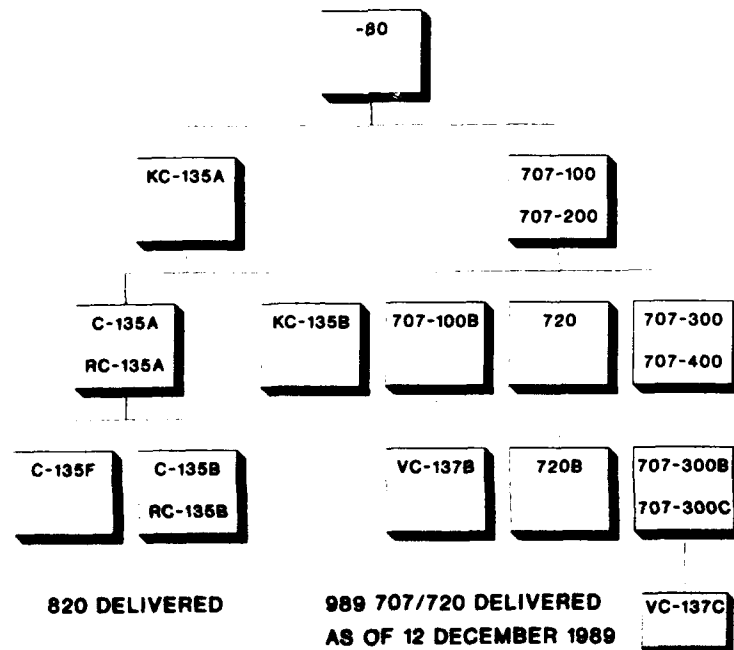


Figure 3. The Boeing -80 Family Tree

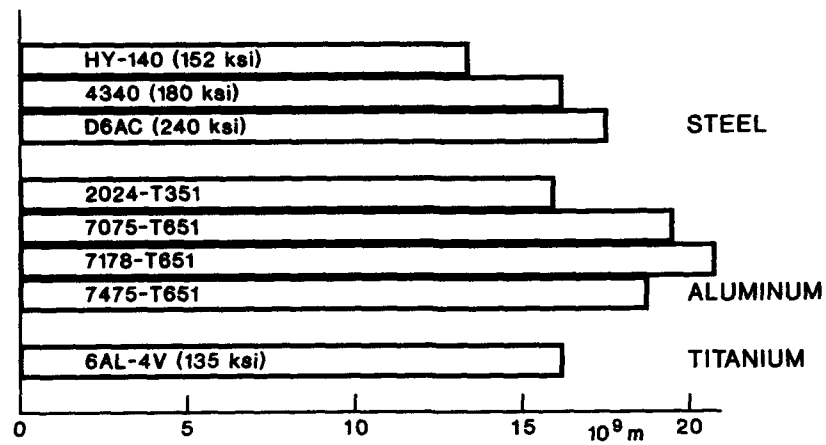


Figure 4. Strength to Weight Ratio Comparison for Metals

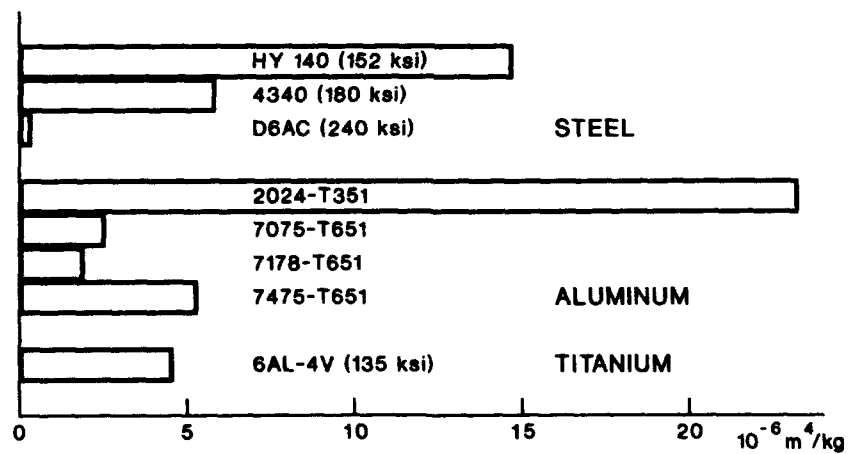


Figure 5. Limit Load Critical Crack Length to Weight Ratio

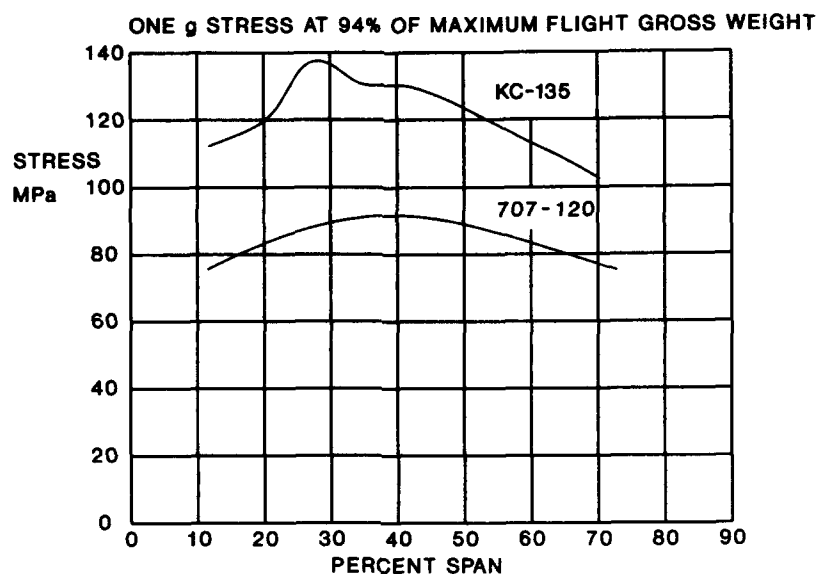


Figure 6. KC-135 Lower Wing Skin Surface Stress

EXPECTED NUMBER OF LOSSES

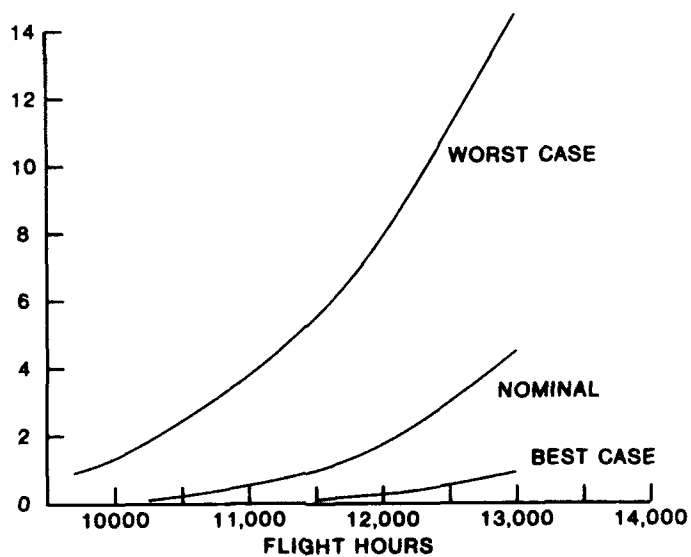


Figure 7. Results of KC-135 Risk Assessment

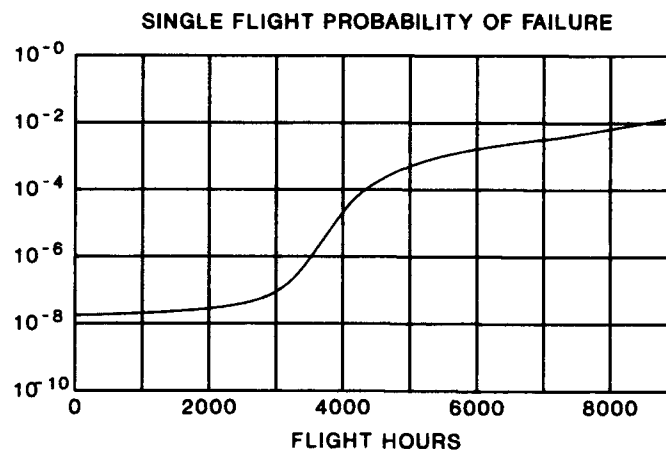


Figure 8. C-5A Probability of Failure with Failed Panel

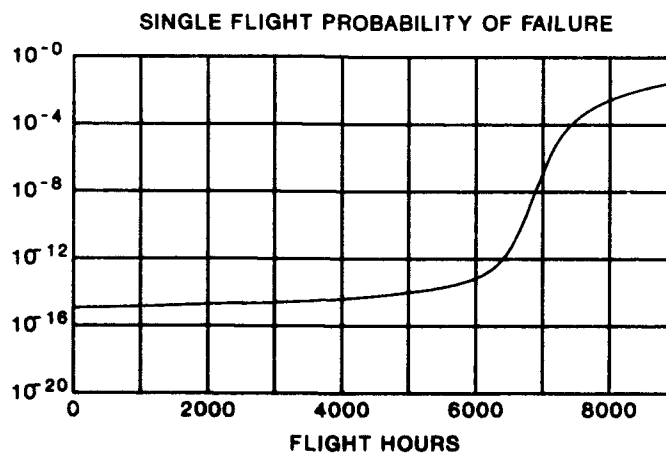


Figure 9. C-5A Probability of Failure for Intact Structure

BASED ON TEARDOWN INSPECTION OF A/C 68-0214

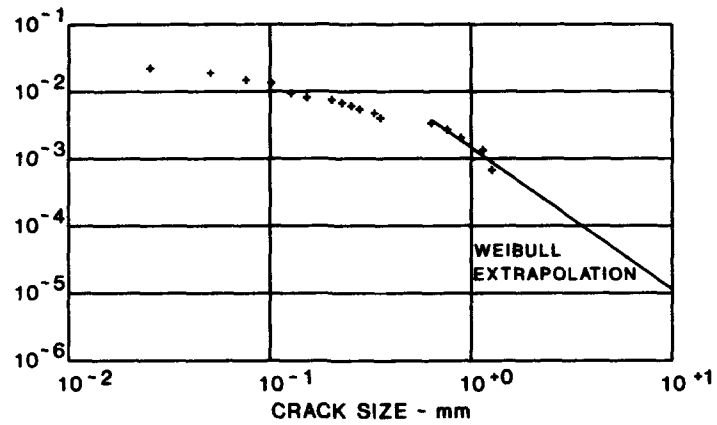


Figure 10. C-5A Teardown Inspection - Crack Size Data

LOWER WING SKIN SPANWISE SPLICES

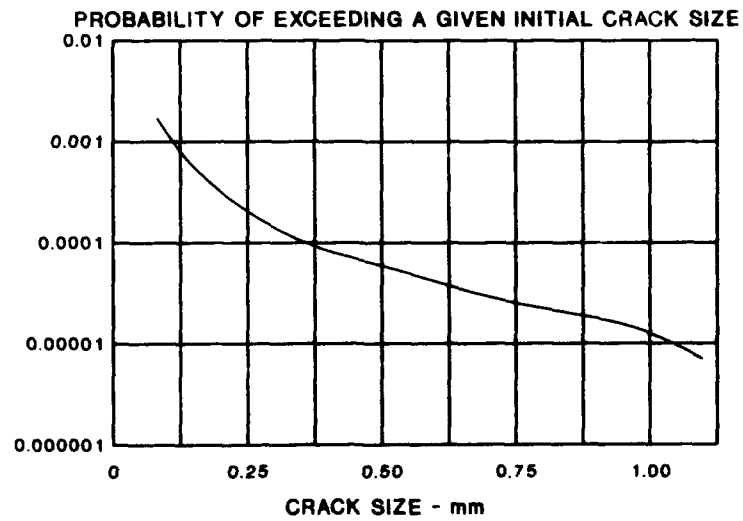


Figure 11. Crack Distribution from C-5A

LARGE AREA QNDE INSPECTION FOR AIRFRAME INTEGRITY

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SUMMARY

Quantitative Nondestructive Evaluation (QNDE) technology is being developed to provide new options for cost effective inspection of airframes. An R&D effort based on five NDE technologies is addressing questions of structural bonding assessment, corrosion detection, multisite damage detection, and fatigue characterization. The research/applications are being conducted by prioritized focussing and staging of the following technologies: 1) Thermal NDE, 2) Ultrasonic NDE, 3) Coherent optical NDE, 4) Magnetic imaging NDE, 5) Radiographic NDE.

This paper focuses on the most recent applications of thermal NDE technology to large area inspection of lap-joint and stiffener bonds. The approach is based on pulsed radiant heating of the airframe and measurement of the surface temperature of the structure with an infrared imager. Several advantages of the technique are that it is noncontacting, inspects one square meter area in a period of less than 2 minutes and has no difficulty inspecting typical curvatures of the fuselage. Numerical models of heat flow in these geometries are used to determine appropriate techniques for reduction of the infrared images thereby delineating regions of disbands. These models are also used to determine the optimum heating and measurement times for maximizing the contrast between bonded and unbonded structures. Good agreement is found between these results and experimental measurements and a comparison of the two are presented. Also presented are results of measurements on samples with fabricated defects which show the technique is able to clearly indicate regions of disbands. Measurements on an airframe also clearly image subsurface structure.

INTRODUCTION

The recent inflight structural failure of an Aloha Boeing 737 has increased the awareness of the need for better NDE techniques for characterization of commercial airframes. Current techniques are typically single point measurements, with correlative relationships developed between probe outputs and characteristic defects. Interpretation of such data are typically subjective, relying principally on the skill of the inspector. As a result of the monotonous nature of the measurement and the unlikely occurrence of a defect, this type of testing is also very susceptible to operator fatigue.

New NDE technologies are being developed to

improve safety of the aircraft by increasing the reliability of the inspection while reducing its cost. These new systems typically incorporate large area scans to develop a global image of the integrity of the airframe. In parallel to the development of these systems, a science base is under development to assist in the interpretation of the measurements and reduce human factor fluctuations in inspection integrity. A second advantage of imaging technologies is a significant reduction in inspection cost.

At NASA Langley Research Center, 5 different technologies are being considered for inspection of aging aircraft fuselages. While these technologies offer complementary information, their integration into a total inspection package will result in a system for aircraft integrity which is greater than the sum of the parts. The pursued technologies in order of staging are thermal, ultrasonic, coherent optical, magnetic imaging, and radiographic NDE. Each of these technologies provides information for inspection of aging aircraft and assessment of structural integrity.

Typical of these systems is thermal NDE. Recently developed real time image processors have already been used to significantly improve the signal-to-noise ratio in thermographic images. These image processors also enable the digitization of the time evolution of the surface temperature of a structure. These digitized records can be processed using physical models of heat flow to extract quantitative structural integrity information from systematic noise sources such as camera artifacts or uneven heating effects.

These advances are coupled with the inherent advantages of thermography detection. Thermography is noncontacting and able to inspect a large area in a short period of time. Thermography has successfully demonstrated detection of disbands in laminated structures, which makes it a good candidate for inspection of bonds in airframe lap joints. It is therefore an appropriate technology for aging aircraft inspection. This paper reports the development of a system at NASA Langley Research Center for thermographic inspection of lap joints.

The technique presented for the detection of lap joint disbands utilizes quartz lamps to heat the front surface of the lap joint and an infrared imager to image its thermal response. An image processor averages and digitizes the thermal images. The time derivatives of these images are calculated to enhance the contrast between bonded and unbonded regions of the lap joint. To quantitatively compare the

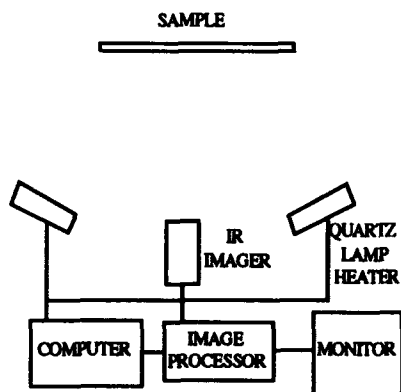


Figure 1. Infrared Measurement System

different time derivative images, contrast is defined to be a function of the moments of the histograms for the bonded regions and unbonded regions of the images. Comparison of the bonded-unbonded contrast for different heating times has made it possible to establish the optimum parameters to maximize the probability of detection of a disbond.

MEASUREMENT SYSTEM AND SAMPLES

A schematic of the measurement system is shown in figure 1. Two banks of tubular quartz lamps (4000 Watts each) were used to heat the front surface of the sample, shown in figure 1. An infrared imager consisting of a single liquid nitrogen cooled HgCdTe detector ($8 - 12 \mu\text{m}$) was used to measure the infrared radiation (thermal response) from the surface of the sample during and after the application of heat. The video output of the imager was digitized and averaged by an image processor. The averaged images were obtained for twelve different time periods, six during heat application and six during cool down, with a constant time interval between averages. A microcomputer controlled the image processor and the application of heat. The twelve images were then transferred to the microcomputer and stored for later analysis.

A typical sample is shown in figure 2, consisting of two sheets of aluminum, bonded with a three inch overlap using a room cure epoxy. The dimensions of the sheets are 123.8 cm by 61 cm and 0.102 cm thick. Disbonds were created by inserting pull tabs (.013 cm thick) of different dimensions into the bonded region before curing. After curing, these pull tabs are then removed to leave voids in the bond.

ONE DIMENSIONAL MODEL

Insights into the basis of the technique are obtained by considering a one dimensional model. The application of heat to a lap joint

results in a difference in temperature between the lap joint regions and regions adjacent to it due to the increased heat capacity of the lap joint. For early times, the lap joint is modeled as a three layer system with an adhesive layer bounded by two other layers of identical material. Initially, the temperature of the system is assumed at equilibrium and a flux is applied to the surface of one of the layers. The time dependence of the temperature is determined using a numerically inverted Laplace transform technique. The thermal properties of the adherents and adhesives are chosen to correspond to the properties of aluminum and a typical epoxy, respectively. The thermal conductivity used for the aluminum is $0.48 \text{ cal}/(\text{sec cm } ^\circ\text{K})$ and for the epoxy is $0.00041 \text{ cal}/(\text{sec cm } ^\circ\text{K})$. The thicknesses of the aluminum and epoxy are 0.101 cm and 0.013 cm.

The time dependence of the temperature of the heated surface is shown in figure 3. Also shown as a comparison, is the time dependence of temperature for both a single layer of aluminum (0.101 cm thick) and a double thickness of aluminum (0.202 cm thick). The input flux is assumed to be 0.0166 watt/cm^2 which is a measured value for a typical configuration of the measurement system. As can be seen from figure 3, after the layers achieve an equilibrium thermal flow condition, the bonded structure increases in temperature at a rate which is approximately $1/2$ the rate of a single layer of aluminum for the one dimensional case.

As heating progresses, the temperature difference between the bonded and the unbonded regions results in heat flowing from the unbonded to the bonded regions. This lateral heat flow reaches an equilibrium state in a period of time dependent on the shape of the disbond, with smaller disbonds obtaining an equilibrium sooner than the larger disbonds. After lateral equilibrium is reached,

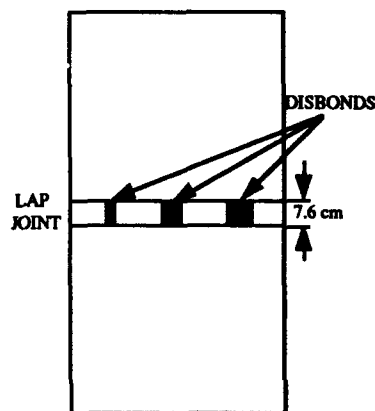


Figure 2. Typical lap joint sample with delaminations of 2.5, 3.8 and 5.1 cm in width.

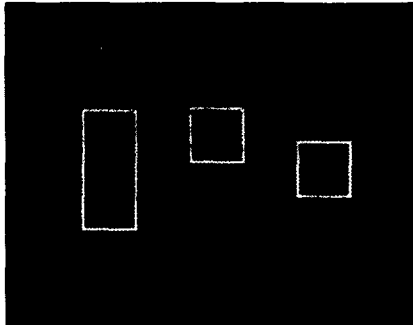


Figure 4. Typical temperature image from simulation of lap joint with disbonds. Regions of disbonds are superimposed on images.

the temperature of the entire sample increases at a fixed rate.

When the heat source is removed, the process reverses itself. After a short period of time, there is lateral heat flow from the bonded region, with its larger heat capacity, to the unbonded region. The removal of a heat flux is modeled as the continuation of the initial flux and with the addition of a new sink of magnitude equal to the initial flux. If the lateral heat flow has reached an equilibrium from the initial flux, then variations in the rate of change in the temperature are due to the application of this sink plus the convection losses. Since the addition of these two heat sinks is greater than the initial flux into the sample, the greatest variation in the time derivative of the temperature follows the removal of the heat source.

THREE DIMENSIONAL SIMULATION OF TECHNIQUE

The limits of the technique are determined by

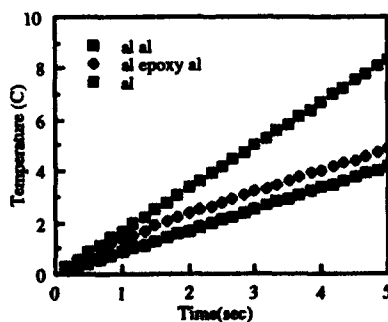


Figure 3. Time evolution of temperature for al layer(al), double layer(al al) and aluminum epoxy aluminum layers(al epoxy al).

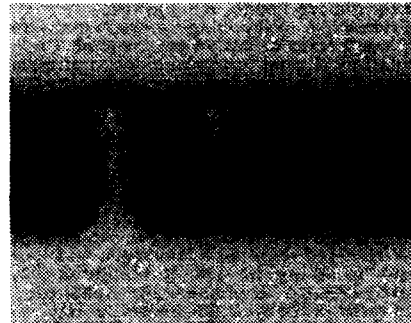


Figure 5. Typical time derivative of temperature image from simulation of lap

the size and shape of the disbond. Therefore accurate modelling of the technique requires three dimensional simulations of the system. These simulations can be used to optimize the technique and determine its limits. Also, the simulations are available for comparison of different data analysis methods using controlled conditions and variations in a manner which is often difficult to reproduce in the laboratory. In this study, the simulations yield an optimum heating time for a particular size and shape defect which can be utilized as a guideline for experimental procedures.

A typical lap joint is represented in figure 2. The sample consists of two thin sheets of aluminum bonded together. The layers are then riveted together. For the region of overlap, the heat capacity is twice the heat capacity of the region where there is only one layer of aluminum (neglecting the heat capacity of the epoxy, which is 4% of bonded structure). If a disbond is present between the two sheets of aluminum, a thermal contrast due to the disbond must be differentiated from the contrast due to the variations in heat capacity across the sample.

Simulations varying heating time and magnitude of both the flux heating input and the free convection cooling components indicate that it is very difficult to detect a partial disbond in a simple unprocessed thermal image. A typical thermal image with gaussian noise levels added is shown in figure 4. Minor variations in image due to the disbond are seen. The simulation model, however, reveals that instead of looking at temperature, observing the time derivative of the temperature more clearly identifies disbonds. An image of the time derivative of the temperature at the end of the heating pulse is shown in figure 5 with the disbond clearly visible.

To determine the limits of the technique, several simulations were run with decreasing disbond sizes down to 2.54 cm wide. While the 2.54 cm disbond appears visible in the

time derivative image, the addition of gaussian noise to the temperature distribution makes detection of a 2.54 cm disbond difficult as is seen in figure 5. The optimum heating time determined from simulation data for this size defect is approximately 24 sec of heating.

DATA ANALYSIS

Since the numerical modeling indicates that the time derivative of the temperature results in an increased contrast between the bonded and disbonded regions, a technique was sought for taking the time derivative of the infrared images. To increase the signal to noise ratio in the time derivative, a time series of images is used in the calculation. The time varying infrared images are fit, point by point, with a second order polynomial. The time derivative is obtained by taking the derivative of the polynomial. In this effort, the twelve images are divided into two time series, the first six and last six corresponding to heat application and post heat application, respectively.

The resulting images are viewed to determine the resolved area of the disbond. To facilitate the analysis of the images, histograms are taken for the different regions on the images corresponding to bonded, disbonded and single thickness aluminum. The histogram overlap is a measure of the ability of the technique to detect disbands of a given size. Therefore, a comparison of the histograms for different heating times and of the time derivatives at different points in the time series is used to establish the optimum heating protocol and data reduction technique for each disbond size.

A comparison of the histograms is done by defining contrast in terms of the moments of the histograms. The first three moments of the histogram are defined as

$$M_0 = \sum_{i=1}^m \text{counts}_i \quad (1).$$

$$M_1 = \sum_{i=1}^m i \text{ counts}_i \quad (2).$$

and

$$M_2 = \sum_{i=1}^m i^2 \text{ counts}_i \quad (3).$$

where m is the number of channels in the histogram and counts_i is the number of counts per channel. For a gaussian distribution M_1/M_0 is the location of the peak of the distribution and the half width at half maximum (w) is given by

A comparison of the widths of the histograms to the separation of the peaks for the different

$$w = 1.17 \sqrt{\frac{M_2}{M_0} - \left(\frac{M_1}{M_0}\right)^2}^{1/2} \quad (4).$$

regions of the image indicates the contrast between the different regions. For this work the contrast between the bonded and unbonded regions is defined as

$$\text{contrast} = 2 \frac{|p_u - p_b|}{w_u + w_b} \quad (5).$$

where p_u and p_b are the peaks of the histograms of the unbonded and bonded regions, respectively, as approximated from equation (4) and w_u and w_b are the widths of the histograms of the unbonded and bonded regions, respectively, as approximated from equation (5). The contrast between bonded and unbonded regions is determined for each image to establish the best heating time and value for j in equation (1).

RESULTS AND DISCUSSION

Selected time derivative images of thermal data acquisition are shown in figure 6. These are calculated from infrared images obtained using the sample shown in figure 2 and heated for 24 seconds. All of the disbands, the smallest being 2.54 cm wide, are discernable in these images. In these images the contrast between the disbonded and bonded regions is greatest at the beginning of heating and immediately following the heating. As the heating or cooling progresses, the contrast decreases corresponding to lateral heat flow equilibrium. The effect of random noise on the time derivative images is also clearest at the earliest and latest times of the time series used for the analysis. The increase in noise results from the time derivative being calculated at extreme points of the time series. The interaction between the increased noise and increased contrast determines the optimum condition for analysis and measurement protocols.

The computed contrast, as defined in equation 5, is calculated for each of these images and is shown in figure 7. The trends of the computed contrast agree with visual perceptions of the relative variations in contrast between images. There are slight variations in the computed contrast which are visually difficult to differentiate. An example of this is that the largest computed contrast is observed in the first post heating time derivative image, yet this variation is difficult to discern visually. Another advantage of this representation is that it allows a relatively simple comparison of a large number of time derivative images. Figure 8 shows experimental data of the contrast variation for different heating rates. The contrast between the bonded and unbonded regions of the image are shown for all the times corresponding to the acquired images. The effect of noise is clear in this figure, where the shorter heating times have smaller signal to noise ratios. Therefore the

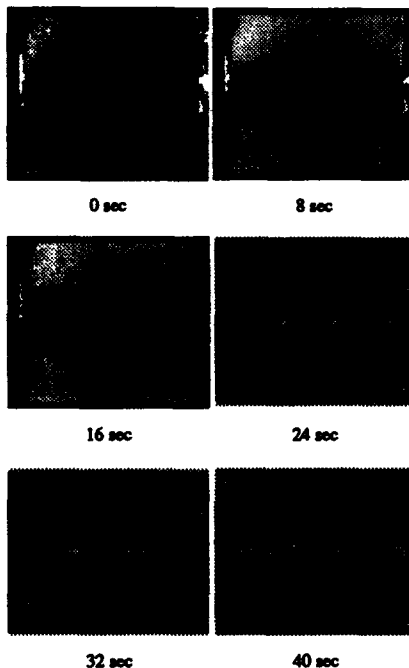


Figure 6. Time derivatives of infrared images at given time intervals both during and following 24 seconds of heating.

shortest heating times have poor contrast in the early images. From figure 8 it is easy to determine that maximum contrast occurs following a 24 second heating of the sample, which agrees with numerical modeling analysis.

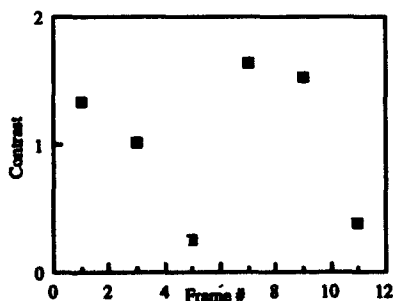


Figure 7. The contrast between the bonded and disbonded regions of the time derivative

CONCLUSION

Significant advancements have been made in quantitative NDE which offers new capabilities of inspection of aging aircraft. In particular, imaging techniques advance possibilities for more reliable inspections at reduced cost. Current efforts are being aimed at realizing those possibilities.

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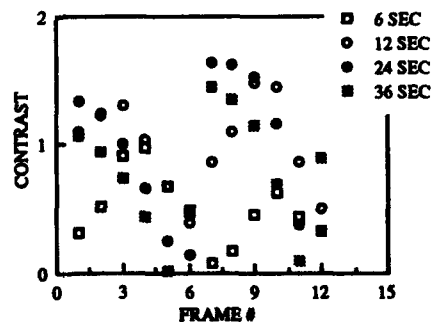


Figure 8. Compilation of contrast between the bonded and disbonded region of the lap joint for different heating rates.

Recent Fracture Mechanics Results from NASA Research Related to the Aging Commercial Transport Fleet

by

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Abstract

NASA is conducting the Airframe Structural Integrity Program in support of the aging commercial transport fleet. This interdisciplinary program is being worked in cooperation with the U. S. airframe manufacturers, airline operators, and the FAA. Advanced analysis methods are under development to predict the fatigue crack growth in complex built-up shell structures. Innovative nondestructive examination technologies are also under development to provide large area inspection capability to detect corrosion, disbonds, and fatigue cracks. This paper reviews recent fracture mechanics results applicable to predicting the growth of cracks initiating at the rivets of fuselage splice joints.

1. Introduction

On April 28, 1988, an Aloha Airlines Boeing 737 experienced an in-flight structural failure when the upper fuselage ripped open and a large section of the skin peeled away. This failure was precipitated by the link-up of small fatigue cracks extending from adjacent rivet holes in a fuselage lap splice joint. This event, brought about by multi-site damage (MSD), helped focus the attention of the industry on the problems of operating an aging commercial transport fleet. Currently, approximately 46% of the jet airplanes in the fleet are over 15 years old, with 26% being over 20 years old. During the past two years the industry has acted to ensure the continued safe operation of the aging fleet. These activities included increased emphasis on maintenance, inspection, and repair as well as mandatory modifications to various models in the fleet. Additional ways of ensuring safety are being vigorously pursued for both the current fleet and aircraft for the next-generation fleet. This paper describes the research activities of the NASA Airframe Structural Integrity Program (ASIP) which has the goal of developing improved technology to support the safe operation of the current fleet and the design of more damage-tolerant aircraft for the next-generation fleet.

Basic research related to the fatigue and fracture of metals, computational fracture mechanics, structural analysis methods, and nondestructive examination (NDE) methods for material defect characterization has

been ongoing at NASA Langley for many years. All of these disciplines have been brought to bear on the problems facing the aging commercial transport fleet. NASA has developed the ASIP in coordination with the FAA and the U. S. airframe manufacturers. The ASIP has two key program elements. They are the development of advanced analysis methodology to predict the fatigue crack growth in complex built-up structure and innovative NDE technologies to detect fatigue cracks, corrosion, and disbonds in adhesively bonded joints. The near-term focus of the program is MSD in lap splice joints. However, the research is generic in nature and the developed methodology is expected to be applicable to many other structural components that may be fracture critical.

The objective of the analysis methodology program is to develop and verify advanced mechanics-based prediction methodology which can be used to determine inservice inspection intervals, quantitatively evaluate inspection findings, and design and certify damage tolerant structural repairs. This objective will be met by developing an analysis methodology that integrates global shell analysis with local fracture mechanics analysis to predict fatigue crack growth in a fuselage structure. This can best be accomplished by developing and exploiting global/local strategies for combining the necessary levels of modeling and analysis. These levels are shown schematically in Figure 1. A sufficiently detailed local analysis is used to obtain the boundary conditions for a physically meaningful yet computationally tractable crack problem. Then a fracture mechanics analysis employing crack closure concepts is used to predict crack growth. The effects of the crack growth must then be integrated upward to the global structural level to insure that correct load transfer paths and internal load distributions are calculated. Of course these analyses may have to be performed in an iterative fashion to achieve correct results.

The program logic is shown schematically in Figure 2. The individual program elements are discussed in more detail in the following sections. A brief overview of the direction of research for each element will be followed by a summary of the status of the work in progress. Relevant results will be discussed in areas

where the research is sufficiently mature for presentation.

II. Fatigue Crack Growth Closure Model

The concept of crack closure to explain crack growth acceleration and retardation was pioneered at NASA Langley almost two decades ago [1]. The closure concept, illustrated schematically in Figure 3, is based on the postulate that the wake of plastically deformed material behind an advancing crack front may prevent the crack from being fully open during the complete loading cycle. Therefore, only part of the load cycle is effective in growing the crack. A plasticity-induced closure model [2] employing fracture mechanics principles was shown to be quite accurate in predicting the fatigue crack growth in aluminum alloys for a number of basic crack configurations for both constant amplitude and spectrum loadings. The closure model has also been successfully used to explain the small-crack phenomenon exhibited by many aluminum alloys. The crack growth rate data must be correlated with the effective stress-intensity factor range rather than the full range to yield meaningful predictions of total crack growth. The successful coupling of the closure methodology with the small-crack growth rate data base has resulted in a total life prediction methodology which treats initiation by predicting the growth of micron size cracks initiating at inclusion particles in the sub-grain boundary microstructure [3]. This type of methodology is necessary to predict the fatigue crack growth of small cracks initiating at a rivet hole before they grow to a detectable size. Furthermore, this methodology may be used to predict the necessary inspection intervals to monitor crack growth before critical sizes are reached and link-up of adjacent cracks occur.

Recent Results of Work-In-Progress

A computer code, FASTRAN [4], has been developed for predicting crack growth using the NASA closure model. The present version of FASTRAN, available in the public domain through COSMIC, runs on a mainframe computer. An example of the accuracy and computational efficiency of this code was demonstrated recently by comparing predicted results to experimental life results obtained from tests of center-cracked tension coupons subjected to the Mini-TWIST spectrum, a standard transport wing structure spectrum of about 64,000 loading amplitudes [5]. At a low mean stress, $S_m = 20$ Mpa, the experimental life (average of two tests) was determined to be 3,230,000 cycles from a 6 mm length crack to failure. The crack length is plotted versus the opening stress calculated from FASTRAN in Figure 4. The predicted life was 20% higher than the experimental value. At a high mean stress, $S_m = 60$ Mpa, the experimental life (average of two tests) was determined to be 152,000 cycles. In this case, the predicted life was 2% below the experimental value. The low mean stress and high mean stress cases required 20 and 1.6 CPU minutes, respectively, on a mainframe (CONVEX) computer using scalar and vector optimization. These predictions were based on calculating the opening stress during very small increments of crack extension throughout the life. However, by using a damage-weighted average opening stress, $\bar{S}_o$, the required computational time was reduced to 11 minutes and 1.4 minutes for the low mean stress and high mean stress cases, respectively. A comparison between the model calculations and the average opening stress is shown in Figure 4. The predicted results were 12% higher and 3% lower than the experimental values for the low mean stress and high mean stress cases, respectively. The first applica-

tion of Mini-TWIST was used to create the characteristic plastic wake for the spectrum. The average opening stress was calculated from the closure model opening stresses during the second time through the Mini-TWIST spectrum. The average value was then used for the repeated spectra until failure was predicted. This approach makes the closure model computationally cost competitive with empirical crack growth models presently used in the aerospace industry. A PC version of FASTRAN is under development which will result in even further computational cost reductions.

III. Fracture Mechanics of MSD

A rigorous fracture mechanics treatment of cracks initiating at rivet holes and MSD will require stress intensity factor solutions for three fundamentally different levels of crack sizes. For very small cracks below the damage tolerance regime, the finite element method will be used to generate solutions to three-dimensional (3-D) crack configurations such as surface and corner cracks initiating at countersunk rivet holes, as shown in Figure 5. The anticipated results from these analyses will be a compendium of analytical expressions for the stress intensity factors for several basic crack configurations. After the cracks extend through the wall thickness and beyond the rivet head, the cracks will be in the detectable range and amenable to the damage tolerance philosophy. The two-dimensional (2-D) boundary element method will be used to generate stress intensity factors for MSD crack configurations prior to extensive link-up, as shown in Figure 6. A PC-based computer code will be developed which will compute the stress intensity factor for each crack tip for an unequal distribution of straight or curved cracks at adjacent rivet holes. After MSD crack link-up, the cracks are quite large and crack growth will be rapid. The stress intensity factor for these cracks will be strongly influenced by the geometric nonlinear response of the stiffened fuselage shell structure. A geometric nonlinear finite-element shell code will be developed that will model the stiffening effects of longitudinal stiffeners and circumferential frames. The crack-tip stress intensity factors will be calculated by employing special crack-tip modeling and an adaptive mesh strategy to properly model the trajectory of the growing crack. Research is underway at NASA to develop methods to rigorously and efficiently address all three levels of crack growth.

NASA has developed several computational methods for computing stress intensity factor solutions to complex crack configurations. The boundary force method (BFM) [6] is well suited to 2-D problems such as through cracks in thin sheet material. An example of the type of problem that can be efficiently treated by the BFM method is illustrated in Figure 7. For more complex problems, the finite-element method and the virtual-crack-closure technique (VCCT) [7] have been successfully employed to obtain solutions to 3-D crack configurations such as a surface crack emanating from a semi-circular notch as shown in Figure 8, where the angle ϕ is 90 at the intersection of the crack and the notch boundary. The equivalent domain integral method (EDI) [8] has also recently been implemented into the NASA finite element codes. This technique is well suited for 2-D and 3-D crack problems involving mixed mode loadings and material nonlinearity. For example, the J-integral computed by the EDI method is compared to analytical results obtained by the VCCT technique in Figure 9. As is illustrated in Figure 9, both techniques give equal accuracy for linear elastic problems. An additional advantage of the EDI method is that it does not require the finite element mesh to be orthogonal to the crack front in order to calculate the

stress intensity factor. Both the VCCT and EDI methods have been implemented into a public domain computer code, ZIP3D [9], available through COSMIC. NASA has also developed analytical equations from the finite element numerical results for the stress intensity factors for many basic crack configurations. Reference 10 provides the stress intensity factor equations for a number of 3-D crack configurations commonly encountered in structural applications.

Recent Results of Work-in-Progress

The near term focus of the NASA fracture mechanics research is on MSD prior to link-up to support the damage tolerance philosophy currently being implemented by the airline industry. An Alternating Indirect Boundary Element (AIBE) computer code [11], a force method version of the more familiar boundary element method, is being developed for MSD crack configurations. The AIBE code will generate the stress intensity factors for the individual cracks of a distribution of unequal, interacting cracks extending from loaded rivet holes. Methods are being developed so that in-plane, out-of-plane, and rivet loads can be included in the analysis. The current version of the code can only model straight cracks. However, a 2-D approach to analyze curved cracks is being developed using the distributed dislocation concept [12]. This method will also be available to analyze MSD crack configurations. Additionally, a special version of FASTRAN has been coupled to the AIBE code so that fatigue crack growth can also be computed.

Each new capability of the AIBE code is being verified by comparing to available analytical results from other computational techniques for simple crack configurations. A complete code verification will be achieved by comparing analytical predictions to experimental results. To date, AIBE code predictions have been compared to experimental results for the special case of multiple fatigue cracks growing from open holes in a thin sheet subjected to remote uniform tension constant amplitude loadings. Figure 10 shows the experimentally measured distribution of crack lengths along with the stress intensity factors for each crack. The stress intensity factors are normalized by the stress intensity factor for an average crack length assuming an infinite periodic array of equal cracks. Figure 11 shows the comparison of the fatigue crack growth calculations for the unequal distribution of crack lengths and the infinite periodic array of equal cracks to the experimental results. It is obvious from these results that the simplistic approach of assuming equal crack lengths may not always generate accurate predictions of crack growth and cycles to crack link-up. Many additional experimental cases of crack configurations and loading conditions are being conducted in order to refine the AIBE code and more fully verify its accuracy.

IV. Nonlinear Stiffened-Shell Structural Analysis

The behavior of large cracks in fuselage structures such as mid-bay cracks or splice joint cracks after MSD link-up are strongly influenced by the stiffening effects of the circumferential frames and longitudinal stiffeners. It is not practical to model all of the structural details in a finite-element analysis. Greater efficiency can be achieved by exploiting a global/local strategy where local details that produce stress gradients can be treated in a companion analysis to the global structural analysis. The structural analysis methodology will have to account for geometric nonlinear behavior as well as large deformation behavior. Effects such as pressure piling, the outward bulge of the skin between stiffeners, must also be accounted

for in the analysis. This is necessary to predict the crack growth direction and crack opening of large cracks that may result in a rapid decompression rather than a catastrophic in-flight failure.

V. Fracture Analysis with Adaptive Mesh

To predict accurately the behavior of a growing crack in a stiffened shell structure, the global/local methodology must be extended to include an adaptive mesh concept so that the local refined mesh can change in a manner dictated by the growing crack. This concept is illustrated schematically in Figure 12 taken from the work of Wawrzynek and Ingraffea [13]. The path of the growing crack is represented by the heavy dark line in Figure 12. The top two schematics illustrate the element deletion and refill concept while the lower two schematics illustrate a crack growth example problem.

VI. Methodology Verification Test Program

The integrated fracture mechanics and fuselage structural analysis methodology must be verified by a test program. As shown in Figure 13, there are various levels of testing required to achieve a full verification of a structural analysis methodology. The goal of the ASIP program is to achieve verification through the curved panel and subscale barrel test article level. This program element will be conducted in cooperation with Boeing Commercial Airplane Company and Douglas Aircraft Company. Each airframe manufacturer will contribute panels with riveted splice joints typical of their manufacturing practices. Furthermore, data from previous fatigue and damage tolerance test programs conducted by the airframers will serve as benchmarks for the analysis methodology verification.

VII. Summary

NASA is conducting interdisciplinary research combining the disciplines of fracture mechanics, structural mechanics, material science, and nondestructive instrumentation sciences for the purpose of developing an advanced integrated technology to support the continuing airworthiness of the aging commercial transport fleet. The objective of the analysis methodology program is to develop and verify advanced mechanics-based prediction methodology which can be used to determine inservice inspection intervals, quantitatively evaluate inspection findings, and design and certify damage tolerant structural repairs. The program is coordinated with the FAA and is being worked cooperatively with the U. S. airframe manufacturers and airline operators.

Fracture mechanics solutions to cracks extending from the rivets in fuselage splice joints are being developed which will be applicable to multi-site damage (MSD). A plasticity-induced closure model is being developed which will be applicable to predicting the fatigue crack growth of MSD crack configurations. This fracture mechanics methodology is being integrated into a finite-element based stiffened-shell structural analysis methodology so that significant geometric nonlinear effects on crack growth may be accurately predicted. The integrated methodology will be verified by conducting fatigue and damage tolerance tests of curved panels with stiffeners and frames.

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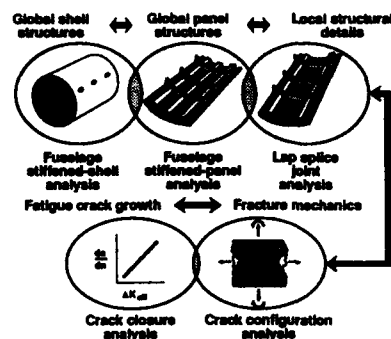


Figure 1. Integrated shell analysis-fracture analysis methodology.

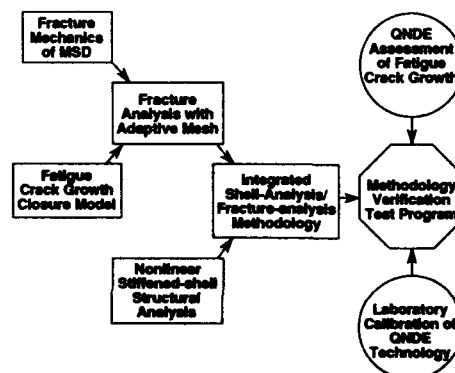


Figure 2. Logic diagram for analysis methodology program.

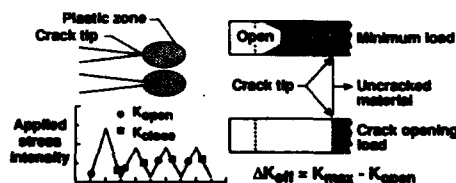


Figure 3. Fatigue crack growth controlled by closure mechanism.

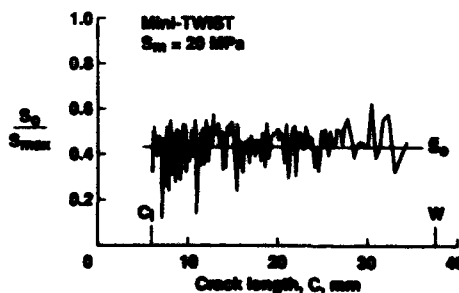


Figure 4. Crack-opening stress variation during right-by-right loading.

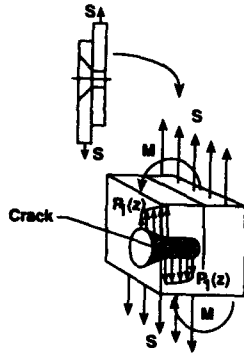


Figure 5. Fatigue mechanics of cracks initiating at countersunk rivets.

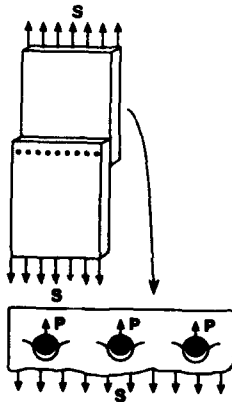


Figure 6. Fracture mechanics analyses of multiple-site cracks.

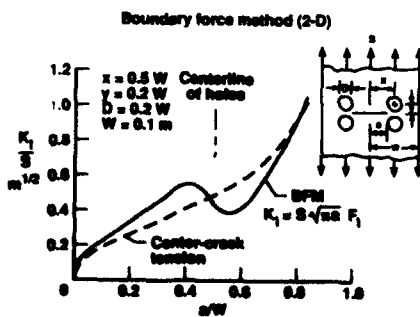


Figure 7. Stress-intensity factors for a four-hole cracked specimen.

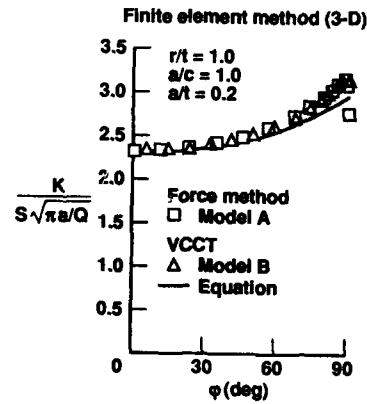


Figure 8. Stress-Intensity factors for a surface crack emanating from a semi-circular notch.

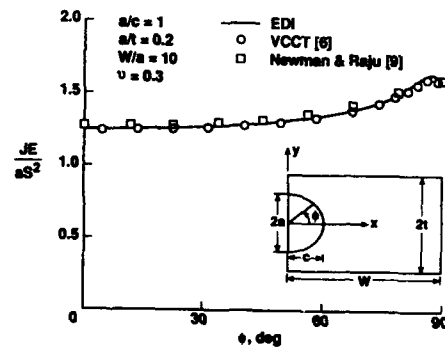


Figure 9. Comparison of normalized J distribution for a semicircular surface crack from EDI, VCCT, and force methods.

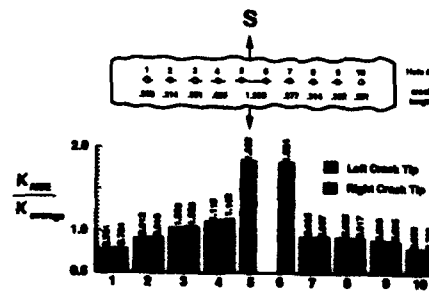


Figure 10. ABE determined stress intensity factor distribution for an open hole field experiment (the crack lengths above are in inches).

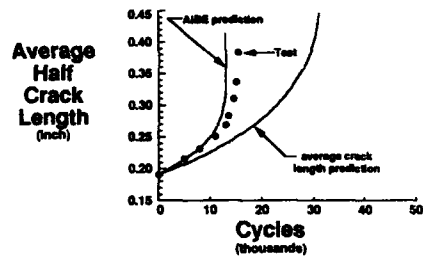


Figure 11. Predicted fatigue crack growth behavior for an open hole MSD experiment.

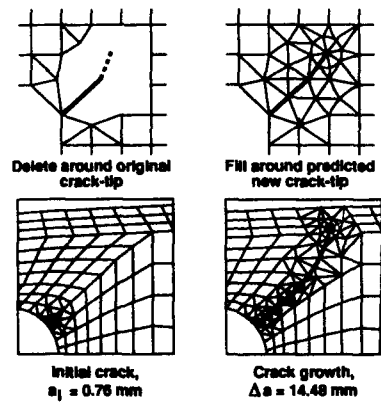


Figure 12. Crack propagation by adaptive remeshing using the finite element method.

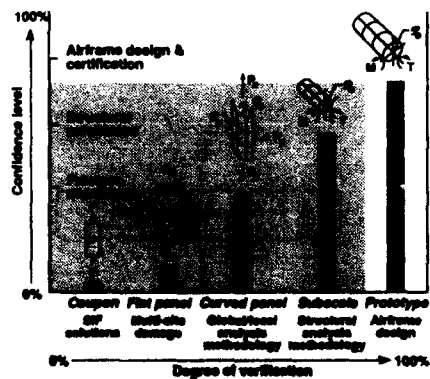


Figure 13. Prediction methodology verification program.

STRUCTURAL AIRWORTHINESS OF AGING BOEING JET TRANSPORTS

by

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ABSTRACT

Structural safety is an evolutionary accomplishment, and attention to detail design features is key to its achievement. A multitude of design considerations are involved in ensuring the structural integrity of Boeing jet transports that have common design concepts validated by extensive analyses, tests, and three decades of service. The active service life of commercial airplanes has increased in recent years as a result of increasing costs and delivery times for fleet replacements. As airplanes approach their design service objectives, the incidences of fatigue and corrosion may become widespread. Continuing airworthiness of the aging jet fleet requires diligent performance from the manufacturer, the airlines, and airworthiness authorities.

Boeing continually reviews reported service data and other first-hand information from customer airlines to promote safe and economical operation of their worldwide fleets. Standard practices of ensuring continuing structural integrity include basic inspection and overhaul recommendations contained in maintenance manuals and service bulletins. Aging fleet support includes timely development of supplemental structural inspection documents applicable to selected older airplanes, teardown inspections of high-time airframes retired from service, fatigue testing of older airframes and structural surveys of more than 100 airplanes operated throughout the world. Lessons learned from these activities are incorporated in service bulletin recommendations, production line modifications, and design manual updates.

Structural maintenance is a cornerstone of continuing safety in commercial air transports. Aging fleet concerns have focused unprecedented industry attention on additional requirements and maintenance actions to ensure continued safety. These are focused on mandatory modification rather than continued inspection; development of improved mandatory corrosion prevention and control programs; consolidation of basic maintenance programs; updates of supplemental structural inspection programs; and development of guidelines to determine the adequacy of structural repairs. These preventive maintenance recommendations will enhance continued safe operation of aging jet transports until their retirement from service. Virtually all commercial airplane manufacturers are involved in similar programs.

BACKGROUND

Diligent attention to detail design, manufacturing, maintenance, and inspection procedures of the last three decades have produced long-life damage-tolerant structures with a credible safety record, as shown in Figure 1. The primary cause of hull loss accidents is attributed to human factors and weather with about 10% coupled directly to the airplane, systems, or propulsion components or how they were maintained. Approximately 3% of hull loss accidents are caused by airplane structure failures. The system used to ensure that the fleet of commercial jet transports is kept flying safely through their service life has three major participants: the manufacturers, who design, build, and support airplanes in service; the airlines, which operate, inspect, and maintain the airplanes; and the airworthiness authorities, who establish rules and regulations, approve the design, and promote airline maintenance performance. Airplane structural safety depends on diligent performance of each participant in this system.

Boeing jet transports are designed to the fail-safe principle (more recently termed damage-tolerant design). Service experience with fail-safe designs has worked well in thousands of cases

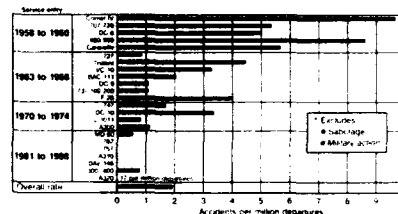
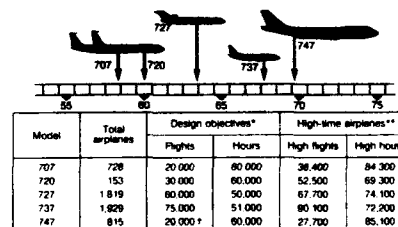


Figure 1. Hull Loss Accident Rates—Worldwide Commercial Jet Fleet

where fatigue and other types of damage have been detected and repaired. The question debated between industry experts is whether the fail-safe design practices used in the 1950s and 1960s are adequate as these airplanes approach or exceed their original design objectives. Figure 2. Boeing jets are designed for conservative flight cycle and flight-hour objectives over a minimum 20 years of economic operational service. There is no service limit to the damage-tolerant designed airplane structures, provided the necessary inspections are performed along with timely repair and/or replacement of damaged structure or preventive modifications. Operational efficiency is affected by the cost and frequency of repair. Structural durability may therefore limit the productive life of the structure.



* At this point in the aircraft lifetime, some initial fatigue might occur.
** Different airplanes are in high flights and high hours.
† Some derivatives have special design objectives.

Figure 2. Boeing Commercial Jet Fleet Summary—January 1991

Test Verification

It is impractical to conduct verification testing for all critical loading conditions and portions of the airplane structure. Analysis methods and allowables verified by test are therefore used for airworthiness certification. Verification tests comprise small laboratory test specimens; large panels, major components representing wing, empennage, and fuselage structure; and full-scale airframes.

Full-scale static testing of new models is conducted to verify limit load-carrying capability and satisfy certification requirements. In addition, full-scale fatigue tests are conducted to locate areas that may exhibit early fatigue problems. This allows necessary modifications of details that might exhibit early cracking and to demonstrate compliance with fatigue design objectives. Full-scale fatigue tests are also useful to develop and verify inspection

and maintenance procedures. Fatigue tests are not intended to demonstrate "safe life" limits of structures certified as damage tolerant, nor are they an alternative to the inspections required for continued safe operation of aging airplanes.

Testing of new airplane structures does not incorporate corrosion and/or accidental damage that can accelerate fatigue cracking. Similar tests are conducted on older airframes to gain insight into the problems that might be experienced on high-time airplanes with repairs and service-caused defects. Extensive pressure testing was conducted on 737 and 747 teardown airplanes to simulate the effects of additional flight cycles. The 737 test extended the pressure cycling from 59,000 cycles at retirement to 129,000 total cycles at test completion, Figure 3. The 747 test extended the pressure cycling from 20,000 service cycles to 40,000 total cycles, Figure 4.



Figure 3. Retired 737 Aft Fuselage Test Article

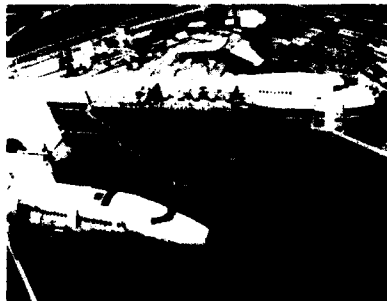


Figure 4. Retired 747 Fuselage Test Article and New Model Test

The 737 test revealed no major new economic- or safety-related fatigue findings. The structure was torn down at test conclusion and examined in detail. The most significant new finding was

cracking at typical frame splices below the aft entry door. A structural advisory was released to all operators, who subsequently found similar cracks in the fleet. A service bulletin has been released to monitor and rectify the problem.

The extended 747 fuselage testing produced timely additional data to define inspection requirements for airplanes operating beyond 20,000 full-pressure cycles. Some local cracking was detected in lap splices past 30,000 cycles. Service bulletins for these areas and some floorbeams and fuselage frames will be issued in 1991. The teardown of the 747 test fuselage is now in progress, after which all test results will be assessed for appropriate follow-on fleet actions.

To supplement airframe testing, two generic pressure test fixtures were fabricated in 1989. One fixture has a radius of curvature of 74 in (1.9m) to match narrow-body 727, 737, and 757 airplanes. The other has a radius of 127 in (3.2m) to match the widebody 747, 767, and 777 airplanes, as shown in Figure 5. Removable test sections are inserted in a cutout in these test fixtures and can be used for fatigue, fatigue crack growth, or residual strength tests. Tests are conducted under pressure loading only, using air as the pressurizing medium. Where appropriate, one crack will be inserted in the test article at a time, and repaired before conducting additional tests.

Test sections are modeled using finite element techniques, and analysis results are compared with a comprehensive set of strain gauge and crack opening displacement measurements. The structural modeling of crack behavior is refined, as necessary, to provide a validated tool for future commercial airplane design. Comparisons of the stress output from initial investigations with the results of the test panel experimental program have shown good correlation, as shown in Figure 6. The initial investigations are now being refined to determine the effect of crack location and trajectory and material nonlinearity on the residual strength of the test panels. An effort is being made to predict the trajectory of cracks that form along fastener lines, as shown in Figure 7.

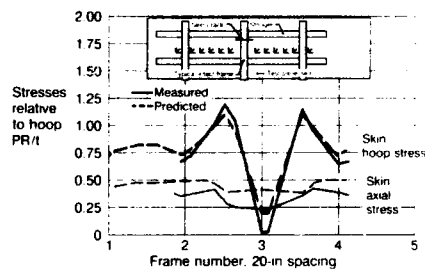


Figure 6. Test Panel Stress Comparison for Crack Midbay Between Stringers—14-in Longitudinal Skin Crack

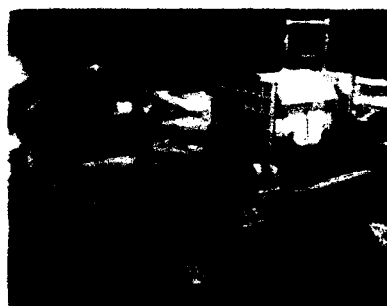


Figure 5. Narrow and Widebody Fuselage Test Fixtures and Test Panel With Safe Decompression at 20 in (50 cm)



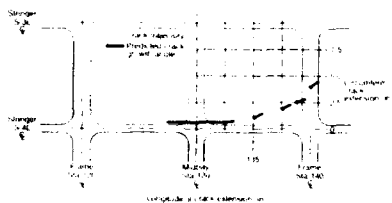


Figure 7. Crack Growth Trajectory Comparisons

Teardown Inspections

In addition to supporting the inservice fleet, Boeing has instituted a number of other efforts to gain more insight into the present and future behavior of aging structures. Figure 8. Such efforts include extensive teardowns and inspection of models 707 and 727 retired from service to determine if any previously unknown or undetectable damage may be starting to develop. These programs started in 1965 and continued through the late 1970s. In some cases, teardown findings have led to design changes, service bulletins, and even airworthiness directives. In general the findings have confirmed previously known or anticipated behavior and have added confidence in both the structures and maintenance programs scheduled for completion in 1991.

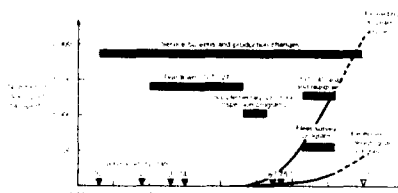


Figure 8. Aging Fleet Support Activities

A 737 with 59,000 flights accumulated during 18 years of service was damaged beyond economic repair in late 1986 and purchased by Boeing for pressure testing of the aft fuselage section and teardown of the total airframe. Figure 9. Inspected structure was found in good condition with little corrosion and with most discrepancies in previously known problem areas. While of no immediate safety concern, some new findings emerged that have been corrected in subsequent derivatives and retrofit kits. A 747 retired from service in 1988 with over 20,000 service flights has been disassembled for detailed wing and empennage inspections with no significant findings requiring fleet actions. The fuselage is currently being inspected after completion of cyclic pressure testing in late 1990.

Fleet Support Actions

For each new airplane model introduced in the United States, the airline operators, the Federal Aviation Administration (FAA),



Figure 9. 737 Structural Teardown of 737 Wing, Empennage, and Forward Fuselage

and the manufacturer collaborate to establish an initial baseline structural inspection and maintenance program. Two documents are typically developed as part of this program: the Maintenance Review Board (MRB) document that defines maintenance requirements and a Maintenance Planning Document (MPD) that suggests means and procedures for complying with those requirements. The MPD is frequently adopted in total by the airline, at least until sufficient experience is gained to warrant program modification that is compatible with its specific operating environment and usage. It is accepted that the airline has the best knowledge of particular needs to maintain the integrity of its fleet of airplanes. Boeing has in place a number of programs, listed in Figure 10, that help support the operation of commercial jet transports and promote a high level of airworthiness in the fleet.

Inevitably, structural discrepancies are discovered during inspection and maintenance, and well-defined procedures exist to handle these problems wherever in the world they occur. In the United States they are reported to the manufacturer and/or the FAA. The manufacturer in turn informs all operators to determine if other operators have experienced similar problems. In many cases, the manufacturer will make a design change in production and/or issue a service bulletin that describes the problem and background and suggests means of corrective action for aircraft in service. For the majority of problems there is no safety issue and therefore no requirement for the operator to follow the service bulletin recommendations. However, for economic reasons, many operators do follow them, either by inspection or terminating modification action.

If the problem has a possible effect on safety, the FAA will consult with the manufacturer and issue an airworthiness directive (AD) that mandates either the corrective actions listed in the service bulletin or an FAA-approved alternative. Typically only 2% will become the subject of ADs that usually are adopted by other airworthiness authorities.

These cooperative efforts between operator, manufacturer, and the authorities to maintain knowledge of aircraft fleet

Fleet Support Programs				
1950s	1960s	1970s	1986	1988
Boeing Customer Support	SSID and other activities	Airplane Fleet Survey Program	Airworthiness Assurance Task Force*	Structures Operator Support Program
- Engineering - Maintenance engineering - Service engineering - Field service reps - Spares	Engineering and customer services	Engineering and customer services	FAA/ATA/AIA Congressional oversight	Service engineering
- Service problems - Service bulletins - Maintenance programs - Repairs - Design changes - Operator conferences	- Supplemental structural inspection programs - Fatigue testing - Teardowns	- Survey individual airplanes - Aging and current fleet leaders - Structures and systems	- Aging airplanes - Structures oriented - All manufacturers	- Field service emphasis on structures - Region focal points - Channel to expedite customer needs

Figure 10. Fleet Support Programs

performance and to take the appropriate action enhance continued fleet safety.

Supplemental Fatigue Inspections

During the mid- and late 1970s, some airplanes of the jet transport fleets began to approach their original service goals in flight-hours and flight cycles. The need was recognized for additional directed inspections to supplement existing maintenance actions to monitor high-time airplanes for the potential onset of fatigue damage with increasing age.

A collaborative effort between the manufacturers, airlines, and the airworthiness authorities resulted in supplemental structural inspection documents (SSID). For each airplane model, approximately 50 principal structural elements were selected for monitoring. No test or service fatigue damage had been experienced in any of these areas; their selection was based on the potential consequences of undetected cracking.

These SSIDs for the 707, 727, 737, and 747 are mandatory, apply to high-time airplanes at specific thresholds, require specified repeat inspections, and have been in place for approximately 10 years. Figure 11. If fatigue damage is found, the appropriate corrective action is taken through release of an AD-mandated service bulletin applicable to all airplanes of the affected model. Periodic reports are provided to the manufacturers and airworthiness authorities regarding findings and their disposition.

Airplane		Number of candidate airplanes
Model	Series	
707/720	All	1417
727	100-200	178
737	100C	72
737	100-200	123
747	100-100F C-200B	117

707 candidate fleet grows with time

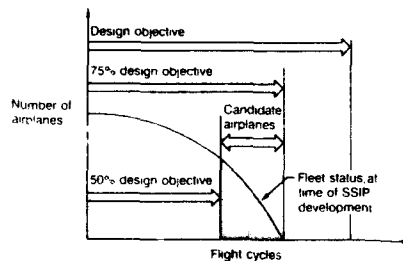


Figure 11. Supplemental Structural Inspection Programs

FLEET SURVEY PROGRAMS

Reported service data and other firsthand information from customer airlines are continually reviewed to promote safe and economic operation of the worldwide fleet. As a result of increasing costs and delivery times for airplane replacements, the active service usage for commercial airplanes has gradually been extended. The average age of the world airline jet transport fleet has increased from 8 to 12 years over the last 10 years. This upward trend in airplane operating age is likely to continue. Boeing design goals have traditionally been established for a conservative number of flights over a 20-year period. This timeframe seemed more than adequate in the 1960s to carry these airplanes throughout their passenger-carrying years. Today there are more than 1,000 Boeing airplanes around the world that are over 20 years old, and this number will double by 1995. Approximately 150 Boeing airplanes had reached their design flight cycle goals in 1990, and this number will increase to about 500 by 1995, representing about 10% of the active Boeing fleet. Figure 12.

Recognizing this trend, Boeing implemented a new program to better understand the condition of older airplanes. The Aging Fleet Survey Program was developed during 1986 and implemented in early 1987. The success of the program

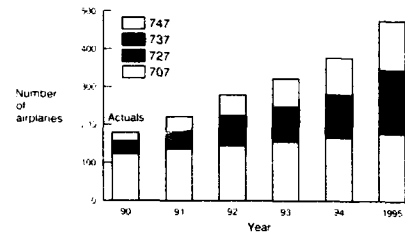


Figure 12. Projected Number of Boeing Airplanes Exceeding Design Goal in Flights

resulted in expansion to include new production airplane models such as the 737-300, 757, and 767, some of which have already seen 8 years of service. By reviewing the structural and systems performance of these airplanes during their early and maturing years, it is believed that Boeing and the operators can take the right actions to preclude the majority of aging fleet problems 10 years or so from now.

The specific objectives of these surveys are:

- Models 707, 727, 737, and 747:
 - Gain an engineering assessment of the condition of older airplanes with emphasis on structures and systems.
 - Observe the effectiveness of Boeing corrosion prevention features and other corrosion control actions taken by operators.
 - Acquire additional fleet data to improve maintenance recommendations and design of new airplanes.
- Models 737-300, 757, and 767:
 - Avoid large-scale aging fleet problems beyond design service objectives.
 - Gain an early identification of problems experienced in service, particularly for new model features.

Survey Procedures

Boeing survey teams are dispatched to observe airplanes during scheduled heavy maintenance checks. Six experienced structures, systems, maintenance, and service engineers record their observations in survey documents covering up to 350 structures and 150 systems items. Typically, 70% of the items are surveyed since access is not available to all areas during the visit. Each survey includes a review of airline practices regarding airplane use, maintenance program, and dispatch reliability. The operator, are briefed by the teams on their findings.

Airline acceptance of the program and cooperation with Boeing has been positive. Observing a significant number of airplanes in a variety of operating and climatic environments around the world provides a composite sample of each model and a better understanding of common and unique aspects within and between model fleets. As of March 1991, 122 airplanes owned by 67 operators have been visited in 33 countries around the world. Figure 13. Although the number of airplanes observed is a small percentage of the total fleet, it does represent about 15% of the high-time airplanes.

All significant findings pertaining to a specific visit are recorded and assigned to appropriate organizations for necessary fleet action. The collected data have been pooled for fleet evaluations to determine if there are trends requiring additional actions by Boeing and/or the operators. A number of detailed action items have resulted from the surveys, and their applicability has been reviewed across all Boeing models. To ensure anonymity, all identifiable operator/airplane data are treated as confidential.

Survey Findings

This section focuses on some of the significant findings that relate to airplane structures and systems. Considerable variation has been observed in both airplane condition and airline maintenance procedures, such as inspection intervals and corrosion prevention measures. The condition of the structure was generally good but considerably below expectations in a few cases. The following findings are worthy of note to illustrate observed variations:

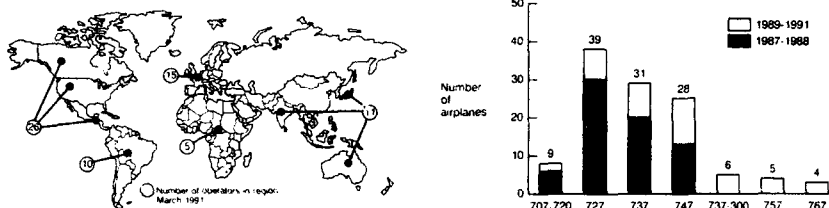


Figure 13. Boeing Fleet Survey Program Participants and Airplane Model Survey Distribution

- The incidence and extent of discrepancies, particularly corrosion, were more than would be expected on well-maintained airplanes. There is evidence that repair action was not taken in a timely manner. As a consequence, repair action was sometimes extensive and very time consuming. Figure 14. Many operators anticipate this likelihood and plan considerable corrosion-related repair activity at major maintenance visits. There is a general acceptance that airplanes typically require increased maintenance activity with age, and many operators have elected to shorten maintenance intervals.

Corrosion prevention and control measures must be aggressively pursued both to reduce the need for extensive repair and to promote continued airworthiness. New mandated corrosion controls and prevention programs developed in cooperation with airline working groups will enhance future condition of aging airplanes.

- Most observed fatigue cracking was previously known and action already identified to the operators by means of service bulletins. While a few new fatigue problems were identified, none were of immediate safety-of-flight concerns.
- The accomplishment of service bulletin action varies with airline maintenance practices and ranges from 20% to 80%. Service bulletins often give alternative compliance procedures in the form of repair or additional inspections. It was observed that airlines frequently chose the option to continue inspection rather than perform the specified repair action that would permit a return to normal maintenance inspection procedures and intervals. Mandatory service bulletin modification programs have been developed by industry task forces for aging Boeing models.
- There is concern over the number of repairs found in close proximity on some airplanes. The damage tolerance of the structure may be impaired in such circumstances even though each repair may be satisfactory in isolation. Incomplete removal of corrosion damage during repair is also frequently encountered. Unless extreme care is taken during corrosion removal, it is inevitable that the problem will recur.

A number of airplanes were found with improper modifications or repairs. Examples include excessive use of blind rivets and improper rivet patterns in primary structure. Recent industry task force activities have resulted in

industrywide procedures for classifications of repair to ensure continued airworthiness by prompt actions depending on repair circumstances.

- Some airplanes subject to short-term changes of owner/operator do not appear to receive adequate maintenance. Lack of continuity in maintenance seems particularly prevalent for leased airplanes. With the steady increase in leased airplanes in worldwide service, the average condition of the fleet could worsen unless steps are taken to ensure that these airplanes receive the required levels of maintenance. Adoption of the previous operator's maintenance program, which is often the case, may not always be appropriate, particularly if there are significant differences in the type of operation. Consolidated maintenance recommendations tailored to aging airplanes are currently under development in concert with industry task force initiatives.

- Observed airplane systems generally appear to be in good to serviceable condition. However, portions of some systems, especially those where performance of function is not dispatch critical, were observed to be in unsatisfactory condition. Similar discrepancies were reported on a number of surveys, and it was obvious that these discrepancies existed for a long period of time.

Significant corrosion of system components was limited to control system cables passing under or near lavatories. Minor corrosion, which had no effect on component performance, was found on some accumulator cylinders and landing gear hydraulic components.

The fleet survey program is continuing with 30 airplanes observed in 1990. Recent findings indicate the condition of the airplanes generally is good and that they are receiving adequate and conscientious maintenance. Most discrepancies noted by the Boeing teams had already been recorded by the operator and corrective action was underway. Recent surveys have confirmed our belief that the aviation industry is more aware of the necessity of applying timely corrosion control and installing well-engineered repairs and, generally, has a more heightened awareness relative to sound maintenance practices.

INDUSTRY AGING FLEET INITIATIVES

While aging aircraft issues have been the subject of considerable attention by industry and government for years, the 1988 explosive decompression of a 737 focused public attention on older



Figure 14. Fleet Survey Corrosion Findings on Fuselage Lower Lobe and Rear Spar Details

airplanes as never before. As with any other accident, it was the result of a combination of factors including design, inspection, and maintenance. Boeing had recognized the design problem for early model 737 airplanes and provided service information to their operators based on recent findings which would normally have been adequate to maintain safety. The FAA also recognized the importance of this problem, and issued an AD making compliance with this service information mandatory. The airline claimed compliance with the AD and all the criteria of "properly designed, inspected, and maintained" had seemingly been met, yet the accident still happened.

Aging fleet concerns have resulted in both specific Boeing initiatives and joint industry, airline, and airworthiness authority actions. Boeing formed a special Corrosion Task Force and held meetings with airline maintenance executives as a result of fleet survey findings described above. A conference on aging airplanes was held in Washington, D.C., in June 1988 that resulted in specific recommendations by the airlines and manufacturers. These recommendations included the establishment of a Steering Committee to guide the international aviation community actions with regard to aging aircraft. This Steering Committee is composed of manufacturers, operators, and regulatory authorities. This combined activity represents the Airworthiness Assurance Task Force (AATF) that is organized and run by the Air Transport Association (ATA).

The ATA, in concert with the Aerospace Industries Association (AIA), is managing AATF activities since responsibility for safe design, operation, and maintenance rests with each manufacturer and airline. The FAA and other major airworthiness authorities serve as guardians of aviation safety, monitoring the manufacturer's or airline's performance through inspections, providing uniform standards and guidance, and ensuring compliance through specific enforcement programs. While these agencies are directly involved in the Task Force, both from the design and maintenance standpoints, and is accountable for mandatory changes, the final responsibility for safety lies with the manufacturers and airlines.

The AATF charter is to develop all measures considered necessary to augment the existing system and ensure the continuing safety of aging airplanes. To accomplish this objective, the AATF sponsored industrywide Structures Working Groups for each airplane model in service. These working groups have five major tasks, as shown in Figure 15:

- Review existing service bulletins and propose mandatory terminating modifications to replace continuing inspection options.
- Develop mandatory corrosion prevention and control programs for each model.
- Update the supplemental structural inspection programs for detection of new fatigue damage locations in fleet leader airplanes.
- Provide industrywide guidelines for repair evaluations and corrective actions to ensure continued damage tolerance.
- Develop guidelines for establishing specific aging airplane maintenance requirements.

The Boeing working groups were formed in August 1988. Formation of the Douglas and Airbus, Convair, Fokker, and British

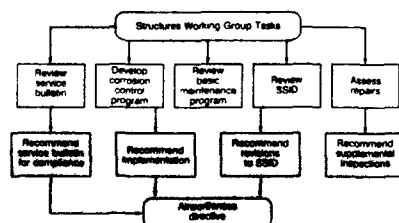


Figure 15. Structures Working Group Tasks

Aerospace working groups followed shortly thereafter. For each Boeing model, the working groups comprised representatives from approximately a dozen airlines around the world, Boeing specialists, and observers from the FAA and the Australian and British civil aviation authorities. The selected airline members operate a high percentage of the older models in their fleets. For example, the 727 is represented by 11 airlines who operate 923 of 1,766 airplanes in service today. The industry working groups have continually demonstrated a cooperative determination to make the right things happen—within models, across models, and throughout the industry. There have been impressive accomplishments on the Boeing models as described below.

Service Bulletin Reviews

As airplanes age, the incidence of fatigue increases and corrosion becomes more widespread. Problems are often addressed in isolation during the early service use of airplanes. With age, two or more problems in an area may degrade airplane structural fail-safe capability. This increases the need to incorporate preventive modifications in areas with known problems. The criteria for selection of service bulletins for high-time airplane modification are based on considerations such as safety problem potential, high probability occurrence, and difficulty of inspection.

A candidate list of service bulletins was established by Boeing as a baseline after a thorough review of those applicable to long-term operation. These service bulletins were reviewed by the respective working groups for recommended terminating actions. The thresholds for these mandated repairs and modifications were typically selected as the design objective in flight cycles for fatigue-related problems. Earlier calendar time thresholds were necessary for items driven by corrosion or stress corrosion considerations. Figure 16 shows the resulting service bulletins for which mandatory modifications are recommended. These selections were guided by a rating system developed by working group members to reflect their own experience.

Model	Candidate bulletins	Mandatory modification recommended	Other action	No status change
707/720	196	141	24	31
727	113/129	74/85	16/19	23/25
737	80/89	58/61	17/20	5/8
747	83/90	31/32	27/30	25/28

Structures Working Groups Recommendations. Original/1990

Figure 16. Recommended Mandatory Service Bulletin Modifications

Using the 727 as an example, 113 candidate service bulletins were reviewed in detail. The working group initially selected 74 bulletins for mandatory modification action. Another 16 were selected to receive more stringent inspection requirements. Compliance involves a major investment in operator resources—approximately 22,000 man-hours for each of the early production line airplanes, as shown in Figure 17. Later airplanes, particularly after line position 735, incorporated many of the changes during production, and therefore require significantly less modification. Compliance is required when the airplane is 20 years old or reaches 60,000 flight cycles. About 940 model 727 airplanes will require partial or complete modification before March 1994. Similar commitments will be required for other Boeing models.

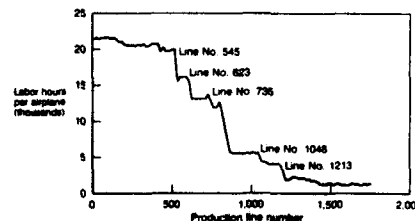


Figure 17. 727 Service Bulletin Modification Effort for Airplanes Exceeding 60,000 Flights

Aging fleet service bulletin summary documents were released in 1989 for each model formalizing Structures Working Group (SWG) recommendations for mandatory modifications or inspections. The details of each modification or inspection and the affected airplanes are described in applicable service bulletins. The summary documents serve as a record of SWG recommendations and as a reference for the airworthiness directive actions. Airworthiness directives were issued in 1990 for incorporation of structural modifications listed in these documents upon reaching the thresholds specified or generally within 4 years after the effective date of the AD, whichever occurs later. Annual reviews are conducted to update the existing program and evaluate any new candidate service bulletins released since the last SWG review. The first annual service bulletin review (held in June through August 1990) resulted in 16 new modification recommendations. After each annual review it is anticipated that separate airworthiness directives will be issued against new recommended actions contained in updated summary documents.

Corrosion Prevention and Control Programs

While corrosion has always been recognized as a major factor in airplane maintenance, each airline has addressed it differently according to its operating environment and perceived needs. Manufacturers have published corrosion prevention manuals and guidelines to assist the operators, but until now there have never been mandatory corrosion control programs.

In the late 1970s, when Boeing was developing fatigue-related SSIDs, a basic assumption was made that the existing approved maintenance programs were controlling corrosion below levels that could affect airworthiness. Therefore, the resulting SSID programs centered around controlling the anticipated increase in fatigue damage that would occur as the fleet aged. However, the Boeing fleet surveys revealed that some operators did not utilize proven or effective corrosion prevention measures. In addition, some instances of very severe corrosion were observed reflecting improper or delayed prevention and repair actions.

It became apparent that without effective corrosion control programs, the frequency and severity of corrosion were increasing with airplane age and, as such, corrosion was more likely to be associated with other forms of damage such as fatigue cracking. This, if allowed to continue, could lead to an unacceptable degradation of structural integrity, and in an extreme instance, the loss of an airplane. As a result, Boeing formed an inhouse Corrosion Task Force in 1988 and the AATF chartered the SWG to develop mandatory corrosion prevention and control programs.

The Boeing Corrosion Task Force reviewed all Boeing sources of information related to known corrosion problems. All problems relating to principal structural elements (PSE) were retained and segregated into selected general areas on the basis of having similar corrosion exposure characteristics and/or common inspection access requirements. Figure 18. Problems found to be significant in relation to continuing airworthiness were included in the program as specific tasks unless already covered by an existing airworthiness directive. It was recognized that corrosion growth rates varied widely and it would be unduly conservative to establish a program based on the most severe operating environment. Therefore, the approach taken was to develop a baseline program that represented minimum requirements for typical operators. Individual operators who would experience significant corrosion after applying the baseline program must then modify or improve their program. The Boeing Corrosion Task Force developed a proposal for the baseline program based on existing recommendations, modified by current experience and knowledge gained by their review of available data.

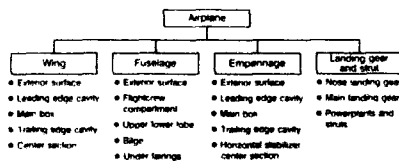


Figure 18. Corrosion Program Areas

The working groups have recognized the need for a universal baseline minimum corrosion control program for all airplanes to prevent corrosion from affecting airworthiness. Furthermore, they are working toward maximum commonality of approach within and between each manufacturer's models to ensure consistent and effective procedures throughout the world.

The Boeing programs were developed first and presented to the FAA in late 1989, and model-specific ADs were released by the end of 1990. Meanwhile industrywide working groups are conferring to ensure a common approach is achieved in both the mandatory and advisory (guideline) parts of the implementing documents. The program requirements apply to all airplanes that have reached or exceeded the specified implementation age threshold for each airplane area. The specific intervals and thresholds vary between models, but all programs follow the same basic philosophy and typically contain the following:

- External and internal inspections of all airplane structural areas are required at specified implementation times and repeat intervals. The program will require major opening up of the structure at these inspections. Figure 19 illustrates the required access to the 727 fuselage. It further details preventative measures including repair action and assurance that drain paths are clear, protective finishes are reapplied, and corrosion inhibiting compounds are applied.

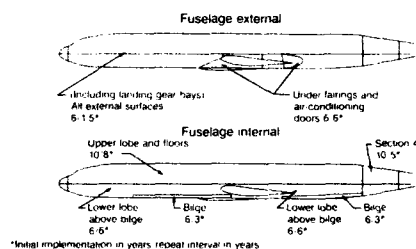


Figure 19. 727 Corrosion Control Program Example

- Corrosion damage must be controlled between maintenance visits to acceptable minimums that will not adversely affect safety. The baseline program must be adjusted if necessary to achieve this standard.
- All cases of corrosion exceeding the minimum level must be reported, with particular emphasis on corrosion that raises an immediate safety concern. This will enable rapid response throughout the fleets to inspect and correct any potential problems.
- Intervals and implementation thresholds are based on area- and model-specific calendar times, Figure 20.
- The maximum period for implementing the program fleetwide in a given structural area is one repeat interval (not to exceed 6 years if over 20 years of age and a minimum rate equivalent to one airplane per year).

Many operators are already incorporating several corrosion program features in their current heavy maintenance visits or when accomplishing the service bulletin modification described earlier. Such preimplementation provides valuable early feedback about the effectiveness of the program and further demonstrates the operators' responsiveness and commitment to the true spirit of safety. Boeing has also initiated extensive training programs that will be available to airline and airworthiness authorities personnel alike to ensure efficient corrosion prevention and control program implementation.

Supplemental Inspection Program Reviews

SSIDs were initially released between 1979 and 1983 for models 707, 727, 737, and 747. Their purpose was to ensure continued safe operation of the aging fleet by timely detection of new fatigue damage locations. These documents have been updated on a regular basis to reflect service experience and operator inputs. In the light of current aging fleet concerns, the AATF directed the

General area		707/720		727		737		747	
		Threshold	Repeat	Threshold	Repeat	Threshold	Repeat	Threshold	Repeat
Wing	Outer-external	10	4	10	5	8	4	10	2
	Leading edge interior	8	2/4	10	5	8	4	6	1.5
	Outer-main box-interior	10	8	10	10	10	10	20	10
	Trailing edge interior	8	2/4	10	5	8	2	10	2
	Center section interior	10	8	10	8	10	8	20	10
Fuselage	External (including doors and landing gear bays)	6	2	6	1.5	5/8	1.5/2	10 upper 5 lower	5 upper 2 lower
	Flightcrew compartment	10	8	10	8	10	8	15	8
	Upper lobe interior	8	8	10	8	8	8	15	8
	Lower lobe interior (except bilge)	6	6	6	6	6	5	6	6
	Lower lobe - bilge	6	3	6	3	6	2/4	6	4
	Section 48 interior	10	5	10	5	8	4	10	5
V/H Stabilizer	External surfaces	10	2/4	10	2	10	2	10	5
	Leading edges	10	8	10	8	10	8	15	8
	Main box interiors	10	8	10	8	10	5	15	8
	Trailing edges	10	8	10	4	10	5	10	5
	Center section	10	5	10	5	8	4	10	8
Center engine inlet duct		-	-	10	8	-	-	-	-
Nose and main landing gear		Landing gear overhaul		Landing gear overhaul		Landing gear overhaul		Landing gear overhaul	
Powerplant and strut		4	2	5	2	5	5	7/15	3/15

Note: Some specific areas/items within the general areas have independent thresholds and repeat intervals

Figure 20. Corrosion Inspection Thresholds and Inspection Interval, Years

SWGs to review these inspection programs to ensure adequate protection of the aging fleet. The major focus of these reviews conducted during 1989 and 1990 was:

- Adequacy of the present fleet leader sampling.
- Inclusion/deletion of PSEs.

The initial candidate fleet leader samples comprised those airplanes exceeding 50% of the design objective in flight cycles when the typical fleet leader reached 75%. These criteria resulted in 450 model 727, 123 model 737, and 117 model 747 subject to SSID compliance. Boeing periodically reviews the candidate airplane list for any significant changes in fleet distribution, composition, or utilization. To date, only minor changes have occurred in the active candidate airplane fleets, although some non-candidate airplanes with higher flight cycles have overtaken candidate airplanes.

Some PSEs were not included in the original SSID on the basis that damage would be obvious before safety was affected. A review of those items resulted in adding several items to the SSID, primarily some hidden wing structure previously deleted on the basis of fuel leaks to signify fatigue damage.

Thin gauge fuselage structure was not included in the SSID on the basis of test and service evidence that skin cracks would turn at frame locations and result in a safe decompression; see Figure 5. Consideration of aging fleet damage in adjacent bays prompted coverage of thin gauge fuselage structure, 0.056 in (1.4 mm) thick or less for models 727 and 737. The 747 fuselage skins were already included in the initial SSID because of thicker gauges.

Much concern has been expressed recently regarding possible widespread fatigue cracking, a phenomenon where a patch or group of multiple small cracks of varying sizes in adjacent holes simultaneously join to form a single crack of longer combined length. This results in a substantially reduced timeframe to safely detect the cracking. The SWG concurred that the SSIDs should include considerations for structure susceptible to that form of cracking with appropriate changes of damage detection periods and inspection intervals.

The original SSIDs allowed credit for detection opportunities based on secondary cracking. Allowing detection credit for secondary skin cracks may be unconservative, especially if the majority of the detection credit were to be derived from external inspection of the skin. For example, when a fuselage frame cracks the next crack may occur in the adjacent frame rather than in the skin as was assumed. It was agreed that the SSID should be reviewed and revised to cover adjacent member cracking patterns wherever they were likely to occur.

Revisions to 727, 737, and 747 SSIDs will be issued by mid-1991 and include changes to approximately 10 significant structural items for each model. The SSID for model 707 was first released

in 1979 and therefore provides less sophisticated inspection options. However, the basic information and approach are similar to those for later models. Revisions to this SSID that incorporate service bulletins for areas with known fatigue cracking will also be completed in 1991.

Structural Repair Assessments

Inevitably, airplanes accumulate repairs. For each model, structural repair manuals (SRM) assist the operator in ensuring that typical repair action maintains the airframe structural integrity. Other larger repairs are handled by individually prepared and approved engineering drawings. Traditionally, these repairs have primarily focused on static strength and fail-safe aspects of the structure after repair, with common sense attention to durability considerations. For several years, however, there has been an additional emphasis on the need for structures to be damage tolerant. Achieving damage tolerance demands knowledge of potentially critical structural elements, an understanding of damage growth and critical size, and an inspection program to ensure timely detection.

Repairs may affect damage tolerance in different ways. An external repair patch on the fuselage can hide primary structure to an extent that supplemental inspections may be required, as shown in Figure 21. Other repairs may interfere with obvious means of detecting damage such as skin repairs on the lower wing with sealant that prevents fuel leakage. Repairs located in low stress areas with slow crack growth rate can have damage tolerance provided by existing maintenance.

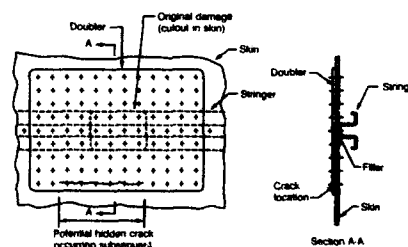


Figure 21. Typical Fuselage External Skin Repair

The objective of the SWG repair review process is to provide the airlines with a practical guidance material (based on damage tolerance principles) that allows repairs to be evaluated by operators without complex analyses. This process is directed toward showing that an installed repair, which matches the strength and durability of the original structure, is found damage tolerant if the

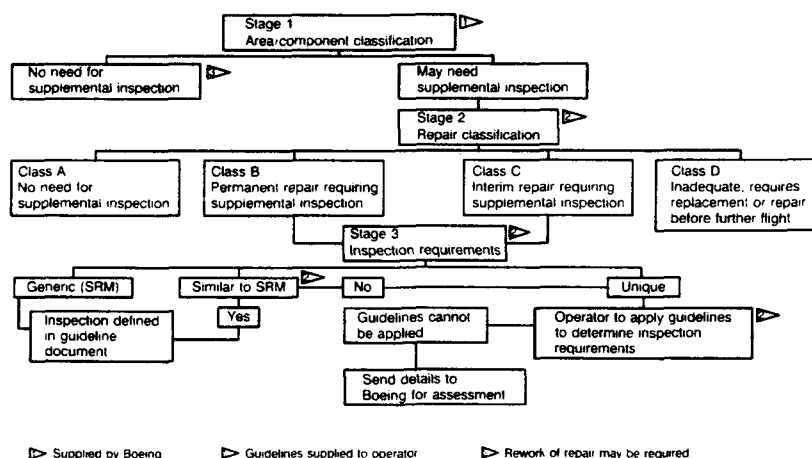


Figure 22. Proposed Repair Assessment Logic

existing inspection program is adequate. The evaluations provide inspection requirements in terms of inspection methods, threshold, and repeat intervals and will be added to SRMs.

One of the tasks in the SWG review process is to coordinate with industry to develop a common approach. After meetings with other manufacturers, industry agreement has been reached on categorization into four classes of repairs, as shown in Figure 22. Agreement has also been reached on a list of minimum required data for classification. A list of optional data (from existing records) that would give additional credit for repair quality has also been developed. A common survey form has been developed for operators to use recording the data needed to classify and assess repairs.

The general procedure for establishing both inspection and replacement thresholds is based on service history, test results, fatigue analyses, and design criteria. The goal is to present options in SRM tables and graphs. This will allow both the operator and manufacturer to determine inspection requirements for a large number of repairs in an efficient and consistent manner or if the repairs are unique and not covered by the SRM. The repair documentation would be sent to the manufacturer for evaluation if the operator is not able to apply the guidelines.

Results from the working group activities are scheduled to be published in model-specific SRMs in late 1991. Once the guidelines are approved, operators will be required to examine, classify, and document repairs to principal structural elements. In addition, operators will be required to determine and implement supplemental inspections or other actions necessary to maintain structural integrity. Again, as a pro-active measure, many airlines are already documenting the location and details of existing repairs as they conduct routine maintenance.

Structural Maintenance Guidelines

Modern transport category airplanes are designed to meet continuing airworthiness requirements indefinitely provided structural integrity is maintained by effective inspection and corrective maintenance programs. Comprehensive maintenance program guidelines that properly address older airplanes do not exist. Supplemental structural inspection programs, for example, only address fatigue cracking. Therefore, the AATF chartered an industry subcommittee to establish guidance material for developing aging aircraft maintenance programs in two parts, general and model-specific guidelines, shown in Figure 23.

The general guidelines were developed by an industry task group and submitted to the AATF in June 1990. The general guidelines cover a wide range of generic topics related to the maintenance of aging airplanes. These topics include:

- Escalation of maintenance check intervals.

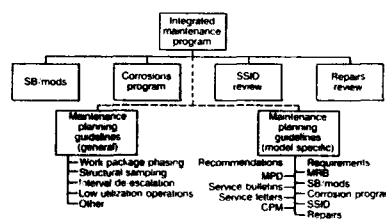


Figure 23. Aging Airplane Maintenance Program Development

- Escalation of total scheduled downtime.
- Increases in workload/manpower requirements.
- Number of discrepancies between successive checks.
- Expected trends in the ratio of nonroutine to routine maintenance work.
- Documentation, record keeping, and reporting.
- Effects of changes in airplane utilization.
- Qualification and training of maintenance personnel.
- Maintenance facilities and equipment.
- Phasing of structural maintenance checks.
- Sampling of age exploration as it relates to the aging fleet.
- Deferral of permanent structural repairs.
- Maintenance program change considerations.

The model-specific guidelines provide information for establishing structural maintenance programs that meet or exceed the mandatory requirements for that particular model. They will be developed by the manufacturer and include all existing mandatory requirements (including those resulting from aging fleet SWG activities) as well as other economically significant recommended maintenance actions. The intent is to provide operators with a single comprehensive source that references all significant maintenance recommendations. The current schedule is to have the model-specific structural maintenance program guidelines completed by the end of 1991.

FOCUS ON THE FUTURE

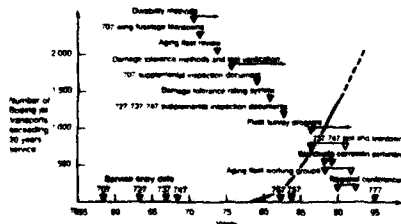
A better understanding of how airplanes withstand the rigors of long-term use will help future generations of commercial airplanes to be safer and less maintenance-intensive with age. Just

Boeing has recently developed a comprehensive Corrosion Design Handbook reflecting fleet experience to provide the structural engineer with the same corrosion prevention expertise that parallels methods used to develop producible, durable, and damage-tolerant structures. Figure 24. Similarly, improved structural arrangements and concepts will enhance the inherent robustness and forgiveness of the structure, facilitate simpler repairs when damage occurs, and facilitate accessibility and inspectability. The knowledge gained in the past 3 years will also enable better focus on the initial and continuing maintenance needs of new airplanes. In turn, this will allow the most effective and timely distribution of airline resources to maintain their airplanes indefinitely to achieve continued safe and economic operation.



Finally, the FAA and other airworthiness authorities will be better prepared, informed, and trained as they establish standards and monitor fleet performance. The growing commitment to "hands-on" airplane monitoring will strengthen and cement the three-way partnership in structural safety assurance.

Boeing is dedicated to design and manufacture safe commercial jet transports. The successful accomplishment of this responsibility over the last three decades has contributed significantly to a position of industry leadership and reflects the top priority given to safety. This paper illustrates the structural integrity assurance of commercial airplane structures is a serious and disciplined process. Figure 25. High standards must be maintained to ensure the safety of aging airplanes until economics dictate their retirement. Standard Boeing practices to ensure continuing structural integrity include providing structural maintenance programs, continuous communication through customer support services, and recommendations for maintenance actions through service letters, structural item interim advisories, and service bulletins.



To help identify potential problems associated with the aging jet transport fleet, Boeing has implemented additional activities:

- Supplemental structural inspection programs that require airlines to regularly inspect structurally significant items on selected older airplanes and report defects to Boeing for prompt fleet action.
- Teardown of older airframes to help identify corrosion and other structural safety defects.
- Fatigue testing of older airframes to determine structural behavior in the presence of service-induced problems such as corrosion and repairs.
- An engineering assessment of the condition of a representative sample of older Boeing airplanes to observe effectiveness of corrosion prevention features and acquire additional data that might improve maintenance recommendations to the operators.

Aging fleet concerns have also resulted in joint industry, airlines, and airworthiness authority actions. Task forces consisting of representatives of airlines, Boeing, and the FAA have addressed the following:

- Selection of service bulletins for which structural modifications *should be made mandatory at some threshold*.
- Development of mandatory corrosion inspection, prevention, and repair programs.
- Reviews of the supplemental structural inspection programs for completeness and clarity.
- Development of guidelines to determine the damage tolerance adequacy of structural repairs.
- Development of comprehensive maintenance guidelines for older airplanes in the fleet.

These initiatives have provided timely preventive structural maintenance recommendations and permit continued safe operation of aging jet transports until their retirement from service.

The design, construction, operation, and maintenance of airplanes take place in a changing and dynamic arena, with new technology, new needs, and new players. The structural safety system may never be perfect, but has produced an enviable record and the aging fleet initiatives will measurably improve that record. If the lessons being learned today by the manufacturers, the operators, and the authorities are properly reflected in our next-generation airplanes, they should fly longer and safer with progressive maintenance that ensures continued structural airworthiness until retirement from service for economic reasons.

The authors are greatly indebted to several Boeing Structures Engineering and Maintenance specialists whose contributions made this presentation possible. Technical contributions by J. Hall, P.J. Harradine, K.O. Henzler, G.J. Houlihan, M. Miller, and J. Nychay are truly appreciated. Graphics support coordination by W. Brewer and R. Lam enhanced the end product significantly.

Goranson, U. G., and Miller, M., "Aging Jet Transport Structural Evaluation Program," 15th Symposium of the International Committee on Aeronautical Fatigue (ICAF), Jerusalem, Israel, June 21-23, 1989, published by Engineering Materials Advisory Services Ltd. (EMAS), London, England.

"AIRCRAFT FATIGUE MANAGEMENT IN THE ROYAL AIR FORCE"

Presentation to the 72nd AGARD Structures Specialist Meeting

on "Fatigue Management"

by

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OPENING REMARKS

Introduction

Good ... ladies and gentlemen. I am Squadron Leader Mike Render from the Royal Air Force currently serving in the Ministry of Defence in London. The presentation today on 'Fatigue Management in the Royal Air Force' is intended to give you an overview of how fatigue is managed for all our aircraft fleets (see Figure 1). Much of our fatigue management policy affects my colleague - Squadron Leader John Stevens of the Maintenance Analysis and Computing Division at RAF Swanton Morley - so we felt it only fair that he too should say a few words as part of this presentation. You will note that we use the Tornado aircraft as our main theme, relevant in that the aircraft represents the largest proportion of the capital value of our inventory.

Reasons for RAF Aircraft Fatigue Management

We set ourselves 3 objectives within the Royal Air Force for our aircraft structural integrity policy (see Figure 2) to make aircraft operations as safe as is reasonably possible; to ensure that the aircraft are available to the front line; and to minimise the costs to the United Kingdom taxpayer.

Let me just dwell on these objectives to put them in context for today's presentation. You will note that they are inter-related.

Safety

Clearly, safety has always been paramount and in peacetime we must adhere to strict rules - not just on static strength but on fatigue life too. Accordingly, our senior engineer, the Chief of Logistics Support - or CLS(RAF) - has a remit to assure our Controller Aircraft that we in the RAF are guaranteeing the airworthiness of our fleets. In turn, CLS delegates the airworthiness requirement to those who are capable of managing the airworthiness of their fleet on a day to day basis.

Aircraft Availability

To fail to make aircraft available to the operators at the front line is a shortcoming which we cannot tolerate. Quite apart from the credibility of RAF engineers in the eyes of our aircrew, it is essential, with front line jets costing £25M (US \$45M) apiece, that we make best use of the Defence Budget.

Financial Implications

Financial considerations have never before been so important and so we are nowadays feeling the pinch - as indeed are our colleagues. We have been forced to keep many of our front line

aircraft in service for much longer than was intended at the design stage. We cannot afford always to buy new ones and so we keep them operationally viable by updating their avionics. Usually our aircraft are scrapped only when they run out of fatigue life rather than when they become obsolescent. Fatigue management is therefore something that we are putting considerable effort into at present.

Case Summary of Selected Fleets

We can quote 3 examples of where we have failed to meet the objectives of our structural integrity policy.

Firstly, the Buccaneer: after the 2 catastrophic accidents about 10 years ago the Buccaneer fleet was grounded for 6 months; one third of the fleet was scrapped; and the technicians still face a massive penalty in the form of frequent time-consuming inspections of the structure.

Secondly, the Hawk: we are having to re-wing all the Hawks at only two thirds of the way through their designed life. That will cost £50 million without considering the down-time of the fleet.

Thirdly, the Tornado: there are some substantial fatigue-related modifications which were not bargained for at the time of introduction. The total down time for these is over 200 working days - that means that every single Tornado will have spent a year on the hangar floor, over and above the time spent on normal servicing - and that is before the aircraft have reached one sixth of the way through their life.

Infrastructure

We have a tiered management structure for the implementation of policy (see Figure 3). At its peak, the Fixed Wing Structural Integrity Working Party, consisting of one-star officers or their Civil Service equivalents, directs the policy on behalf of CLS(RAF). At the level below are Structural Integrity Working Groups - one for each aircraft - in which the detailed discussion on the structural health of the aircraft takes place, including all matters relating to fatigue. In turn, these Groups are accountable to the one-star committee. Specialist attendees at the Group, which are chaired by the MOD Procurement Executive Project Office, include: the Design Authority; the Support or Engineering Authority; the Air Staff; and the Materials and Structures Department of the Royal Aerospace Establishment, Farnborough. As necessary, specialists from our Maintenance Analysis and Computing Division and our Central Servicing Development Establishment attend to offer detailed advice. The Ministry of Defence policy branch for aircraft structural integrity (Air Eng 17), to which I belong, is also represented at these Group meetings. The topic of fatigue management therefore gets

a thorough airing by all interested parties. It is important to note here the objective viewpoint that the structural integrity policy branch holds since we are not responsible for the day to day management of the individual aircraft and therefore are not clouded by the short term priorities the Support Authorities may face. Such a philosophy continues to be endorsed by our senior management even in these times of staff cuts.

Statements of Operating Intent (SOIs)

In order that the Design Authority can understand fully how we are operating the aircraft, we produce a Statement of Operating Intent or SOI (see Figure 4) one for each type. This publication gives a detailed account of each of the flying profiles (see Figure 5) we expect to use in the following 2 years, supplemented by relevant information on for example, fuel loads and stores configurations. The Design Authority will therefore be able to underwrite the structural integrity of the aircraft with a real rather than assumed knowledge of how the aircraft are flown.

Damage Recording

By monitoring the parameters most related to the fatigue damage we can attempt to assess the fatigue life consumed in each sortie. Our most universal way, for the next few years at least, is by means of a fatigue meter (see Figure 6) This mis-named device is but a counting accelerometer measuring g levels in the pitching plane near the centre of gravity. The shortcomings of such a device are well known, of course, but with care we can make reasonable estimations.

Fatigue Formulae

It is from the detail within the SOI that the Design Authority will be able to determine the relationship between the fatigue meter readings and the actual fatigue damage caused by the manoeuvres. This relationship is expressed by means of a formula (see Figure 7), perhaps a number of them to account for different stores configurations. Each formula is so constructed such that the cleared life of the aircraft - either anticipated or already underwritten by fatigue testing - equals 100 fatigue index or 100 FL. All concerned can therefore easily relate the life used so far as a proportion of the cleared life. Simplistic of course but it does afford a greater visibility of the fatigue issue to all parties.

Operational Loads Measurement (OLM) Programmes

No matter how accurate the SOI may be and how thoroughly the fatigue formula may have been derived, there will always be a difference between the assumed fatigue damage and that actually incurred. For the last 15 years we have therefore had Operational Loads Measurement (or OLM) programmes wherein aircraft are fitted with strategically placed strain gauges to determine the loads directly (see Figure 8). Information thus derived is used to supplement the fatigue formula results and we can adjust the calculations as necessary. Providing the OLM programmes are properly managed the benefits are well worth the investment. As a very useful extra, the OLM results can let us know the damage caused to structure which could not otherwise be determined. Fatigue damage in the tail and fin, for instance, often bears little relation to the fatigue meter readings.

Basic Introduction to Tornado - Design Philosophy

Our combined Tornado fleet amounts to about 300 aircraft divided between 2 variants - the Interdictor Strike (IDS) and the Air Defence Variant (ADV). Structurally the variants are 80% common, the main distinction being the extended fuselage of the ADV. Both were designed to a safe life of 4000 hours. Although there are some individual component tests, essentially each vari-

ant is underwritten by a main airframe fatigue test, the IDS test at IABG in Munich and the ADV at British Aerospace Warton. All 3 partner nations of the Panavia consortium draw results from the IDS test. Germany and Italy have no direct interest in the ADV results since the UK alone operates this variant - although the other nations still have responsibility for the ADV test under the terms of the Panavia contract. The fatigue safety factors adopted by us differ from those of our partners, as do the fatigue management principles. That said, it is not my place to dwell on the matters relating to our partner nations. Suffice to say that we adopt 3.33 for monitored structure and 5 for unmonitored structure. Monitored structure is that whose damage can more easily be aligned to the readings on the fatigue meter and is therefore lifed in FL terms. Unmonitored structure is that whose life is usually assessed purely by the number of hours flown. In peacetime the safety factors are those which we must adopt in order to retain the acceptable probability of failure from fatigue arising during the life of the aircraft. Hence, in all but the extreme circumstances, we cannot fly beyond the limits that the safety factors dictate. If we do, we are usually faced with a fly-by-inspection regime until we can reach the next servicing opportunity. I must emphasize that such events are rare - not even during Operation Desert Storm did we face such a situation. I should also add that our Air Staff are quite content with this approach - providing we do not change the rules, they are happy to take them at face value. Bearing in mind the safety factors, we have cleared nearly a full life for the IDS and about half a life for the ADV. Partly due to the sheer size of the fleet and partly because of the complexity of structure on what are our foremost fast jet aircraft, the fatigue management aspects on the Tornado represent our most advanced thinking across all the Royal Air Force. Suffice to say, if we get it right with Tornado, then the chances are good with any other aircraft. Indeed, we often use Tornado as a prototype for any new ideas we wish to develop.

Having set the scene, then, I will let John Stevens take up the discussion and explain how we track our aircraft and component fatigue data.

MACD FATIGUE MONITORING

Good Ladies and Gentlemen, I am one of the five Systems Analysts responsible for Aircraft Fatigue Systems at the Maintenance Analysis and Computing Division (MACD) which forms part of the Royal Air Force Logistics Establishment (see Figure 9). Our aircraft fatigue system is centrally controlled by MOD Air Eng 17(RAF) as the sponsor branch responsible for aircraft structural integrity and fatigue policy. In conjunction with the appropriate Support and Design Authorities, they provide MACD with fatigue calculation formulae, approved by RAE Farnborough.

The objectives of the MACD aircraft fatigue system are twofold: Firstly, to monitor the damage experienced by individual service aircraft each time they fly, and secondly to identify any trends in operational fleet usage which deviate from the profiles defined in the Statement of Operating Intent.

Some 12 aircraft types are currently processed at MACD (see Figure 10). Under present policy all fatigue ADP processing is to be centralized at MACD with priority being given to fast jet aircraft types. The MACD fatigue system comprises nearly 3 million individual flight records covering over 1200 aircraft and more than 4000 major structural components. We are also currently developing systems for NIMROD, SENTRY and HARRIER GR7.

Tornado

Tornado represents by far the most complicated of our current systems.

Formulae

The Tornado IDS has no less than five independent formulae associated with critical areas of structure (see Figure 11) each of which may dictate the fatigue life of an individual airframe. Three formulae relate to the fuselage frames, the remaining two formulae apply to the wing pivot and inboard pylons.

Configuration Cases

Each of these 5 formulae is further divided into 4 "Configuration Cases" to reflect wing load distribution (see Figure 12). As Tornado OLM data analyses are completed, factors are applied to adjust these formulae; we are then required to reprocess all Tornado archive data, applying the revised formulae and factors.

Component Fatigue

In addition, the Fatigue Index consumption of individual critical components is also calculated (see Figure 13). Tornado has essentially become modular in concept for component management. There are presently nine major Tornado components each of which may migrate from one aircraft to another and we must therefore transfer with it the complete previous fatigue usage history in order that subsequent fatigue formulae reprocessing exercises are applied to the correct component usage data. Tornado wing management is further complicated by corrosion, repairs to which generate increased fatigue consumption rates for individual wings.

AIRCRAFT AND COMPONENT TRACKING

Mod Form 725

At the end of each sortie Fatigue Meter readings are entered on the MOD F725 "Flying Log and Fatigue Data Sheet" (see Figure 14) together with other flight details such as fuel load, weapons configuration, and so on. In order to track individual aircraft and components, identification of the aircraft and its major components is included within this F725 data. Whilst operational flying data is predominantly captured at unit level by such manual recording on the F725, we are increasingly converting to magnetic media capture utilising microcomputer terminals; Development of interfaces with automatic "on-board" aircraft fatigue data capture systems is also in hand.

SEMA

Where flying stations have local microcomputer systems, such as the Station Engineering Management Aid (known as SEMA), the equivalent F725 flight data is captured directly by input to the local terminal and subjected to validation to achieve a high level of data integrity (see Figure 15). At the Tornado units SEMA systems provide fatigue calculation and generate local management information prior to data transfer on magnetic disc to the MACD mainframe system.

Data Processing

Data from all Tornado flying units is then aggregated and the fatigue life consumed during each flight is calculated utilising the design authority formulae; global fleet management data is then generated either in hard copy form or on magnetic disc for transfer to external microcomputer systems (see Figure 15).

Fatigue Data Outputs

Fatigue system outputs in tabular and graphic printout form are forwarded on a monthly and annual basis to those external

agencies appropriate to the relevant aircraft type (see Figure 15). We have also developed automatic analysis of data such as aircraft mass, weapon configuration and g spectra, in order to review the accuracy of the Statement of Operating Intent. Ad hoc analysis of fatigue data is also carried out in response to requests from users. Detailed interrogation down to individual aircraft sortie parameters is available. Our customers are no longer restricted to RAF support authorities and operational units. Many of our ad hoc interrogations support OLM data analyses or fatigue formulae development work at Design Authorities and RAE Farnborough. For example, during Operation Desert Storm, the Design Authority were concerned at the high flying hour usage of Tornado ADV in relation to the cleared life of a particular weapon launcher. By analysing all Desert Storm Tornado ADV weapon configuration and fatigue meter data, MACD demonstrated that the actual operational 'g' spectrum was more benign than the design criteria and the integrity of the equipment was no longer in doubt. This was achieved within 24 hours of the original request.

Graphical Output

In order to assist fleet managers and other users in identifying trends more rapidly our system development was aimed at improving the analysis and presentation of fatigue data. We have therefore developed outputs in graphic form (utilising ICL software) to facilitate more rapid identification and analysis of fatigue trends than is possible with tabular data outputs. Graphic data presentation not only presents data in a format that is readily assimilated by the recipient but has the additional benefit of significantly reducing the volume of our reports. (For example, a single graphic of 'g' spectra has replaced 200 pages of report text).

Microsystem Interface

We have developed data formats which are compatible with external microcomputer systems such as that used by the MOD Tornado Support Authority (see Figure 15); mainframe data, transferred via floppy disc, enables Tornado staffs to manipulate data locally and to present outputs in whatever format they might require. Similar interfaces are being implemented on other aircraft fleets.

In summary, it is this combination of graphical presentation, together with the use of relatively powerful microcomputer systems at operational units and Support Authorities which offers the greatest potential improvement in RAF fatigue monitoring and analysis of aircraft usage indeed without such improvement fleet management of Tornado for example might well have proved impracticable.

Having described our current fatigue systems, I will ask Mike Render to conclude with an outline of the future way ahead.

DISSEMINATION OF FATIGUE MANAGEMENT POLICY

Towards the Future

I mentioned earlier that the fatigue meter is the best system we have for a while, albeit supplemented by our OLM programmes. In 3 years time we should have the beginnings of a fleet of aircraft fitted with strain gauges. The Harrier GR7, a variant of the AV8B, will be retrofitted with the Fatigue Monitoring and Computing System or FMCS. The measurement of fatigue damage using FMCS is no different a philosophy to that on the OLM programmes, except that every aircraft will be so fitted; calculation of the damage is done on board and is therefore available to the staff at the operating units. We favour such a system for the European Fighter Aircraft and we may retrofit Tornado - although with such a mature aircraft there may be insufficient time to

amortize the costs. We still have difficulty in convincing our seniors that FMCS does not reduce the amount of fatigue damage, rather it allows a better assessment of it to be made and therefore, indirectly, may afford the fleet managers the information to decide how best to conserve fatigue to within the budget.

Management Information

Improvement in our management-information methods is a worthwhile objective in itself. We believe that they can help overcome the disadvantages inherent in our fatigue systems of earlier eras of technology. We started to use colour graphics outputs for the processed fatigue data about 5 years ago, thus making the information readily digestible. Since then, there has been a sharp increase in the interest shown by all fleet managers - engineers and aircrew. The transfer of fatigue data by modem to Command and Ministry staffs is also under consideration. We are maintaining considerable momentum in these areas since there are vast potential savings from giving the managers the right information in the right format.

Education to Air and Engineering Staffs

We have embarked on a major programme of education, tailored specifically to fatigue management. Our efforts have been well received especially on the fleets that have stringent fatigue budgets. Our audiences for our lectures vary from senior aircrew to our engineer contemporaries who may be about to fill a Support Authority post. We are keen to maintain the 2-way dialogue and strive not to be misunderstood as a policing function, a mistake made by our predecessors with a consequent diminishing of trust between engineers and aircrew.

Manual of Structural Integrity

We are conscious that much of the good work may be lost over time with the inevitable personality changes. A Royal Air Force Manual of Structural Integrity is therefore in course of production. This will not only document the techniques which we have found favourable but also itemize, in considerable detail, the necessary management structure to cope with the fatigue management task.

Close

So, ladies and gentlemen, in the short time available we have attempted to give you a flavour of how we in the Royal Air Force are managing the precious resource of aircraft fatigue. We do not claim to have the best procedures, and certainly not the most advanced aircraft systems in service. Nevertheless, the combination of our available technology together with a commitment by all those concerned with the structural health of our aircraft, has enabled us to approach aircraft fatigue management with confidence.

Figures:

1. Title.
2. Aircraft Structural Integrity - Objectives.
3. Organization - Infrastructure - Fixed Wing Structural Integrity Working Party.
4. Statement of Operating Intent.
5. Statement of Operating Intent - Flying Profile.
6. Fatigue Meter.
7. Fatigue Formulae.
8. Operational Loads Measurement.
9. The Logistics Establishment.

10. Aircraft Types on the MACD Fatigue System.
11. Tornado IDS - Fatigue Formulae.
12. Tornado IDS - Configuration Cases.
13. Tornado Component Management.
14. MOD Form 725 Flying Log and Fatigue Data Sheet.
15. Tornado Fatigue System - Data Processing.



Aircraft Fatigue Management

In The

Royal Air Force

Squadron Leader M.E.J. Render RAF & Squadron Leader J.E. Stevens RAF

Figure 1

Aircraft Structural Integrity

Objectives

- Flight Safety
- Aircraft Availability
- Cost

Figure 2

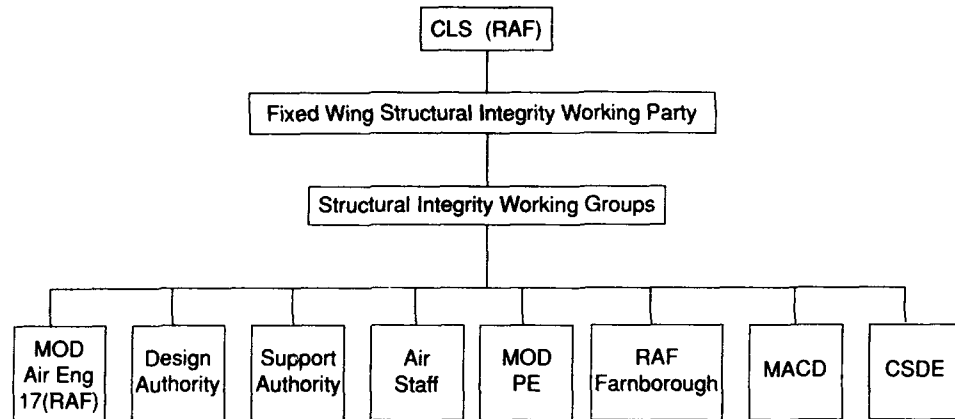


Figure 3



Statement Of Operating Intent

Tornado F3

W. G. J.

May 1989

By Command of the Defence Council

Sponsored for use in the Royal Air Force by D Air Eng 1 (RAF)

MINISTRY OF DEFENCE

Figure 4

STATEMENT OF OPERATING INTENT - FLYING PROFILE

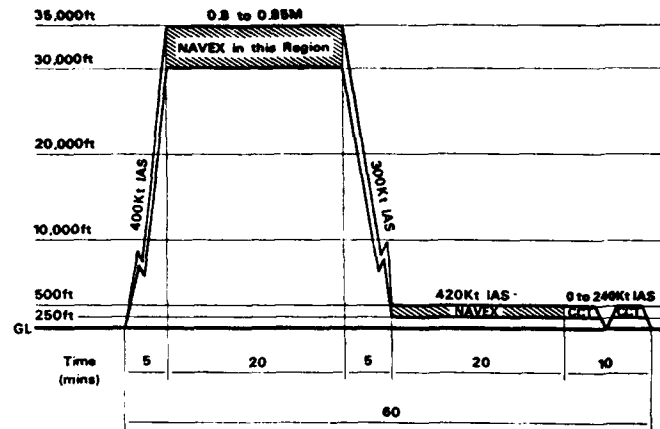
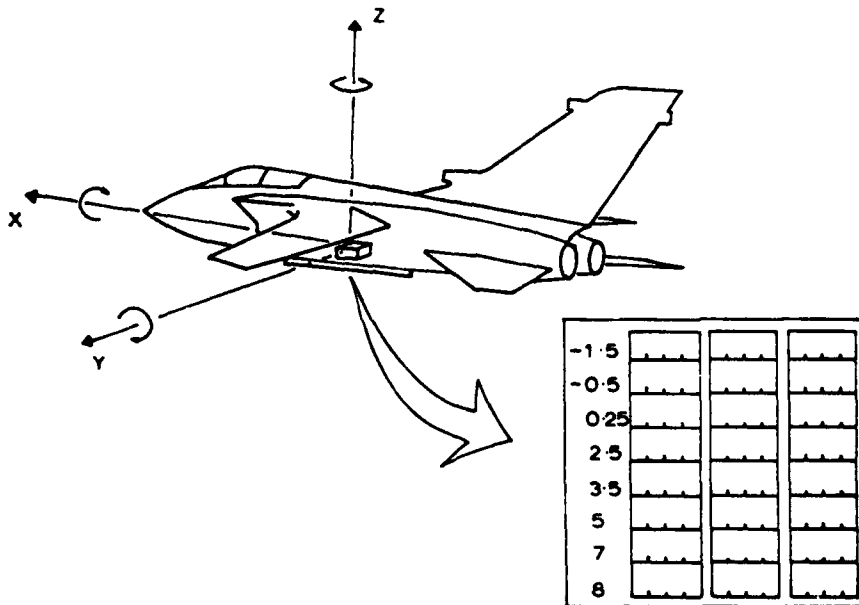
Sortie Code 4. HI-LO NAVEX
Attack and Reconnaissance

Figure 5



TORNADO FATIGUE METER

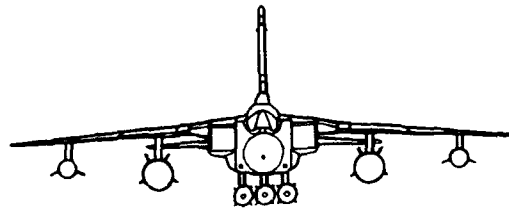
Figure 6

Fatigue Formulae

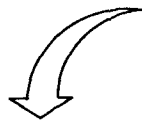
Fatigue Index (FI) =

f [g' Levels, Mass, Configuration, Landings]

Figure 7



Fatigue Data



Operational Loads Measurement

Figure 8



Royal Air Force
The Logistics Establishment



Maintenance
Analysis &
Computing
Division

Figure 9

AIRCRAFT TYPES ON THE
MACD FATIGUE SYSTEM

BULLDOG	JETSTREAM
CHIPMUNK	PHANTOM
HARRIER GR3	SEA HARRIER
HAWK	JAGUAR
TORNADO GR1/1A	TORNADO F2/3
TUCANO	HARRIER GR5

NIMROD, SENTRY AEW, HARRIER GR7

Figure 10

TORNADO IDS - FATIGUE FORMULAE

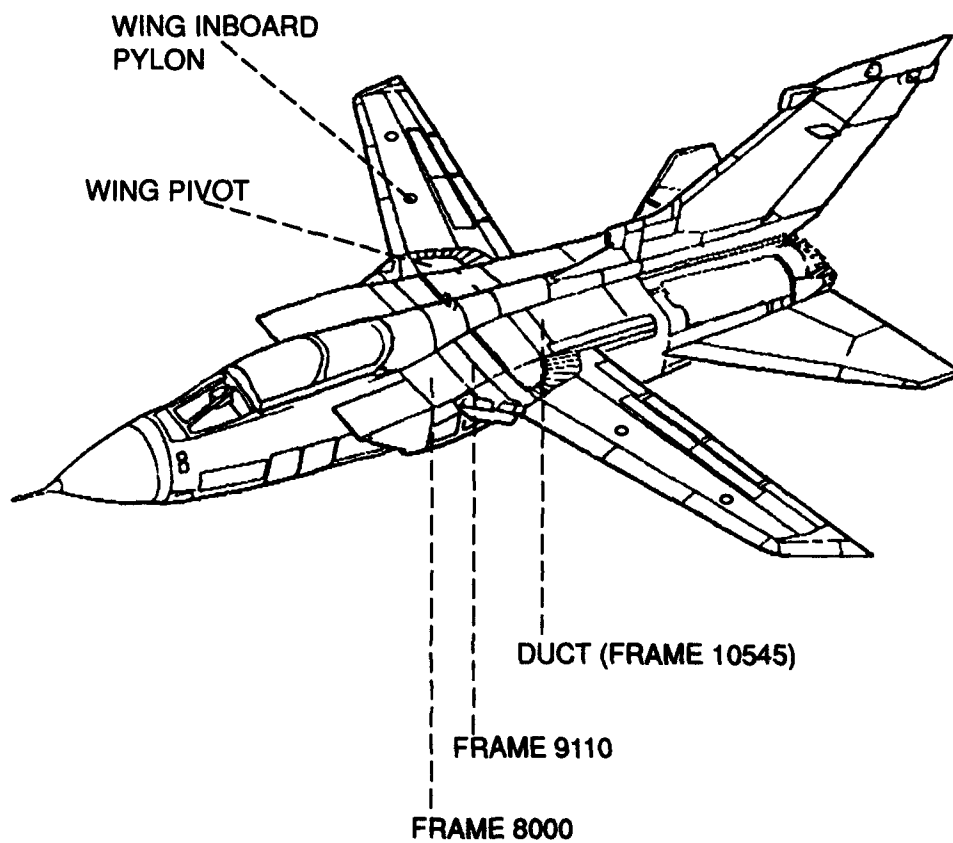
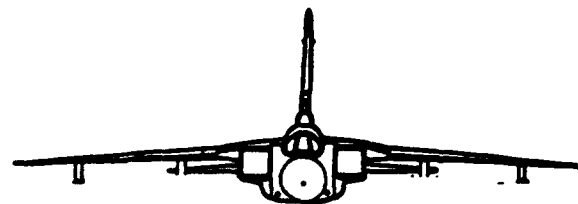
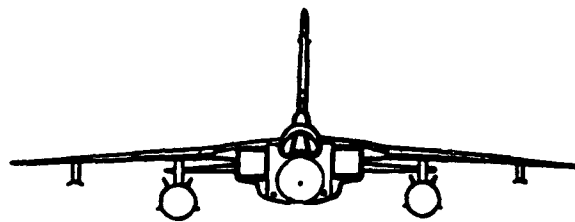


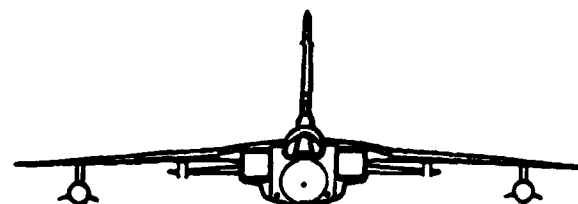
Figure 11

TORNADO CONFIGURATION CASES

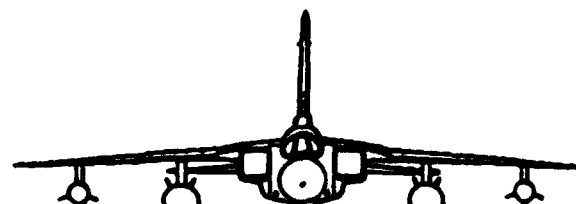
CLEAN WING



FUEL TANKS

CHAFF/FLARE
DISPENSER

ECM POD

CHAFF/FLARE
DISPENSER

FUEL TANKS

ECM POD

Figure 12

TORNADO COMPONENT MANAGEMENT

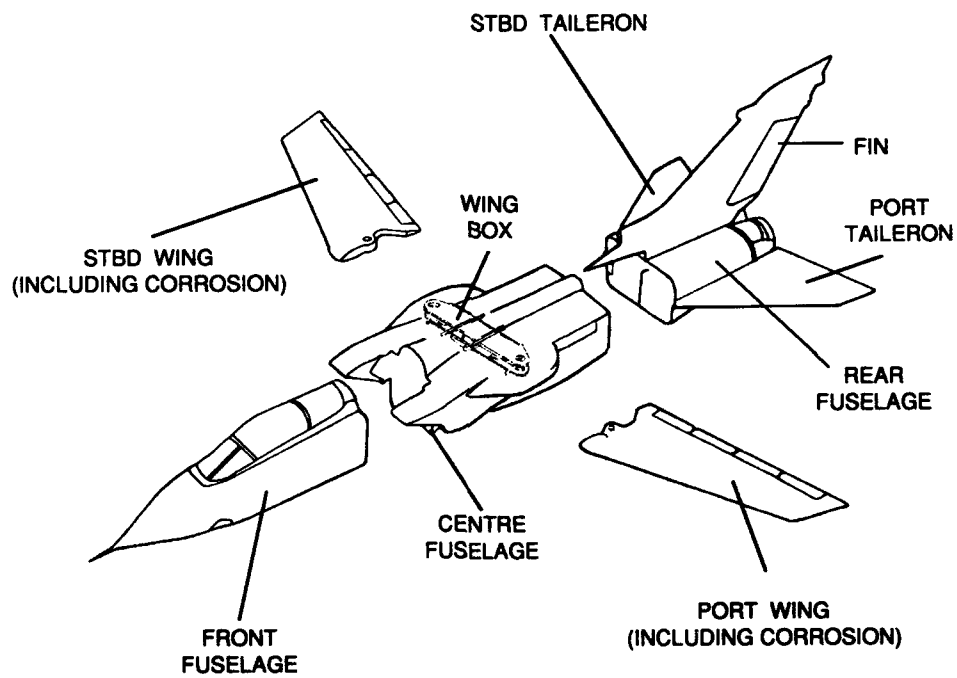


Figure 13

Flying Log and Fatigue Data Sheet

MOD Form 725 (Tornado GR1) (Revised May 88)

MACE Pro-Input[illegible]

The diagram shows a 16-bit bus system with two processors, P and S, and four memory modules. Processor P is on the left, S on the right. Each has a 16-bit data bus. Between them are four memory modules, each with a 16-bit data bus. The bus lines are labeled with 'Start-up' and 'Shut-down' signals.

External Stores Codes		AI Start - up	No Pyrotech		No Stores Carried	
C	Clean (No Store, No Pyrotech)		B	TSC (No Stores Carried)		
P	Pyrotech/Adaptor Only		G	BL755		
F	2350 L Tank		H	2 x BL755		
T	1500 L Tank		R	Mk 13/15 Bomb		
E	ECM		K	2 x Mk 13/15 Bomb		
M	ALARM		Y	JP 233		
L	Sidewinder (SW)		D	Laser Designator Pod		
S	Launcher (SW)		U	600lb Special Weapon		
L	Launcher (SW)		V	950lb Special Weapon		
A	CBL5 200		N	Chaff/Flare Dispenser		
W	Laser Guided Bomb					
Q	Non Standard Store (See Reverse of Form)					
AI Start - down						
X	Stores Discharged		J	Stores Jeisoned		Z No Change

Component Serial Numbers

Port Wing	Fin	Starb Wing
Port Taileron	Wing Box	Starb Taileron

Captain	(Name)	(Service No.)	
Tradesman (Name)			

[illegible]

Flight Details									
Date (Day/Month/Year)									
Take-off Time									
B/F Aircraft Hours									
Duration									
Total Aircraft Hours									
Presentations									
Basic Aircraft Weight									
MAX Profile Flew Yes / No / N									
Fuel at Start - up									
Fuel Received									
Conducts									
Fuel at Shut - down									
Refuel Flew Yes / No / N									
Acquisition (Round)									
Full Stop									
Arrested									
Landing									

Sortie Profile Codes	
1	Display
2	Air Combat Training:
	Basic Defensive Manoeuvring:
	Tail Chase:
3	Operational Low Level
4	Lo-Lo with Eversion
5	Lo-Lo
6	Hi-Lo with Eversion
7	Hi-Lo
8	General Handling: Check Flights:
	Instruments Readings: Air Test
9	High Level Transit
10	Weapons Range

Note: Further explanation of Sortie Profile Codes can be found in the Tornado GR1 Statement of Operating Intent - Issue 2.

Fatigue Counter									
Range	Brought Forward Readings			Reading This Flight			Difference		
	Bank 1	Bank 2	Bank 3	Bank 1	Bank 2	Bank 3	Bank 1	Bank 2	Bank 3
1 -1.50g	1	2	3	1	2	3	1	2	3
2 -0.90g	2	3	4	2	3	4	2	3	4
3 +0.25g	3	4	5	3	4	5	3	4	5
4 +2.50g	4	5	6	4	5	6	4	5	6
5 +3.50g	5	6	7	5	6	7	5	6	7
6 +8.00g	6	7	8	6	7	8	6	7	8
7 +7.00g	7	8	9	7	8	9	7	8	9
8 +8.00g	8	9	10	8	9	10	8	9	10

Certification by MCO IC Flight Servicing
 I certify that the form is complete, the data is
 accurate and logical and that the appropriate detail
 has been carried forward to the appropriate
 sections. A review has been carried forward to the
 appropriate sections.

Signature

Figure 14

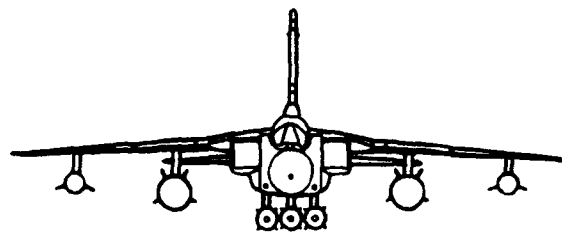
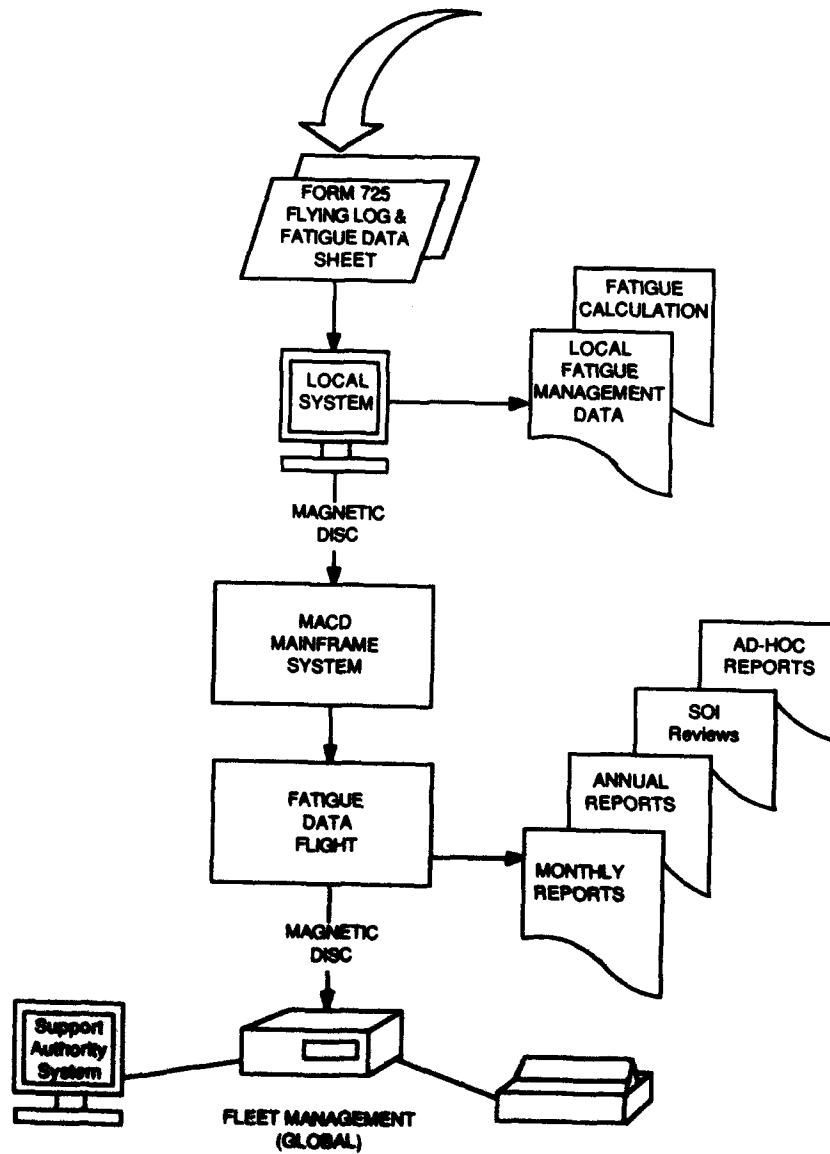
**FATIGUE DATA**

Figure 15

TORNADO STRUCTURAL FATIGUE LIFE ASSESSMENT OF THE GERMAN AIR FORCE

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SUMMARY

The structural development and the fatigue verification of the TORNADO programme has been completed. The majority of the aircraft have been delivered to the German Air Force and some of them have been in full operational use for a period of one decade. This is considered a suitable time to review the approach for ensuring the long-term structural airworthiness, from the users point of view.

1. INTRODUCTION

The TORNADO fleet forms the backbone of the German Air Force air attack capability. Therefore, the long-term structural integrity of this programme is of paramount importance. Under the increasing constraints of available resources, in terms of military budget and manpower, effective fatigue management, which enables the airworthiness to be maintained at the required level, becomes a vital element of the overall maintenance programme.

TORNADO was developed as a tri-national European cooperation programme. For the design purpose, the formulation of common fatigue design requirements and verification principles among the participating Nations was necessary. Having passed the design and structural verification process, the emphasis has changed and is now focusing on the in-service considerations. These tasks have to be established for each Nation according to their own national maintenance procedures.

In Germany, as in other countries, there are several authorities and organizations involved on the subject of military aircraft structures.

The fatigue life assessment for the German TORNADO's described here is presented from the view of BWB-ML, which is the national airworthiness authority responsible for all activities involved to provide

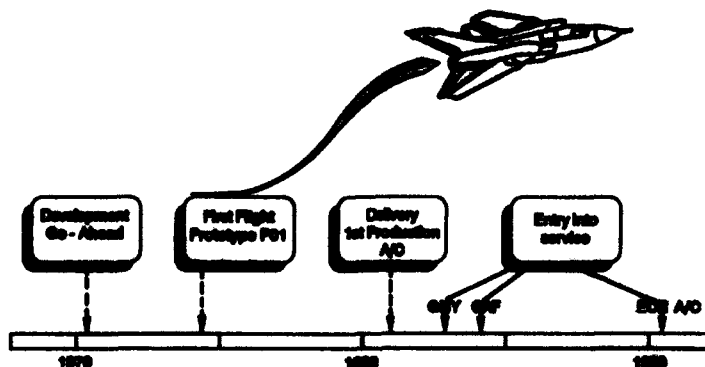


FIG.1 MAJOR MILESTONES TORNADO PROGRAMME

structural integrity of the German military service aircraft and IABG, which conducted the majority of the TORNADO fatigue tests, performing all fatigue life tracking on German service aircraft and monitoring, on behalf of the German Ministry of Defence, the structural matters on TORNADO during development and in-service phase.

2. PRESENT TORNADO PROGRAMME STATUS

The TORNADO aircraft is a two seater, twin engine, variable sweep wing, Mach 2 fighter bomber. Its development started in early 1970. The maiden flight of the first prototype was in 1974. The first production aircraft has been delivered in 1980. Currently more than 900 aircraft have been built. Fig. 1 shows the major milestones history of the TORNADO programme.

Three different variants of the TORNADO exist: IDS (Interdiction/Strike), AD (Air Defence) and ECR (Electronic Combat/Reconnaissance) aircraft. Germany has ordered for its Air Force (GAF) and NAVY (GNY) 322 IDS aircraft and 35 aircraft of the ECR version. Most of them are delivered to the air force and navy squadrons. Since entry into service more than 600,000 flying hours have been accumulated by all TORNADO customers.

TORNADO was jointly developed by the partner countries United Kingdom, Italy and Germany. The national workshares were 42.5:15:42.5 % for the companies BAe (UK), AIT/ALENIA (IT) and MBB (GE).

3. REVIEW OF FATIGUE DESIGN PRINCIPLES APPLIED

The major fatigue design criteria for the development of the TORNADO airframe are summarized in Fig. 2.

This is history now. However, in retrospective, at a time when most of the aircraft have been delivered and are in service now, it is considered useful to review the applied design goals:

Design Spectrum and Service Life
For the design a common spectrum was defined, which covered as an envelope the most stringent requirements of the participating Nations. From the very beginning of the development it was obvious that this severity in the design spectrum gives, at least for the GAF and GNY missions, a considerable margin for a structural life extension beyond the 4000 design service life hours.

Fatigue Safety Factor
To cover scatter in endurance life, a factor of 4 was established. So, for fatigue qualification, a life of

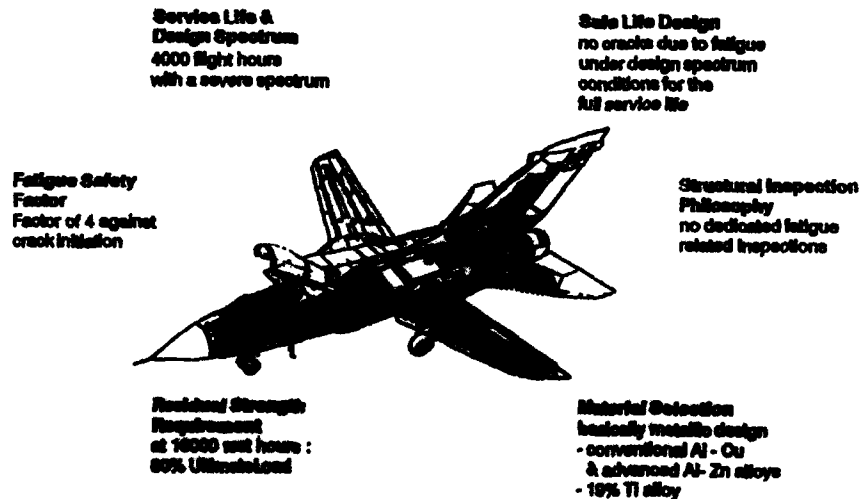


FIG. 2 TORNADO STRUCTURAL FATIGUE DESIGN PRINCIPLES

4000x4 = 16000 test hours had to be demonstrated. However, it was not explicitly specified to which level of strength degradation this factor has to be applied: factor of 4 against what? Generally, there can be expected a wide range between the time of local crack initiation and the point of total collapse of the airframe. Typically for TORNADO fatigue test results, there was often a factor of 2 between the two extremes.

In cases of premature failures a factor of 4 against crack initiation has been used for all those fatigue design improvements which could be implemented during in-line manufacturing. For those aircraft, which require a retrofit modification during in-service, GE applies the rule: The lower value of either a factor of 2 against crack initiation, or a factor of 4 against failure (where the residual strength is expected to fall below 80 % residual strength).

Residual Strength Requirement
The guideline was 80 % Ultimate Design Load at the end of the fatigue life i.e. at 16000 test hours.

However, this goal was not considered as a real design driver and test evidence was shown in a few cases only, so for instance for the outer wing. In general, having successfully reached the 16000 test hours, all the Nations favoured the options either to continue fatigue testing and/or complete the test by a thorough 'tear down' inspection.

Safe Life Philosophy and Structural Inspection Concept

TORNADO is strictly designed to the safe life principles, i.e. a full crack-free life (under the specified design spectrum) has to be assumed and consequently, no dedicated structural inspections due to fatigue would become necessary.

GE has applied this approach as far as possible, because it is believed that this will minimize the structural maintenance effort in service.

Good damage tolerance behaviour was provided in general terms only, as by structural redundancy or for the material selection process. No specific damage tolerance verification tests have been carried out.

Material Break Down

The airframe basically consists of metallic design. As major materials the conventional Al-Cu alloys (2024, 2014) and the advanced Al-Zn alloys (7050, 7475) have been used. Also, a significant portion of 19 % of the structure has been made in titanium alloy Ti-6Al-4V (mainly for the wing carry through box and the wing sweep diffusion area).

Up to now, no service problems are known, which could be related to the material selection.

4. FATIGUE VERIFICATION TEST PROGRAMME

The results of fatigue verification testing provide the basis for any fatigue life assessment in service. A brief summary of the TORNADO-IDS tests is given in Fig. 3. Additional tests conducted for the AD variant are excluded here, because of GE's non-involvement in this task. The ECR aircraft is, with respect to fatigue, identical to IDS aircraft.

The fatigue verification process for TORNADO-IDS is basically completed now. Only the Full Scale Fatigue Test (MAFT), having reached 16000 test hours, will be continued to explore the fatigue strength reserves beyond the design requirements. Generally, this approach was applied on most of the component tests and in many cases there was a considerable margin in service life.

5. SERVICE LIFE CONTROL AND FATIGUE RELATED MAINTENANCE

So far, the TORNADO fatigue life has been considered from the design and verification point of view. Of course, the real in-service fatigue behaviour will differ in a wide range, for a variety of reasons, although the design criteria were tailored to the expected later in-service usage. The major reasons for those deviations result from:

- differences in the actual GAF/GNY usage spectrum, compared with the trinationally agreed design and test spectrum
- influence of environmental and accidental effects, not covered by fatigue testing under laboratory conditions

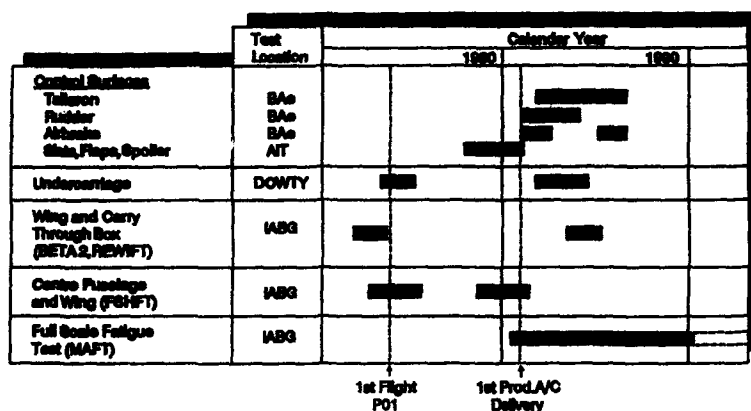


FIG.3 TORNADO (ID6) FATIGUE VERIFICATION TESTS

- simplified test load conditions which do not simulate all actual service loadings, e.g. vibration induced phenomena
- differences within the build standard of the test article and the individual service aircraft.

The presence of these effects cannot be ignored. Only partially are they covered by the fatigue scatter factor, which is applied to the

fatigue test results to derive the allowable service life limits.

Additional measures are required: fatigue load monitoring, structural inspections and, where applicable, fatigue improvement by design modifications or replacement of components after a given time limit.

In the following the methodology of controlling the service fatigue life is described. The principle is shown in Fig. 4: The control tools are

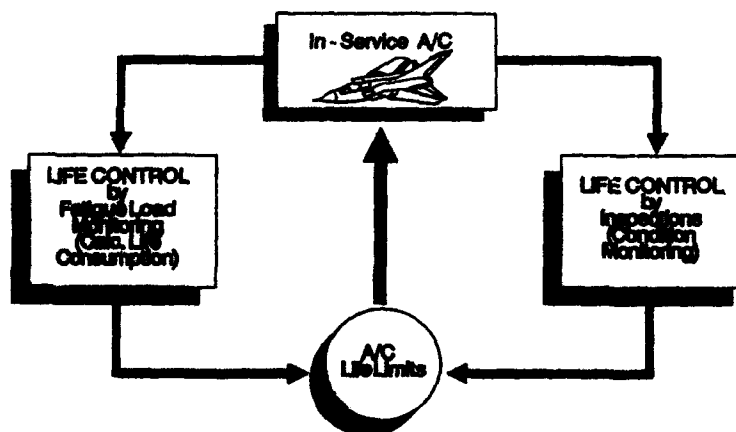


FIG.4 FATIGUE LIFE CONTROL OF IN-SERVICE AIRCRAFT

<i>Device</i>	<i>Function</i>	<i>Purpose</i>
Fatigue-meter (first solution)	accumulative g-counts in 3 wing sweep ranges	for individual A/C tracking of early in-service use
OLMOS (full fleet)	accumulative n * W counts, with other operational parameters as external store config., wing sweeps	for individual A/C tracking - final solution
Maintenance Recorder (selected A/C)	continuous recording of operational flight parameters, strain gauges etc.	for special investigations, as strain gauge measurements, for a limited period

FIG.5 FATIGUE MONITORING SYSTEM TORNADO (GAF)

twofold, by fatigue load monitoring and by inspections. Both methods complement each other: fatigue load monitoring is based on calculated fatigue damage (relative to the fatigue test result) and cannot be made visible or verified - until cracks occur. In contrary, proper fatigue inspections will demonstrate that a certain fatigue limit has not yet reached and therefore, the structure is still safe within a given inspection period (a moderate crack propagation is assumed).

5.1 FATIGUE LOAD MONITORING

Initially, the early GAF TORNADO's delivered were equipped with a so-called 'fatiguemeter' which records the cumulative g's within 3 wing sweep ranges. Soon, service experience showed that this system could be further improved to gain the full benefit of fatigue load monitoring for an effective structural maintenance programme, allowing the fatigue reserves to be exploited by the correction of too conservative assumptions for the calculation of the fatigue consumption rate.

It is typical for TORNADO usage that fatigue loading will vary more than usual for other GAF aircraft, because of large variations of operational parameters as extended flight envelope, external store combinations and flying masses, and

due to several aerodynamic wing configurations.

For these reasons, for the GAF, the fatiguemeter has been replaced by the OLMOS (On-Board Life and Event Monitoring System) device. This system allows, beside the cumulative g times flying mass recording, the registration of additional fatigue relevant flight parameters as external store configuration, wing sweep angle and flap and slat position. Furthermore, OLMOS is a multipurpose device, which also will be used for engine health monitoring and structural limit exceedance monitoring.

In addition to OLMOS, selected GAF aircraft are equipped with a Maintenance Recorder System, which enables the registration of further parameters and strain gauge measurements.

A summary of the GAF fatigue monitoring system for TORNADO is given in Fig. 5. More details are described in Ref. 1.

The results of the fatigue load monitoring will be used for:

- control of fatigue life of the individual aircraft components: fatigue damage consumed and extrapolation to the expected remaining life in flights hours and calendar time

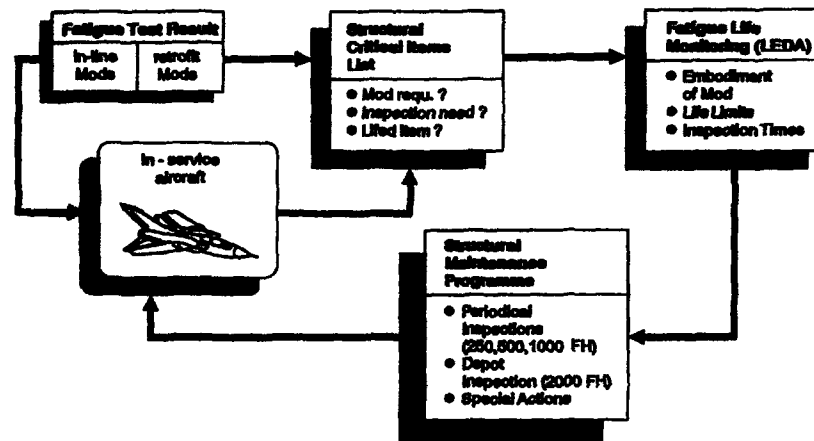


FIG. 5 FATIGUE RELATED STRUCTURAL MAINTENANCE PROGRAMME TORNADO (GAF)

- derivation of the need and time schedule for the implementation of retrofit modifications
- adjustment of the fatigue related part of the structural inspection programme, depending on the actual usage

The principle of fatigue life control and the interactions with fatigue test results and fatigue related maintenance, as applied for the GAF TORNADO, is described in Fig. 6.

5.2 EMBODIMENT OF RETROFIT MODIFICATIONS

As explained already, design improvements have been derived from the fatigue test results and, as far as feasible, implemented during manufacturing of production aircraft. However, due to the overlap of fatigue testing and aircraft manufacturing, this has not always been possible. Also, it has to be realized that not all fatigue problem areas for service aircraft can be identified by ground testing - a matter which probably will be experienced for any aircraft programme.

Consequently, some retrofit modifications have to be embodied

during in-service for the rectification of the fatigue life. The time of incorporation will be, for the German TORNADO's, controlled by the LEDA fatigue management programme, individual for each aircraft and for each location. It is GAF policy to embody, as far as possible, retrofit modifications depending on the flight hour related maintenance programmes, i.e. at the 'major' depot inspection or the 'minor' periodical inspection. A preference to the depot inspection for implementation time will be given, because of manpower, equipment and test facilities available, and the higher degree of accessibility involved.

The fatigue consumption depends on the individual usage spectrum, which in general does not follow a simple correlation with flight hours. Therefore it is necessary to convert the fatigue damage in individual flight hours for the derivation of the point of embodiment for the retrofit modifications. Also, from the economic point of view, it has been shown useful to group certain modification measures, within a given time range, together for simultaneous implementation. Fig. 7 shows for the major structural modification packages applicable for the GAF TORNADO fleet

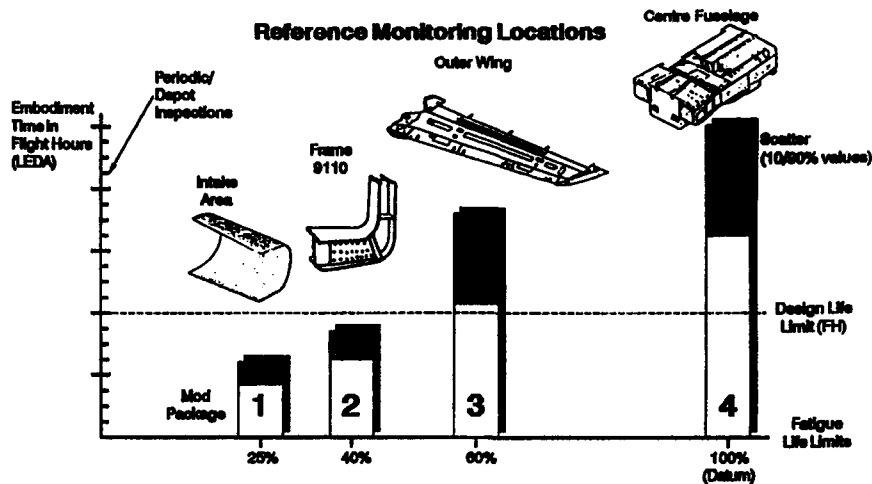


FIG.7 STRUCTURAL RETROFIT MODIFICATION PROGRAMME AND LIFE LIMITS (GAF)

the corresponding life limits, referenced to the design spectrum.

The resulting range in flight hours, expressed by the 10 to 90 % values of the fleet, is also given in this figure. For the purpose of fatigue load monitoring a 'lead' modification per modification package has been chosen, which dictates the latest possible point of embodiment. Also, the differences in build standard within the fleet have to be taken into account for that task.

Fig. 7 also indicates the fatigue life potential for a service life extension exercise (see 6.).

5.3 STRUCTURAL INSPECTION PROGRAMME

Neither fatigue testing nor fatigue load monitoring can fully provide the required assurance for the structural integrity of the service aircraft: not every fatigue prone area can be revealed by fatigue tests, not for every aircraft location the fatigue consumption can be adequately monitored, environment effects could interact the fatigue life, etc. These gaps have to be filled by the application of a structural inspection programme.

According to the safe life principle adopted for the derivation of the GAF inspection needs for TORNADO,

this inspection programme consists basically of visual inspections, looking for environmental and accidental damages only, rather than a fatigue dominating inspection schedule. With respect to fatigue, these inspections are aimed at controlling possible detrimental effects, which could negatively influence the fatigue life. The fatigue life itself plays a secondary role only: to define the criticality of the individual locations and the acceptance thresholds for damage limits.

In addition, a few dedicated 'fatigue control inspections' will be applied. The objective is to provide further confidence, that the fatigue limits have not been reached, yet: fatigue control by inspections, where applicable.

The amount of these inspections is limited to a few 'pilot areas' and involves thorough NDT inspections (by eddy current or ultrasonic aids). The philosophy behind these 'spot check' inspections is the assumption, that a local reference area can be used as an indicator for the actual fatigue life consumed on surrounding large areas as well. If there are no signs of crack at these 'pilot areas', no cracks are to be expected in other regions, too.

However, the adoptability of this principle is restricted by:

- good damage tolerance characteristic at that location is required
- test evidence has to be provided, that cracks always emanating from the same location, far before other areas are affected
- a thorough NDT inspection should be easily feasible at the pilot area

As an example, the applicability of this concept for the outer wing is demonstrated in Fig. 8: Early signs of crack initiation (crack length below 0.5 mm) occurred at 30 % of the overall wing life, at 56 % the lower flange of the front spar failed locally, without a reduction in strength below 150 % limit load, and finally at 100 % of the test life numerous fatigue cracks developed at the lower skin, which terminated the life of the wing box structure. This sequence in fatigue crack occurrences has been confirmed on several test articles.

For the derivation of the fatigue control inspection interval, a safety factor of 3 was applied against crack initiation time. Because the inspection will be performed during regular flight hour

related servicing, the actually applied factor scatters something for the individual aircraft.

5.4 SERVICE EXPERIENCE SO FAR

The GAF fleet leaders, in terms of flight hours, have exceeded 2000 flight hours, whilst the latest aircraft have just been delivered to the squadrons.

Certainly, it is too early to derive definitive conclusions on the effectivity of the fatigue control programme applied for the GAF TORNADO's. However, at least some indications can be given on the present experience available:

- (a) Up to now, no serious fatigue damages or other structural events have occurred in service use, which would have been directly affected the structural airworthiness.
- (b) For all fatigue critical locations, identified by Full Scale Fatigue Tests, after embodiment of design improvements, the service aircraft do not show any signs of fatigue deterioration, indicating that the fatigue sensitive areas are effectively eliminated.

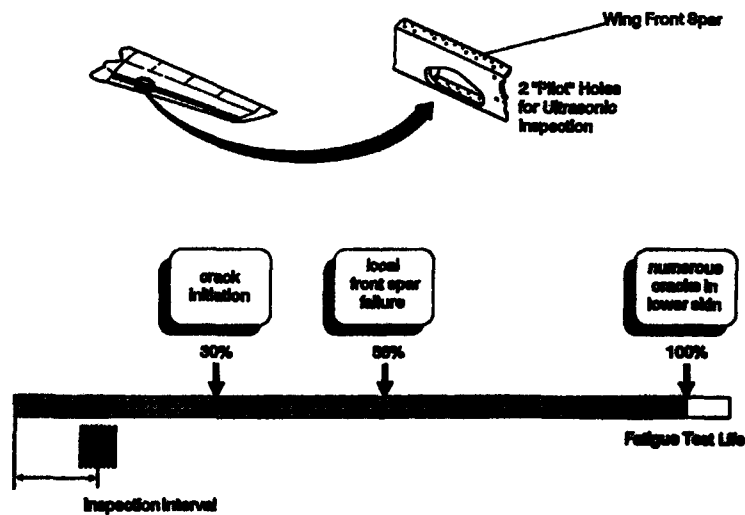


FIG. 8 "PILOT AREA" INSPECTION CONCEPT (GAF)

- (c) The general in-service behaviour of the TORNADO structure appears to meet the expectations. So, it was possible for the GAF to extend, after some updates, the initial inspection intervals, proposed by the companies, from 300 to 500 flight hours for the periodical inspection and from 900 to 2000 flight hours for the depot inspection. The potential for a further reduction of the structural inspection effort is currently investigated. However, the amount of fatigue related maintenance is only a minor contribution to the overall programme.
- (d) The severe fatigue design spectrum applied for the design and verification process of the TORNADO airframe offers a significant potential for stretching the maximum service life. Presently, the GAF is flying, as an average figure, only 25 to 40 % in severity of the design spectrum.
- (e) A somewhat disappointing feature is the fact that service use demonstrates, once more, that fatigue testing and fatigue monitoring cannot fully prevent all fatigue and wear related problems. So, some minor damages on secondary structures

occurred, caused by aerodynamically induced vibrations on thin skin panels. Solutions in form of design modifications have been prepared to overcome these problems.

However, this situation is not considered as peculiar to TORNADO structure.

6. SERVICE LIFE EXTENSION

The majority of the GAF TORNADO fleet is in the early phase of service use. However, it is becoming increasingly apparent that the long-term strategy shows there will be a probable need to extend the service life of TORNADO beyond the original design goals, both in terms of flight hours and in calendar time. The amount of life extension and the way ahead is presently in an early stage of consideration. The structural disciplines involved in this task are briefly described (Fig. 9):

Fatigue Test Continuation

The Major Fatigue Test (MAFT) will be extended beyond the required design life to show up the actual reserves in fatigue life. For some other component tests these reserves are known already, which can be made available for life extension.

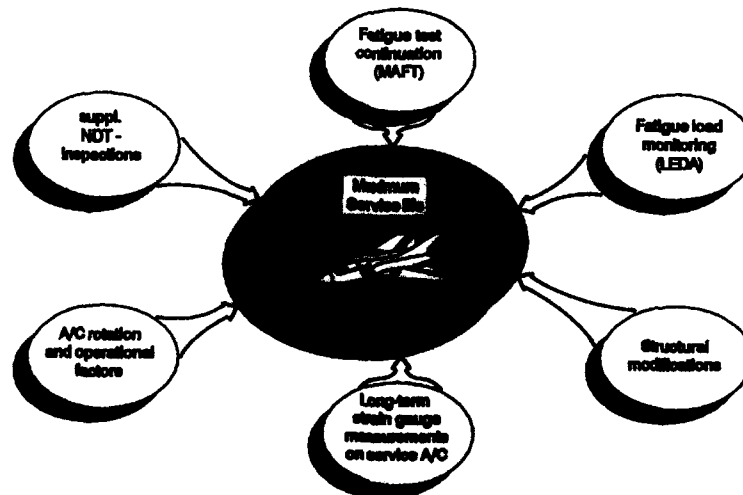


FIG.9 STRUCTURAL LIFE EXTENSION PROGRAMME TORNADO (RAF)

Fatigue Load Monitoring and Structural Modifications

The amount of life extension potential for the individual components of each aircraft and for the whole fleet will be controlled by the LEDA monitoring system. Also the need for structure modifications, depending on the fatigue life consumed, will be determined. Fig. 7 gives some indications on this subject (for instance, modification package No 3 includes extensive cold working of wing fastener holes).

Strain Gauge Measurements on Service Aircraft

The fatigue load monitoring controls basically the fatigue life limits of the g-related components as wing, wing carry through box and centre fuselage, which are regarded as the most fatigue sensitive areas. However, for the life extension the overall airframe has to be considered: How much fatigue is consumed of the remaining components, where for the fatigue life other parameters than g's are dominating? Certainly, it cannot be assumed that for fin, taileron, undercarriage and large areas of front and rear fuselage the fatigue rates will follow the same rules. In order to gain a more detailed picture of the overall fatigue situation of the aircraft, long-term strain gauge measurement for these non-g-related components are intended. Presently these strain gauges will be installed on several GAF service aircraft.

Aircraft Rotation and Operational Factors

It is obvious, that some aircraft or some squadrons will fly more severely than others. Rotation of aircraft, i.e. mixing high fatigue missions with low stress missions will enable to delay early retirement of the TORNADO fleet.

Life Extension by Inspection

As a measure for expanding the allowable service life, the application of a supplementary NDT inspection programme for the late in-service phase is envisaged. This would allow, in combination with other measures, to extend the service life for a limited period. The inspection need would be derived from a damage tolerance approach, i.e. considering the remaining life during the crack propagation phase.

CONCLUSION

It has been attempted to provide a brief summary of the overall fatigue control programme for the TORNADO aircraft of the German Air Force. The approach described is based on the interactions of the key elements fatigue test results evaluation and fatigue consumption control on service aircraft by fatigue load monitoring and structural inspections.

The principles applied probably do not significantly differ from those for other aircraft and other western Air Forces practices. However, the emphasis on the detailed features involved might vary, depending from the particular individual engineering, operational and budgetary situation, and the maintenance resources available.

REFERENCE:

1. Neunaber, R.: "Aircraft Tracking for Structural Fatigue" presented at the 72nd AGARD Structures and Materials Panel, Specialists' Meeting on Fatigue Management, 29 April - 1 May 1991 in Bath, United Kingdom

FATIGUE MANAGEMENT FOR THE A-7P

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SUMMARY

One objective of an Aircraft Structural Integrity Program (ASIP) is to ensure that all primary structure is both durable and damage tolerant; that is, it is able to resist both cracking and failure due to cracking. The heart of any durability and damage tolerance assessment is crack growth predictions. This requires a crack growth program, accurate stress intensity and load interaction models, and reliable material properties. With these tools both durability analysis and damage tolerance analysis can be performed. All potentially critical locations were assumed to exhibit slow crack growth. No fail-safe concepts were used, although many locations have alternate load paths that carry limit load.

LIST OF SYMBOLS

ΔK	Stress Intensity Range
Δn_z	Aircraft Incremental Normal Load Factor
$\Delta \sigma$	Stress Range
σ	Peak Stress
ASIP	Aircraft Structural Integrity Program
BL	Butt Line
BTAB	Computer Program for Stress Intensity Factors
C	Forman Equation Material Constant
c	Crack Length
CA	Counting Accelerometer
c_{cr}	Critical Crack Length
c_{fl}	Functional Impairment Flaw Size
D & DT	Durability and Damage Tolerance
da/dN	Crack Growth Rate
D.I.	Damage Index
e	Base of Natural System of Logarithms
EBH	Equivalent Baseline Hours
ETPCRO	LTV Crack Growth Computer Program
E-MODEL	FLISPEC Multivariable Load-Environment Model
FH	Flight Hours
FLISPEC	LTV's Flight Spectra Development methodology
g	Acceleration Due to Gravity
H	Altitude

ΔK	Threshold Stress Intensity Range below which no crack will grow
K_C	Critical Stress Intensity for Plane Stress Failures
K_{IC}	Fracture Toughness for Plane Strain Failures
K_f	Forman Equation Material Constant
KSI	Thousands of Pounds/in ²
Ln	Natural Logarithm
LWS	Lower Wing Skin
m	Wheeler Retardation Exponent
M	Mach Number
n	Forman Equation Material Constant
Nkg	N_z exceedances normalized to 1000 FH at $k = 5, 6, 7, 8$ g's
n_z	Aircraft Normal Load Factor
$\bar{n}_z$	Peak n_z
OWP	Outer Wing Panel
POAF	Portuguese Air Force
R	Stress Ratio
RAM	Reliability and Maintainability
SEAFAN	Sequence Accountable Fatigue Analysis Routine
SLA	Structural Life Assurance
SOR	FLISPEC Spectra Ordering Program
t_a	Time for an 0.05 in. flaw to grow to a critical crack size of 1.21 in. at WS 32.2 (Loc. AD)
USN	United States Navy
USAF	United States Air Force
W	Weight
WCS	Wing Centre Section
WS	Wing Station
FS	Fuselage Station
FEM	Finite Element Model
LWS	Lower Wing Station
NDI	Nondestructive Inspection

1. INTRODUCTION

Since its design and development in 1964, the A-7 has consistently demonstrated superior maintainability and reliability. The A-7 airframe has been especially noted for its durability, with many aircraft reaching and exceeding their guaranteed service lives.

Although the durability of the A-7 structure was well established, high-strength metal fatigue failure in other aircraft had become widespread.

In response to this problem, the United States Air Force initiated an Aircraft Structural Integrity Program (ASIP), which required aircraft to be designed to operate under projected service loads, and analyzed and tested to demonstrate a safe fatigue life. In 1972, later revised in 1975, these ASIP requirements were documented in MIL-STD-1530A and the concepts of durability, damage tolerance control, and fleet management were introduced.

In September 1974, Vought Aero Products Division, under the authority of the Air Force, began a Damage Tolerance and Fatigue Assessment Program to qualify the A-7D for MIL-STD-1530 ASIP requirements. This Program effort concluded in January 1977, with the qualification of the A-7D. A result of this program was the establishment of inspection and aircraft modification requirements designed to extend the life of the aircraft to 8000 flight hours and beyond, when proper inspection and maintenance procedures are observed. Subsequently, several programs were conducted to enhance and update the basic effort.

The extensive A-7 ASIP data base provided a firm and very cost effective foundation for development of a Portuguese A-7P aircraft inspection and maintenance program. While the A-7P will comfortably attain its guaranteed service life, the structural integrity potential of the airframe will be greatly enhanced if properly maintained and monitored.

The current usage of the A-7P aircraft indicate an average service life of 16,000 hours. The A-7P have now an average flight hour of 4500 (MIN-2950 FH, MAX-5600 FH).

2. A-7P ASIP

The program consisted of two phases spanning twenty-seven months. Phase I dealt with POAF A-7P CA installation and a POAF CA flight survey. Phase II provided for those tasks consistent with MIL-STD-1530A to perform durability analysis, damage tolerance control and fleet management. Results of this program were included: (1) A-7P airframe maintenance and inspection requirements, and (2) a structural damage monitoring program for POAF to continuously track and update aircraft damage.

Phase I program tasks was:

- A. CA Installation
- B. A-7P CA Flight Survey

Phase II program tasks was:

- A. A-7P Usage Determination - Mission Profiles
- B. Critical Structure Selection
- C. Critical Location Evaluation
- D. Operational Limits and Inspection requirements Definition
- E. Structural Damage Monitoring Program Development
- F. Documentation and Reporting

Phase I

Phase I of A-7P ASIP consisted of Vought's delivery of a data package to POAF to define CA Installation of CA's. Also included under Phase I is a six month CA flight survey to establish POAF usage.

Phase II provided the POAF with a Damage Tracking Computer Program and information necessary to establish the maintenance and inspection requirements that will improve the service life and airframe reliability of the A-7P for Portuguese usage.

Current damage tracking of A-7P aircraft require monthly inputs of individual aircraft usage data in the form of counting accelerometer n_2 counts and logged flight hours.

The current method of tracking fatigue damage (crack initiation) in individual A-7P aircraft is based on early fatigue analysis techniques coupled with the results of a full-scale wing fatigue loading consisted of positive manoeuvre loads that were related to one severe (limit load) flight condition. Loads associated with landings and negative flight manoeuvres and the effects of chemical environment and flight-by-flight sequence effects were not included in the test. The damage tracking method is insensitive to deviations in n_2 usage spectrum shape, the chemical environment and the landing usage spectrum.

The current method of tracking damage on individual A-7P aircraft involves a fracture mechanics approach. That is, the damage index (D.I.) is defined in terms of the number of flight hours it takes for an 0.005 inch assumed initial crack at LMS station 32.2 to grow to a critical size. This crack growth time is a function of accrued CA- n_2 counts and logged flight hours.

2.1 Initial Quality Assessment

During the A-7 ASIP the initial manufacturing quality assessment of the A-7 structure was conducted to determine the maximum size of initial flaws that could be expected in the fleet. Eight coupons containing a total of 44 holes were cut from the lower wing skin of a low-time (690 flight hours) A-7. Each coupon was fatigue tested using a modified constant amplitude spectrum until flaws developed and propagated to failure. Each hole was broken open after failure to reveal subcritical cracks. Fractographic analysis traced the growth of each crack backwards in time to obtain its initial size. A total of 85 initial flaws were identified and fractographically measured. They resulted from both mechanical and chemical processing. After initial flaw sizes were determined, the data were statistically evaluated. A log-normal probability distribution was estimated from the initial flaw size data. Seven hundred holes per aircraft were located in potentially critical areas, with approximately 500 A-7's in the fleet. Therefore, the maximum flaw in any one hole in this fleet was assumed to be that which occurred less than once in the 350,000 holes. Based on the statistical distribution and a 95% confidence bound, the flaw size was estimated to be 0.0015 inch. This statistically derived flaw size was significantly less than the Air Force 0.0050 inch initial flaw size used for crack growth evaluation and provided the confidence that a conservative flaw size criterion had been selected. The choice of the 0.0050 inch flaw size criterion during this program will maintain confidence that A-7P safety of flight will be assured by use of a sufficiently conservative criterion.

2.2 Critical Structure Selection

2.2.1 Critical Location Selection

Of primary importance in the durability and damage tolerance control of structure in any vehicle, whether it is newly designed or updated as in the case of the A-7, is the identification of potentially critical locations. The selection process for this update is a combination of choosing those previously picked during the original ASIP and choosing additional locations based on current teardown results, Air Force and Navy field inspection results, FEM, and A-7 service history.

2.2.2 Original Locations

The initial selection process began with the original A-7D ASIP when over 200 potentially

critical areas were identified by reviewing: service history, material type and thickness, stress analysis margins, fatigue analysis margins, static test results, fatigue test results, stress concentration factors, test failures, critical load paths, critical crack lengths, fastener types, lugs, stress corrosion and non destructive inspection program results.

These potentially critical areas for the A-7 were systematically reduced by retaining only primary structure, structure that was uneconomical to repair or replace, and structure that was deemed safety of flight, and eliminating areas that did not lead to cracking. These locations are defined in table 1 and are pictorially located in figures 1.1 through 1.3. (Fig. 1.2 - appendix A1)

IDENTIFICATION OF CRITICAL LOCATIONS

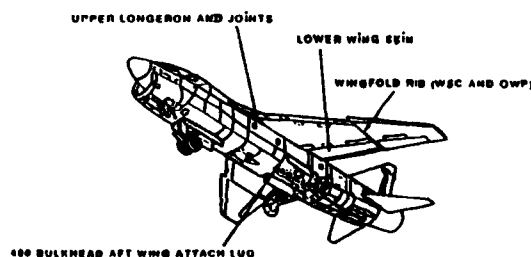


Figure 1.1

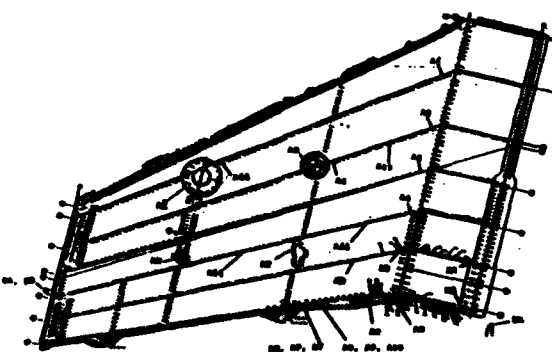


Figure 1.3

TABLE 1 A-7 POTENTIAL CRITICAL LOCATIONS

a) Original Locations

Loc.	Description
A1	Wing Root
A2	Wing Root
A3	Wing Root
A4	Wing Root
A5	Wing Root
A6	Wing Root
A7	Wing Root
A8	Wing Root
A9	Wing Root
A10	Wing Root
A11	Wing Root
A12	Wing Root
A13	Wing Root
A14	Wing Root
A15	Wing Root
A16	Wing Root
A17	Wing Root
A18	Wing Root
A19	Wing Root
A20	Wing Root
A21	Wing Root
A22	Wing Root
A23	Wing Root
A24	Wing Root
A25	Wing Root
A26	Wing Root
A27	Wing Root
A28	Wing Root
A29	Wing Root
A30	Wing Root
A31	Wing Root
A32	Wing Root
A33	Wing Root
A34	Wing Root
A35	Wing Root
A36	Wing Root
A37	Wing Root
A38	Wing Root
A39	Wing Root
A40	Wing Root
A41	Wing Root
A42	Wing Root
A43	Wing Root
A44	Wing Root
A45	Wing Root
A46	Wing Root
A47	Wing Root
A48	Wing Root
A49	Wing Root
A50	Wing Root
A51	Wing Root
A52	Wing Root
A53	Wing Root
A54	Wing Root
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A72	Wing Root
A73	Wing Root
A74	Wing Root
A75	Wing Root
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A78	Wing Root
A79	Wing Root
A80	Wing Root
A81	Wing Root
A82	Wing Root
A83	Wing Root
A84	Wing Root
A85	Wing Root
A86	Wing Root
A87	Wing Root
A88	Wing Root
A89	Wing Root
A90	Wing Root
A91	Wing Root
A92	Wing Root
A93	Wing Root
A94	Wing Root
A95	Wing Root
A96	Wing Root
A97	Wing Root
A98	Wing Root
A99	Wing Root
A100	Wing Root

b) Additional Locations

Loc.	Description
A1	Wing Root
A2	Wing Root
A3	Wing Root
A4	Wing Root
A5	Wing Root
A6	Wing Root
A7	Wing Root
A8	Wing Root
A9	Wing Root
A10	Wing Root
A11	Wing Root
A12	Wing Root
A13	Wing Root
A14	Wing Root
A15	Wing Root
A16	Wing Root
A17	Wing Root
A18	Wing Root
A19	Wing Root
A20	Wing Root
A21	Wing Root
A22	Wing Root
A23	Wing Root
A24	Wing Root
A25	Wing Root
A26	Wing Root
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A39	Wing Root
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A43	Wing Root
A44	Wing Root
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A66	Wing Root
A67	Wing Root
A68	Wing Root
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A73	Wing Root
A74	Wing Root
A75	Wing Root
A76	Wing Root
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A79	Wing Root
A80	Wing Root
A81	Wing Root
A82	Wing Root
A83	Wing Root
A84	Wing Root
A85	Wing Root
A86	Wing Root
A87	Wing Root
A88	Wing Root
A89	Wing Root
A90	Wing Root
A91	Wing Root
A92	Wing Root
A93	Wing Root
A94	Wing Root
A95	Wing Root
A96	Wing Root
A97	Wing Root
A98	Wing Root
A99	Wing Root
A100	Wing Root

2.2.3 Additional Locations

Based primarily on the teardown results and field inspections, additional potentially critical locations have been selected. Figure 2 (appendix A1) summarizes the teardown results. Cracking locations fell in three categories.

The first category are locations where high bolt load transfer occurs. As spar caps end, the cap axial load must be transferred into the skin and a fastener load peaking effect occurs at the end fasteners. Each of these locations were added to the potential critical location list.

The second category are locations of high stress. These hot spots were also identified by FEM and lie primarily in the rear spar near wing station 50.0 through 53.7 and at fastener holes in the third through the fifth intermediate spars near wing station 32.2. Locations in these areas were already selected in the original ASIP, however, additional locations in these zones were added to the list for a more complete coverage.

The last category of cracking identified by the teardown are areas that bound the high stress areas and are entitled secondary critical locations. Only tiny flaws were detected in these areas (on the order of 0.001 through 0.005 inches, much smaller than can be detected by NDI). Additional potentially critical locations were identified in these zones so that the extent of cracking could be bounded analytically.

3 GUIDE-LINES FOR THE DURABILITY AND DAMAGE TOLERANCE ASSESSMENT OF THE A-7

The effectiveness of any military force depends in part on the operational readiness of the weapon systems. The military airplane is a weapon system, and one of the major items in that system is the structure. The structure of the A-7 airplane has to be evaluated. Two tools the engineer has available to help perform this evaluation are Durability and Damage Tolerance (D & DT) Assessments. Using tailored criteria based on A-7 characteristics, one can be assured that the airplane will be both economical and safe to fly.

Durability addresses economics. As it considers average usage, average environment and average types of crack growth, it predicts the economic life of an airplane or fleet of airplanes. Damage tolerance addresses safety. As it considers severe environment and rogue types of cracking, it predicts inspection intervals with safety factors included.

3.1 A-7D Crack Types

During the teardown inspection three different types of cracking were observed. Figure 3 depicts these (appendix A1). The average type of cracking was the single point initiation, followed by slow crack growth to failure. Occasionally there were multi-point initiation of a few points that eventually coalesced into a thru flaw and grew to failure somewhat earlier. On a few holes there were multi-point initiation of many points that coalesced very early into a thru flaw and exhibited rapid growth and early failure. In addition, from the A-7 teardown inspections, there were twice as many holes with two cracks emanating from them than with only one crack.

3.2 Crack Growth Methodology

To assess the Durability and Damage Tolerance of a structure, crack growth methodology must address both extremes of the crack growth types. Figure 4 illustrates this (appendix A2). A stress spectrum which includes the effects of buffet must be generated for each potentially critical location. Stress intensity models for both thru and part thru models must be generated which include the effects of bolt load. Retardation parameters which include the effect of buffet must be obtained by test. These, together with a crack growth program, should accurately predict crack growth from

initial quality to failure for both multi-initiation thru flaws and single point initiation part thru flaws. Only then, with defined criteria, can economic lives and safety limits be established.

3.3 Durability Criteria

The following criteria has been established for the durability assessment for the A-7 ASIP update. For each potentially critical location, calculate the time it takes for two part thru flaws to grow from an initial quality size to a functional impairment size under baseline usage with buffet in a lab air environment. This will establish the economic life of that location. For a typical hole in the lower wing skin, the functional impairment flaw size is 0.06 inches, the theoretical size fuel leakage. This size is also economical to repair.

3.4 Damage Tolerance Criteria

The following criteria has been established for the damage tolerance assessment for the A-7 ASIP update. For each potentially critical location, calculate the time it takes for two part thru rogue flaws to grow from a detectable size of 0.05 inches to failure under baseline usage with buffet in a severe environment. Also calculate the time it takes for two thru rogue flaws to grow from a detectable size of 0.02 inches (should have high confidence of detectability) to failure under baseline usage with buffet in a severe environment. The calculation that gives the shortest life will establish the safety limit of that location. Dividing this by a safety factor of two will provide an inspection interval which will allow at least two opportunities to detect the flaw before it becomes critical.

3.5 Crack Growth Program

The computer program "EPPGRO" was used to perform crack growth prediction. It was originally obtained from the Air force in the early seventies and has been enhanced in-house over the years as the state of the art has progressed. This program has been used exclusively on all A-7 programs, the crack growth data base for A-7 parts and components spanning over fifteen years.

EPPGRO is an automated procedure which predicts the propagation of various flaw shapes in structural members under variable amplitude spectral loading environments. The linear elastic fracture mechanics approach calculates the stress intensity for each min-max stress

pair as it passes through the stress sequence, makes adjustments according to the retardation load interaction model if a plastic zone exists from a prior overload, and then determines the incremental crack growth using established growth rate property for the given material. This process is automatically repeated until the critical size is reached.

4 MATERIAL PROPERTIES

Material properties required for a crack growth analysis are growth rates, fracture toughness (or critical stress intensity values if the failure is in plane stress), and retardation parameters (for use in the load interaction model).

4.1 Crack Growth Rates

Crack propagation behaviour for metals is normally defined in curves plotting the growth rate (da/dN) versus the stress intensity range (k). The growth rates used for the A-7 ASIP update are the same as those used during the original A-7 ASIP (Reference 4). Years of testing and correlation have verified these curves. Recent testing during the teardown phase have also verified these curves. The data is displayed in appendix B1 and B2. Where needed, curves are also provided for different values of critical stress intensities (K_{IC}), as this value changes with thickness for the various locations. A curve is also provided for the steel strap at WS24.6 (Location A0). The growth rates used for severe environment were established by defining a curve half way between lab air rates and rates for immersion in water, a precedent set during the original A-7 ASIP.

The crack growth program utilizes the Forman Equation to define the growth rate curves:

$$da/dN = C(\Delta k)^n / [(1-R)k_f - \Delta k]$$

where C , n , and k_f are constants for each curve. "R" is the stress ratio which the Forman equation uses to vary the growth rate for different combinations of min/max stress pairs. For a better fit, EPPGRO requires two sets of the above constants; one for the lower and one for the upper portion of the curve. Also shown are values for k threshold; the stress intensity range below which no crack growth will occur.

4.2 Fracture Toughness

Toughness is a property which is a measure of a material's ability to resist failure. It is a value required in a residual strength analysis for determining the critical, or failing, flaw size. Fracture toughness (K_{IC}), for plane strain failures, and critical stress intensity values (K_c), for plane stress failures, were obtained from the original A-7 ASIP (reference 4). Data is shown in Figure 5. Critical stress intensity values for steel were obtained from the Damage Tolerance Design Handbook (reference 5).

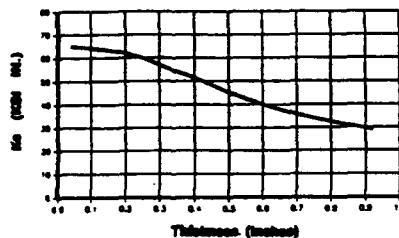


Figure 5

4.3 Retardation

The effects on crack growth under the influence of variable loading have been investigated for some time. It is known that peak overloads will blunt the crack tip and retard crack growth. EPFGRO uses the "Wheeler Retardation" model to mathematically represent this response. In essence the model lowers the growth rates for cycles following an overload until the flaw grows through the zone yielded by the overload. The amount of this reduction is characterized by the Wheeler "n" parameter. This must be determined by test, since it is a function of both spectrum type and material. The retardation curves for 7075-T6 aluminum that were developed during the testing phase of this program are shown in Figure 6 (appendix A2). Data points from both the second intermediate spar tests and rear spar tests are plotted. The curves are well defined and will be used for each location the lower wing skin. As shown, the Wheeler "n" varies with the maximum stress in the spectrum and flaw shape. No retardation was assumed for any components made with steel.

4.4 Stress Intensity Models

Stress intensity models for most potentially critical locations in the A-7 have been developed and verified by coupon and/or component testing during the original A-7D ASIP (reference 4) and during the testing phase of

this program. For those that have not, closed-form solutions for stress intensity factors were obtained using the computer program STAB (reference 6). For the simple cases, these solutions are quite accurate. For more complex cases with load transfer, these solutions are extended by the computer routine using the method of superposition and tend to be conservative since no load is assumed transferred by friction. Significant life increases were demonstrated by clamp-up resulting in load transfer by friction during the 2nd Intermediate Spar Testing. Output was then plotted in the form of unit solutions.

5 DURABILITY ANALYSIS

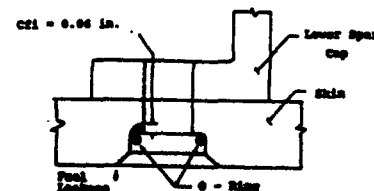
Durability addresses the economics of the structure. It is the ability of an airframe to resist cracking for a specified period of time. This cracking can be brought on by stress corrosion, hydrogen embrittlement, corrosion, wear, etc. for which established manufacturing controls or good design practices are in-place to prevent. Cracking can also occur from small flaws inherent in the material during manufacturing, often called initial quality.

The economic life of a component is the time it takes a flaw to grow from initial material quality size to a size causing functional impairment of that component.

5.1 Initial Quality and Functional Impairment

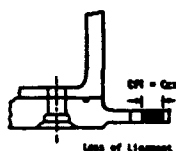
As discussed in the durability criteria in Section 3, the initial quality used for existing structure was 0.005 inches, a value resulting from an evaluation program conducted during the original A-7 ASIP and verified during the testing phase of this program.

Functional impairment refers to a loss of function, such as fuel or pressure leakage, compressive instability, significant load redistribution or in some cases failure. For instance, the functional impairment flaw size for location AD is illustrated in Figure 7



A) LWS, WS-32.2 @ 5th Intermediate Spar
Location AD

Figure 7A



B) Dry (tab) hole at Rear Spar

Location A7

Figure 7B

At this area the wing is wet (forms the lower portion of an integral fuel tank). The hole in the skin is countersunk, and at the bottom of the counterskin is a counterbore which contains an O-ring to prevent the fuel from leaking out of the wing. If a flaw grew from the counterbore-to-shank hole interface, fuel could possibly seep around the O-ring and leak from the wing and the skin would then have lost its function of preventing fuel leakage. For this location the functional impairment flaw size was set at 0.06 inches. This flaw size is the theoretical minimum for leakage and is probably conservative. Another example is the functional impairment flaw size for the dry (tab) hole at the rear spar, Figure 7. It was set at the critical flaw size, a point between the fastener hole and edge, when the flaw went unstable and the ligament was lost. This is also conservative in that "function" of the tab is not lost even at this size of flaw.

5.2 Crack Growth Analysis

During the original A-7 ASIP, all crack growth predictions for the durability analysis were performed using lab air growth rates. This assumption was considered justified since the economic limit is an average situation rather than a safety of flight or extreme situation. For the update, this precedent was followed only at locations with very small functional impairment flaw sizes (< 0.06 in.), where environment should not significantly impact the growth rate. At locations with larger functional impairment flaw sizes (> 0.06 in.), such as the wing fold rib or wing attach lug, analysis were performed in a severe environment.

Summary of Durability Analysis Results can be seen in appendix C.

Locations with more than 6000 FH but less than 8000 FH of economic life are A8 with 7400 FH and A14 with 7075 FH. At each of these locations, it takes over 2900 FH for flaws to grow from a detectable size (0.02 in.) to a repairable size (0.06 in.). If inspections are performed at intervals less than 2900 FH, flaws

can be detected before they become too large to repair.

The following location exhibit economic lives less than 6000 FH. They represent the highly loaded rear spar wet holes and the 2nd through 5th intermediate spar ends.

Rear Spar

Loc	Econ Life	Hole Number
A8	5900	1433-1442
AP	5325	1433-1442
A8	5475	1418-1432
A10	5300	1418-1432

25 Holes

Spar Ends

Loc	Econ Life	Hole Number
A2	4175	448A-448B
A3	5995	542-544
A4	5500	619-621
A5	5400	698-700

11 Holes

A minimum expense for enhancing 36 holes/side, along with scheduled inspections, can result in an airframe economic life in excess of 8000 FH.

5.3 Damage Tolerance Analysis

Damage tolerance addresses the safety of the structure. It is the ability of the airframe to resist failure due to the presence of flaws, cracks or other damage, for a specified period of unrepaid usage. These cracks can be brought on from initial material quality flaws through normal usage or rogue flaws such as nicks, scratches, gouges, and corrosion, through maintenance actions. Whatever causes the onset, the structure must be damage tolerant. Damage tolerance analysis determines the safety limit.

The safety limit of a component is the time it takes a rogue flaw to grow to the critical flaw size. The recommended inspection interval to insure the safety is usually set at one half the safety limit. This will provide two chances to detect the flaw before it reaches a critical Length.

5.3.1 Initial Rogue Types and Sizes and Residual Strength

As discussed in the criteria in Section 3, there are two flaw types (part thru from single point initiation and thru from multi-point

initiation). The rogue flaw sizes used for the damage tolerance analysis are 0.05 inches for the part thru type and 0.02 inches for the thru type. These are considered detectable sizes at inspection based on the geometry and location of each potentially critical point and the inspection techniques available.

Residual Strength refers to the ability of a structure to withstand failure in the presence of an unusually large stress. For this update, this is defined as the stress due to a load that occurs once in twenty lifetimes (one lifetime was assumed as 8000 FH). In other words, only one in twenty airplanes will see this load in its life. This requirement is used for structure considered in-service non-inspectable. This stress need be no larger than 1.2 times the design limit stress, however, it shall not be smaller than the design limit stress.

5.3.2 Crack Growth Analysis

During the original A-7 ASIP, all crack growth predictions for the damage tolerance analyses were performed using a severe environment (i.e. growth rate assumed half way between lab air and immersion in the water). This assumption was considered justified since these calculations considered safety. For this update that precedent was followed.

Damage tolerance analysis required generation of crack growth curves from the rogue flaw sizes to the critical flaw sizes for the baseline usage. The shortest safety limit for the lower skin at WS24.6 at the rear spar is 1800 FH for the thru type of rogue flaw. Applying a safety factor of two yields an optimum inspection interval of $(1800/2) = 900$ FH. This should give two opportunities to detect the flaw before it becomes critical.

Summary of Damage Tolerance Analysis
Results can be seen in appendix C.

The following locations have safety limits less than 3800 FH. These are all located along the rear spar.

Rear Spar Attach Rib Holes

Loc	Safety Limit	Hole Number
AC	1800	1404 1406-1417 13 Holes

Rear Spar Wet Holes

Loc	Safety Limit	Hole Number
-----	--------------	-------------

AE	2270	1433-1442
AF	2100	1433-1442
A8	2180	1418-1432
A10	2040	1418-1432

Enhanced for Dur.

Rear Spar Dry Holes

Loc	Safety Limit	Hole Number
A7	1920	1546-1552
A9	2050	1525-1544

27 Holes

The minimum safety limit is 1800 FH defined by location AC, rear spar skin holes just outboard of the attach rib. Using a safety factor of two yields an inspection interval of 900 FH.

Assuming the rear spar wet holes are enhanced for economic purposes, enhancing only 40 additional holes per side can increase the safety limit of the airframe to 3800 FH (Location AD); resulting in a minimum inspection interval of 1900 FH. The cost avoidance by eliminating a 900 FH inspection interval would justify the enhancement.

6 FORCE STRUCTURAL MAINTENANCE CONSIDERATIONS

The major objective of a damage tolerance analysis is to identify the structure that needs to be inspected, and then recommend inspection intervals and techniques to ensure that structure is safe to fly. These recommendations form the basis of the Force Structural Maintenance Plan.

6.1 Critical Location Summary

A critical location summary is tabularized in appendix D. These locations will require intermittent inspection to protect the safety of the airplane. Also included in the Table are critical flaw sizes. The crack growth for all locations transition to thru flaws before critical lengths are reached with the exception of location AC, the skin at the rear spar near WS24.6, the wingfold rib and the wing attach lug. The skin and wingfold rib can fail as either a part thru or thru flaw, depending on whether multi-initiation occurs. The wing attach lug should always fail as a part-thru flow as demonstrated by several full scale tests during the original ASIP and related NAVY programs.

6.2 Inspection Intervals and Techniques

Each A-7P phased to depot makes an ASIP inspection at all potentially critical locations. A scheme have been made to identify those airplanes with the largest risk and phase those into depot first. After this, optimum intervals are defined to subsequent inspection to control safety. It must be stressed that these intervals are in terms of Updated Baseline (UB) Flight Hours. The A-7P tracking program must be able to convert individual aircraft usage to UB Flight Hours.

An optimum inspection interval for each critical location was obtained by dividing the shortest safety limit by a safety factor of two. This gives two opportunities to detect the flaw before it becomes critical. These are tabulated in appendix D. Also included for information are surface eddy current inspection intervals, based on a surface detectable flaw size to failure. The shortest optimum in-hole eddy current inspection interval is defined by a rogue thru flaw for the lower wing skin at the rear spar outboard of WS24.6, location AC. The interval is 900 FH. Choosing the nearest interval for other locations without going over the optimum interval yields the recommendations shown as Option 1 in appendix E. A major inspection with wing removal every 1800 FH with intermediate inspections of the rear spar every 900 FH is required.

If the enhancements defined in Sections 5.2 and 5.3.2 are incorporated, the safety requirements reduce to a major inspection every 1900 FH as defined by Option 2 in appendix D.

Also included in appendix D are the recommended inspection procedures found in the 1A-7D-36 NDI manual. It is recommended that these procedures be updated to reflect the latest state-of-the-art techniques. In final review, the ASIP inspection plan for the A-7P has changed very little. The number of holes has increased slightly to include the intermediate spar ends where high fastener loading occurs. Also, holes to be inspected around the high field stress locations have been expanded somewhat. The major inspection interval has been lowered from 2000 FH to 1800 FH with an intermediate inspection of the rear spar. If enhancements are performed, the major inspection is only reduced to 1900 FH and no intermediate rear spar inspections are required.

A thorough damage tolerance analysis of the structure on the A-7P airplane coupled with proven inspection techniques ensure that the airplane can be flown safely until the service life of each is complete.

7 COMPOSITE OF ALL AIRCRAFT

Composite of all A-7P aircraft can be seen below:

COMPOSITE OF ALL AIRCRAFT			
TOTAL HOURS =	35091.5	GOOD HOURS =	28583.8
GOOD DATA EXCEEDANCES			
5G	6G	7G	8G
32662	6388	861	294
GOOD DATA EXCEEDANCES PER 1000 HRS			
5G	6G	7G	8G
1142.68	223.48	30.12	10.29

8 AIRCRAFT EXCEEDANCE DATA

The aircraft exceedance data can be seen in appendix F.

9 AIRCRAFT DAMAGE BY YEAR

The damage Index (DI) by aircraft and year for the next seventeen years can be seen in appendix G.

10 CONCLUSIONS

- 1 A size of 0.005 inches is still representative of A-7 initial quality.
- 2 There is higher probability of flaw growth from both sides of the hole than from one side.
- 3 The probability of multi-point initiation is sufficiently high that it should be accounted for in aircraft safety. Multi-point initiation is highly likely to occur from damaged holes, especially scratches from fastener installation. This has been verified by test.
- 4 Buffet cycles were identified during the flight strain survey, especially during manoeuvres at high angle of attack.
- 5 Buffet events in the stress sequence can reduce crack growth life by as much as 30 percent.
- 6 Testing provided increased confidence in analytical predictions.
- 7 Multi-point initiation leading to thru crack growth can reduce the life by over 50 percent.
- 8 Proper length fasteners are essential to good airframe life. Clamp-up can increase the life by a factor of two over non-clamped joints.

9 Additional critical locations have been identified. These are primarily the intermediate spar ends.

10 The minimum economic life is 4175 FH for the second intermediate spar end holes just outboard of the attach rib. Enhancing 36 holes per side and performing scheduled maintenance can increase this to over 8000 FH.

11 To ensure the safety of the airplane, a major inspection interval is required at 1800 FH with intermediate inspections of the rear spar every 900 FH. Enhancing 40 additional holes/side can eliminate the 900 FH inspection and increase the major inspection interval to 1900 FH, only slightly less than originally defined for the A-7P.

12 The damage tracking program must convert individual airplane usage to Updated Baseline Flight Hours so that airplanes can be identified for inspections as required.

10 LIST OF REFERENCES

1 Report No. 9-51220/9R-020, A-7P/TA-7P Aircraft Structural Integrity Program, dated April 1989.

2 Report No. 9-51220/9R-033, A-7P/TA-7P Structural Life Monitoring Computer Program User's Manual, dated 25 April 1989.

3 Report No. 9-51220/9R-114, A-7D ASIP Update - Volume I Scope, Methodology, Results, Conclusions, Recommendations, dated 22 December 1989.

4 Report No. 2-53440/7R-5928, Vol. 1, Damage Tolerance and Fatigue Assessment, dated 31 January 1977.

5 MCIC-HB-01, Damage Tolerance Design Handbook, dated May 1984.

6 SUN-1000, Practical Techniques for the Development of Mode I, Linear Elastic Stress Intensity Solutions for Typical Structural Details, dated 19 July 1985.

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Figure 2 - Categories of Cracking
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B1 Growth Rates for 7075-T651 in Lab Air
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C Summary of Predicted Economic Lives
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Appendix A1

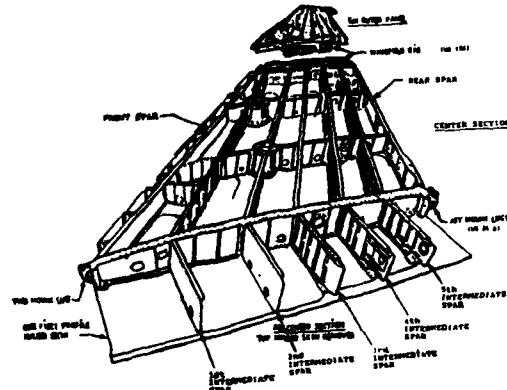


FIGURE 1.2 WING STRUCTURE

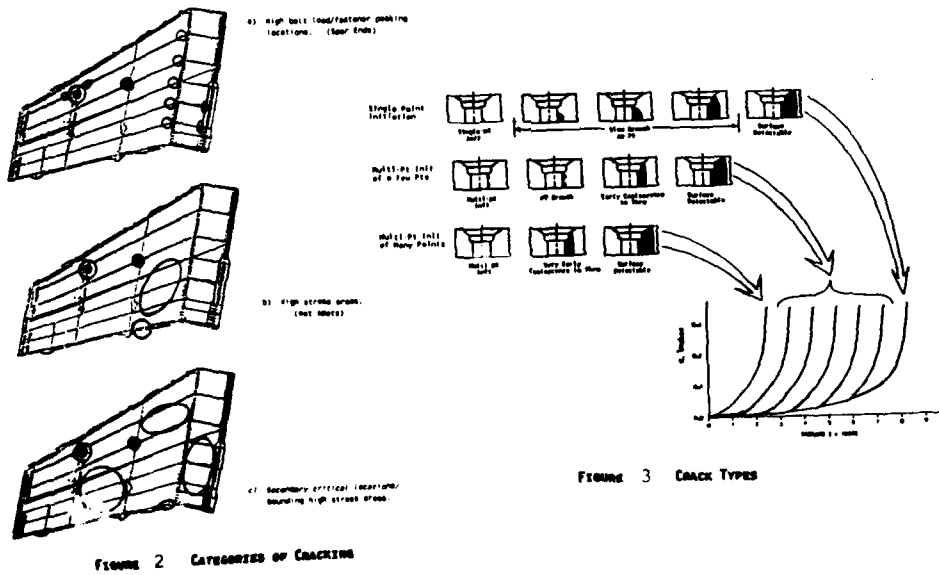


FIGURE 2 CATEGORIES OF CRACKING

FIGURE 3 CRACK TYPES

Appendix A2

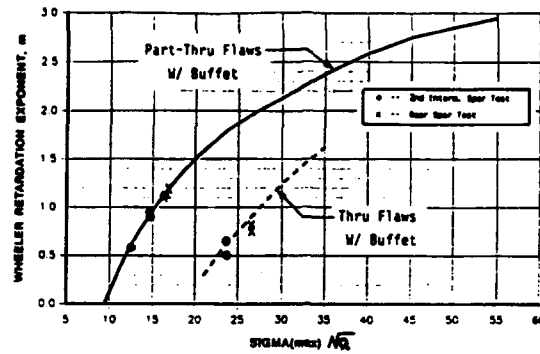


FIGURE 6 WHEELER RETARDATION EXPONENT FOR 7075-T651
ATTACK AIRCRAFT SPECTRUM

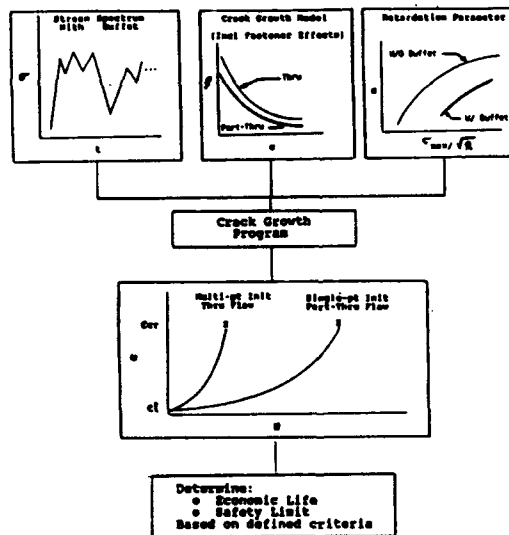
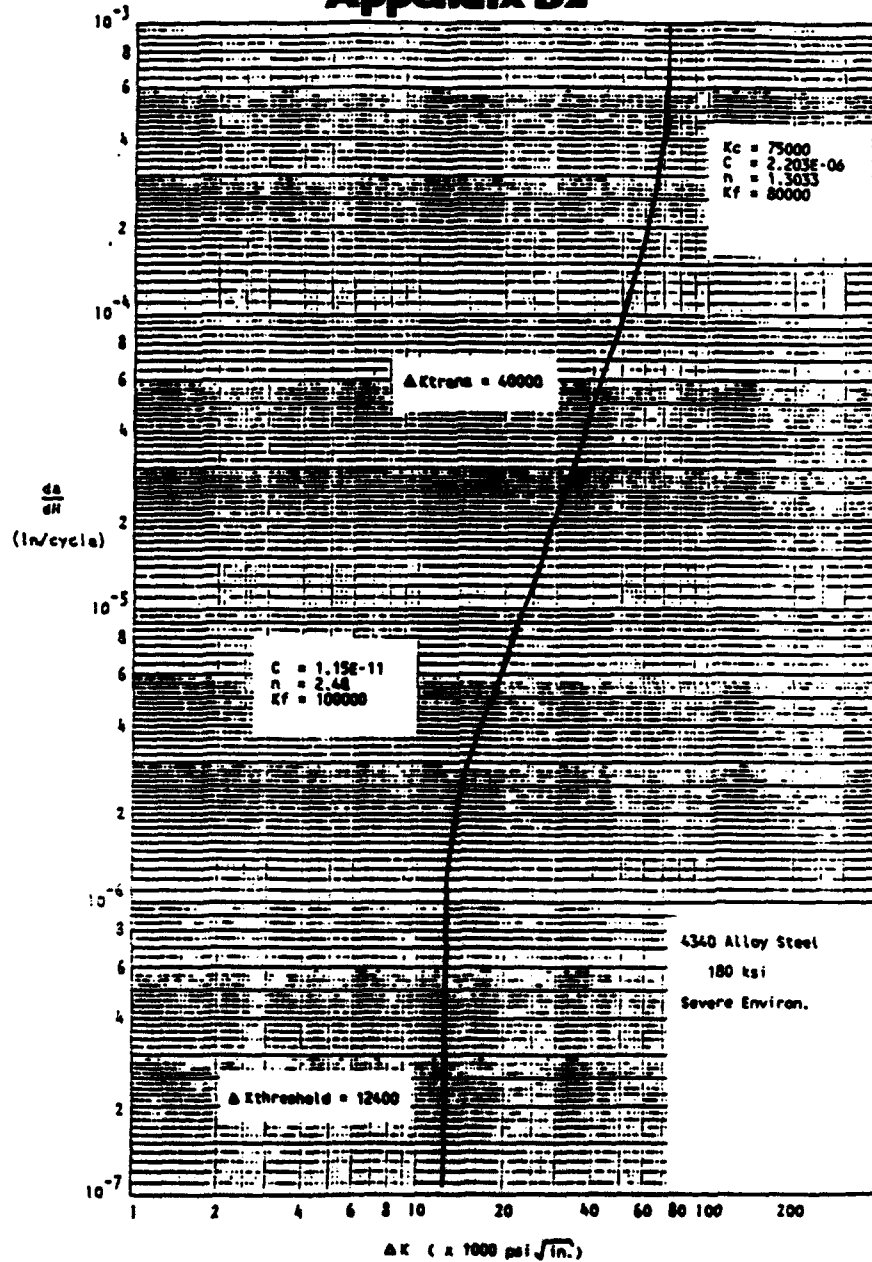


FIGURE 4 CRACK GROWTH METHODOLOGY

Appendix B2



GROWTH RATES FOR 4340 ALLOY STEEL, 180 KSI,
SEVERE ENVIRONMENT (Loc. A0)

Appendix C

SUMMARY OF PREDICTED ECONOMIC LIVES

Code	Description	Econ. Life Lab Air En PT005-CFI (UB FH)	Functional Impairment C/FI (inches)
AA	LWS, BL-0 @ 5th Intermediate Spar	9380	0.92
AB	LWS, BL-0 @ Rear Spar	9380	0.125
AC	LWS, WS-24.6 @ Rear Spar	12200	0.39
AD	LWS, WS-32.2 @ 5th Intermediate Spar	7400	0.04
AE	LWS, WS-53.7 @ Rear Spar (Wet Hole)	5900	0.04
AH	LWS, Center Pylon Aft Stub Hole	>13900	1.10
AI	LWS, Center Pylon Aft Stub Hole	>13900	1.10
AJ	LWS, Inb'd Pylon Aft Stub Hole	13900	1.10
AK	Boomerang Strap, WS-24.6 @ Rear Spar	>20000	1.10
AP	Lug Base, Wingfold Rib (OWP)	5325*	0.47
BA	Lug Hole, Wingfold Rib (OWP)	>20000	0.26
BB	Lug Hole, Wingfold Rib (OWP)	>20000	0.26
DA	PS480 Bulkhead Wing Aft Attach Lug	27000	0.46
A1	LWS, WS24.6 @ 1st Intermediate Spar	8970	0.04
A2	LWS, WS24.6 @ 2nd Intermediate Spar	4175	0.06
A3	LWS, WS24.6 @ 3rd Intermediate Spar	5595	0.06
A4	LWS, WS24.6 @ 4th Intermediate Spar	5500	0.06
A5	LWS, WS24.6 @ 5th Intermediate Spar	5400	0.06
A6	LWS, Inb'd Pylon @ 2nd Intermediate Spar	10300	0.06
A7	LWS, WS-53.7 @ Rear Spar (Dry Hole)	8250	0.35
A8	LWS, WS-50.0 @ Rear Spar (Dry Hole)	5475	0.06
A9	LWS, WS-50.0 @ Rear Spar (Dry Hole)	9300	0.34
A10	LWS, WS-50.0 @ Rear Spar (Dry Hole)	5300*	0.47
A11	LWS, WS-32.2 @ 4th Intermediate Spar	8900	0.06
A12	LWS, Cntr Pylon @ 1st Intermediate Spar	8740	0.06
A13	LWS, WS-32.2 @ 2nd Intermediate Spar	12500	0.06
A14	LWS, WS-88.0 @ 4th Intermediate Spar	7075	0.06

* Does not include continuing damage
UB FH - Updated Baseline Flight Hours

PT ... Part thru crack

SUMMARY OF PREDICTED SAFETY LIMITS

Code	Description	Safety Limit Severe Environment Inch-lbs Eddy Current		Critical Size Cer ** (inches)
		(PT00-x) (UB FH)	(T02-x) (UB FH)	
AA	LWS, BL-0 @ 5th Intermediate Spar	>10760	>5770	0.92
AB	LWS, BL-0 @ Rear Spar	10760	5770	0.92
AC	LWS, WS-24.6 @ Rear Spar	5300	1900	.33(.39)
AD	LWS, WS-32.2 @ 5th Intermediate Spar	5000	3000	1.21
AE	LWS, WS-53.7 @ Rear Spar (Wet Hole)	4900	2270	0.50
AH	LWS, Center Pylon Aft Stub Hole	>9900	>4000	1.10
AI	LWS, Center Pylon Aft Stub Hole	>9900	>4000	1.10
AJ	LWS, Inb'd Pylon Aft Stub Hole	8900	4000	1.10
AK	Boomerang Strap, WS-24.6 @ Rear Spar	>9900	>4000	1.10
AP	Lug Base, Wingfold Rib (OWP)	3050	2075	0.27
BA	Lug Hole, Wingfold Rib (OWP)	2000*	2100*	0.47
BB	Lug Hole, Wingfold Rib (OWP)	11100	5300	15(.28)
DA	PS480 Bulkhead Wing Aft Attach Lug	>11100	>5300	15(.28)
A1	LWS, WS24.6 @ 1st Intermediate Spar	13300	11300	4.80
A2	LWS, WS24.6 @ 2nd Intermediate Spar	8100	4360	1.55
A3	LWS, WS24.6 @ 3rd Intermediate Spar	7300	4770	1.47
A4	LWS, WS24.6 @ 4th Intermediate Spar	6700	4300	1.15
A5	LWS, WS24.6 @ 5th Intermediate Spar	6400	4200	1.19
A6	LWS, Inb'd Pylon @ 2nd Intermediate Spar	11800	7300	2.62
A7	LWS, WS-53.7 @ Rear Spar (Dry Hole)	2350	1920	0.35
A8	LWS, WS-50.0 @ Rear Spar (Dry Hole)	4700	2180	0.47
A9	LWS, WS-50.0 @ Rear Spar (Dry Hole)	2420	2060	0.34
A10	LWS, WS-50.0 @ Rear Spar (Dry Hole)	2720*	2040*	0.47
A11	LWS, WS-32.2 @ 4th Intermediate Spar	8350	4120	1.29
A12	LWS, Cntr Pylon @ 1st Intermediate Spar	13450	9120	2.74
A13	LWS, WS-32.2 @ 2nd Intermediate Spar	5900	4510	1.55
A14	LWS, WS-88.0 @ 4th Intermediate Spar	5700	4040	1.29

* Does not include continuing damage

PT ... Part thru crack

** Number in parenthesis is Part Thru Critical flaw size.

All others are Thru Critical flaw sizes

-- Not applied based on original ASIP testing of full scale components

UB FH - Updated Baseline Flight Hours

Appendix D

CRITICAL LOCATIONS AND OPTIMUM INSPECTION INTERVALS

-- BASELINE USAGE --

Code	Description	Safety Limit/2 (Inspection Interval) Severe Environment (UB FH)				Critical Size Ccr ** (in.)
		In-hole Eddy Current		Surface Eddy Current		
		(PT05-x)/2	(T02-x)/2	(PTdet-x)/2	(Tdet-x)/2	
AA	LWS, BL-0 @ 5th Intermediate Spar	>5375	>2005	--	--	0.92
AB	LWS, BL-0 @ Rear Spar	5375	2005	--	--	0.92
AC	LWS, WS-24.6 @ Rear Spar	2650	900	--	--	.33(.39)
AD	LWS, WS-32.2 @ 5th Intermediate Spar	2900	1900	735	1325	1.21
AE	LWS, WS-53.7 @ Rear Spar (Wet hole)	2450	1135	--	--	0.50
AH	LWS, Center Pylon Aft Stub Hole	>4450	>2000	--	--	1.10
AI	LWS, Center Pylon Fwd Post Hole	>4450	>2000	--	--	1.10
AJ	LWS, Inb'd Pylon Aft Stub Hole	4450	2000	--	--	1.10
AK	LWS, Inb'd Pylon Fwd Post Hole	>4450	>2000	--	--	1.10
AO	Boomerang Strap, WS-24.6 @ Rear Spar	1525	1040	--	--	0.30
AP	Lever Spar Cap, WS-53.7 @ Rear Spar	1400*	1050*	--	--	0.47
BA	Lug Base, Wingfold Rib (OWP)	5550	2050	--	--	.15(.26)
BB	Lug Hole, Wingfold Rib (OWP)	>5550	>2050	--	--	.15(.26)
DA	FS400 Bulkhead Wing Aft Attach Lug	7850	NA	--	--	(0.49)
A1	LWS, WS24.6 @ 1st Intermediate Spar	6650	5690	3830	4870	4.00
A2	LWS, WS24.6 @ 2nd Intermediate Spar	3050	2180	1300	1700	1.55
A3	LWS, WS24.6 @ 3rd Intermediate Spar	3650	2385	1050	1725	1.47
A4	LWS, WS24.6 @ 4th Intermediate Spar	3350	2150	1050	1725	1.19
A5	LWS, WS24.6 @ 5th Intermediate Spar	3300	2100	975	1515	1.19
A6	LWS, Inb'd Pylon @ 2nd Intermediate Spar	5800	3650	1980	2840	2.82
A7	LWS, WS-53.7 @ Rear Spar (Dry hole)	1175	960	--	--	0.35
A8	LWS, WS-50.0 @ Rear Spar (Wet hole)	2350	1090	--	--	0.47
A9	LWS, WS-50.0 @ Rear Spar (Dry hole)	1210	1025	--	--	0.34
A10	Lever Spar Cap, WS-50.0 @ Rear Spar	1380*	1020*	--	--	0.47
A11	LWS, WS-32.2 @ 4th Intermediate Spar	3175	2080	925	1470	1.29
A12	LWS, Cntr Pylon @ 1st Intermediate Spar	6725	4560	2680	3800	2.74
A13	LWS, WS-32.2 @ 2nd Intermediate Spar	2950	2255	1185	1570	1.55
A14	LWS, WS-60.0 @ 4th Intermediate Spar	2050	2020	915	1400	1.29

* Does not include continuing damage

** Number in parenthesis is Part Thru Critical flaw size

All others are Thru Critical flaw sizes

-- Surface eddy current inspections do not apply

UB FH - Updated Baseline Flight Hours

PT -- Part thru crack

T -- Thru thickness crack

Appendix E

RECOMMENDED INSPECTION PLAN

Code	Description	Option 1 (No Enhance.) Intervals*		Option 2 (W/Enhance.) Intervals*		Inspection Procedures **	Special Considerations
		900	1000	900	1000		
AA	LWS, SL-0 @ 5th Intermediate Spar		X	X		Figure 6-10	Wing Removal Required
AB	LWS, SL-0 @ Rear Spar		X	X		Figure 6-10	Wing Removal Required
AC	LWS, WS-24.6 @ Rear Spar	X	X	X		Figure 6-3, 6-8	
AD	LWS, WS-32.2 @ 5th Intermediate Spar		X	X		Figure 6-12	
AE	LWS, WS-33.7 @ Rear Spar (Wet hole)	X	X	X		Figure 6-9	
AH	LWS, Center Pylon A/R Slab Hole		X	X		Figure 6-4	Pylon Removal Required
AI	LWS, Center Pylon Fuel Pool Hole		X	X		Figure 6-3	Pylon Removal Required
AJ	LWS, In/d Pylon A/R Slab Hole		X	X		Figure 6-2	Pylon Removal Required
AK	LWS, In/d Pylon Fuel Pool Hole		X	X		Figure 6-3	Pylon Removal Required
AO	Boomerang Strap, WS-24.6 @ Rear Spar	X	X	X		Figure 6-3, 6-8	
AP	Lower Spar Cap, WS-33.7 @ Rear Spar	X	X	X		Figure 6-9	
BA	Lug Base, Wingfold Rib (OWP)		X	X		Figure 6-6	
BB	Lug Hole, Wingfold Rib (OWP)		X	X		Figure 6-11	
BA	PD400 Bulkhead Wing A/R Attach Lug		X	X		Figure 6-7	Wing Removal Required
A1	LWS, WS24.6 @ 1st Intermediate Spar		X	X		Figure 6-12	
A2	LWS, WS24.6 @ 2nd Intermediate Spar		X	X		Figure 6-12	
A3	LWS, WS24.6 @ 3rd Intermediate Spar		X	X		Figure 6-12	
A4	LWS, WS24.6 @ 4th Intermediate Spar		X	X		Figure 6-12	
A5	LWS, WS24.6 @ 5th Intermediate Spar		X	X		Figure 6-12	
A6	LWS, In/d Pylon @ 2nd Intermediate Spar		X	X		Figure 6-12	
A7	LWS, WS-33.7 @ Rear Spar (Dry hole)	X	X	X		Figure 6-9	
A8	LWS, WS-36.0 @ Rear Spar (Wet hole)	X	X	X		Figure 6-9	
A9	LWS, WS-36.0 @ Rear Spar (Dry hole)	X	X	X		Figure 6-9	
A10	Lower Spar Cap, WS-36.0 @ Rear Spar	X	X	X		Figure 6-9	
A11	LWS, WS-32.2 @ 4th Intermediate Spar		X	X		Figure 6-12	
A12	LWS, Ctr Pylon @ 1st Intermediate Spar		X	X		Figure 6-12	
A13	LWS, WS-32.2 @ 2nd Intermediate Spar		X	X		Figure 6-12	
A14	LWS, WS-36.0 @ 4th Intermediate Spar		X	X		Figure 6-12	

* Intervals are Updated Baseline Flight Hours

** Refer to 1A-70-35 Inspection Manual

AIRCFT LAST		GOOD + GAP FILLED DATA		GOOD		GOOD DATA ONLY		EX 7G		EX 6G		EX 5G		EX 4G		EX 3G		EX 2G		EX 1G	
SN	BASE	FW	FT	5G	6G	7G	8G	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	EXC	
5502	MONR	1173.4	1391.	270.0	39.0	12.0	1173.4	1391	270	39	12	1185.4	230.1	33.24	10.23						
5503	MONR	1386.5	1641.	329.0	43.0	16.0	1386.5	1641	329	43	16	1183.6	237.3	31.01	11.54						
5504	MONR	1172.3	1246.	220.0	27.0	8.0	1168.8	1246	220	27	8	1056.1	188.2	23.10	6.84						
5506	MONR	1022.4	1131.	215.0	29.0	10.0	898.5	1131	215	29	10	1258.8	239.3	32.28	11.13						
5507	MONR	1319.5	970.	205.0	29.0	9.0	782.2	970	205	29	9	1240.1	262.1	37.07	11.51						
5508	MONR	955.7	1279.	223.0	25.0	6.0	953.1	1279	223	25	6	1342.0	234.0	26.23	6.30						
5509	MONR	586.3	57.	9.0	1.0	1.0	192.4	57	9	1	1	236.2	46.8	5.20	5.20						
5510	MONR	706.8	889.	169.0	24.0	6.0	706.8	889	169	24	6	1257.7	239.1	33.95	8.49						
5511	MONR	1074.6	401.	39.0	1.0	0.0	475.4	401	39	1	0	843.5	82.0	2.10	0.00						
5512	MONR	640.1	778.	151.0	21.0	7.0	640.1	778	151	21	7	1215.5	235.9	32.81	10.94						
5513	MONR	722.2	955.	180.0	24.0	8.0	722.2	955	180	24	8	1372.4	249.2	33.23	11.08						
5514	MONR	1074.0	1158.	171.0	20.0	5.0	1074.0	1158	171	20	5	1028.2	159.2	18.62	4.66						
5515	MONR	1129.1	957.	157.0	26.0	9.0	982.1	957	157	26	9	974.5	159.9	26.47	9.16						
5516	MONR	277.3	393.	89.0	15.0	3.0	277.3	393	89	15	3	1417.1	320.9	54.09	10.82						
5517	MONR	992.8	129.	14.0	1.0	0.0	199.4	129	14	1	0	646.9	70.2	5.01	0.00						
5519	MONR	738.9	745.	132.0	15.0	5.0	738.9	745	132	15	5	1008.2	178.6	20.30	6.77						
5520	MONR	861.5	53.	21.0	7.0	3.0	26.9	53	21	7	3	1969.5	280.4	260.12	111.48						
5521	MONR	893.2	424.	73.0	8.0	6.0	319.5	424	73	8	6	1327.1	228.5	25.04	18.78						
5522	MONR	1116.9	1377.	265.0	38.0	15.0	1110.8	1377	265	38	15	1239.6	238.6	34.21	13.50						
5523	MONR	898.6	261.	39.0	0.0	0.0	371.8	261	39	0	0	701.9	104.9	0.00	0.00						
5524	MONR	571.5	32.	6.0	4.0	1.0	29.5	32	6	4	1	1084.7	203.4	135.59	33.90						
5525	MONR	1042.8	1175.	233.0	29.0	10.0	1031.3	1175	233	29	10	1139.4	225.9	28.12	9.70						
5526	MONR	900.7	163.	19.																	

MANAGING AIRBORNE ASSETS THROUGH LOADS MONITORING

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SUMMARY

Loads monitoring was a cornerstone of the Air Force Structural Integrity Program from its inception. But the high cost of new systems has provided a new need for the program. It provides the hard data that is required by the systems manager to determine the economic life of their aircraft and the most efficient allocation of aircraft to maintain optimum operational capability. Continued advances in solid state microprocessors and integrated software along with advances in storage media should enhance the capability of the overall program.

1. INTRODUCTION

As a result of a rash of fatigue related failures on aircraft structural components in the late 1950's, the U.S. Air Force developed an Aircraft Structural Integrity Program (ASIP). It was established as a requirement by the issuance of Air Force Regulation 80-13 in 1969. This regulation establishes the chain of command for the management of the ASIP program for the Air Force Fleet as shown in the Figure 1. The regulation outlines the entire program in terms of responsibility such as purpose, funding, required mil-specs and data to support this type of effort. The technical backbone of this effort is contained in MIL-STD-1530A and AFGS-87221A which describes the effort required for the program. Table 1 illustrates the scope of this effort. Performance of these tasks will identify the critical areas of the airframe and establish the inspection/modification requirements for them. This information is necessary for fleet management. However, it is the Individual Aircraft Tracking Program (IATP) and the Load Environment System Survey (L/ESS) that provides the actual usage data that establishes the schedules for managing the force for using commands such as Tactical Air Command, Strategic Air Command, Air Training Command, and Military Airlift Command.

The paper will cover the following topics related to the management of Air Force fleets such as replacement costs for current assets, installation of new recorders on old and new aircraft for loads monitoring, standardization of loads recorders, data presentation, design consideration and future recording systems.

2. ASSETS

The ASIP has reduced the number of fatigue related failures but time and events have increased the replacement costs of aircraft through inflation and technology enhancement. This fact has increased the importance of extracting as much useful life out of airframe structure without reducing safety of flight because of high replacement costs. The following two examples, one of a cargo aircraft and the other a fighter aircraft will be used to illustrate the increased cost factors. The first example is the C-5A manufactured in 1971 at a cost of \$36.5M per copy for airframe and engines. Later, the Air Force decided that additional aircraft were needed for the airlift role. The production line was reopened, and in 1988, the C-5B airframe and engine costs were \$111.7M per aircraft. This effort did not include some of the typical tooling and development cost associated with a new design but did include a new wing. The avionics were also upgraded at a cost of \$25M. The product improvement presents a delta \$100.2M increase over the original cost per aircraft. The second example of this cost increase is illustrated in the production of the F-16 from 1975 to present. Table II represents the USAF production costs for the F-16 with product improvement. The chart shows a base line cost of \$4.1M for the F-16A in 1975 and a current cost of \$11.3M in 1990. The delta cost increase in the production of the F-16A is \$7.2M along with attendant product improvement. Cost increases such as these are putting increased pressure on the Air Force to extract as much life out of these systems as possible through use of the ASIP.

3. LOADS MONITORING

Loads monitoring is used by the ASIP program to record the actual usage of aircraft so that the life of the airframe can be measured against the original design parameters. The program has worked well, but some of the recording equipment has become obsolete because of the extended life of the airframes. Two examples of this are the Whittaker Recorder used on the F-111 and the more commonly used Conrac MXU 533A. The manufacturer for the F-111 recorder went out of business and no spare parts are available for repair. Air Force Logistics Command (AFLC) maintenance has kept them working, but the recorder life is limited through attrition. The F-111 still is an active operational aircraft that needs loads monitoring because it has unique capabilities that are economically difficult to replace with a new aircraft.

The other example is the MXU 533A which is used on the A-10, E-3, C-130, F-5, F-15 and F-16. It has the capability of storing 12 megabytes of continuous data on an 8 track

tape. The data on the tape has to be transcribed and reformatted to be compatible with AFPLC computers where it is compressed to a ratio of 30 or 40 to one using special programs depending on aircraft type.

The recorder was originally intended to operate in a 7.33g environment and was not redesigned to operate in the expanded 9+g flight envelope. The recorder tape drive and recording heads were not capable of recording data consistently in this severe environment. Its major faults have been short frames (data lacking on the tape), parity errors (timing was off and data could not be transcribed from the tape) and the recorder may be broken for six months before being removed (AFPLC takes three to five months to process the data). This has given the Air Force the impetus to investigate the use of solid state microprocessor recorders on new aircraft. The first major system to develop a solid state microprocessor recording system, was the B-1B. It is a 500K 8-bit microprocessor with Electronically Erasable Programmable Only Memory Chips manufactured by ElectroDynamics. The recorder currently handles 120 L/ESS, 28 IATP, 128 Mishap and 5 engine parameters. Software in the computer compresses the data so that only useful data is stored in memory. Once the memory is filled, the data is extracted, placed on a floppy disk and sent to AFPLC to be reformatted and analyzed. The advantage of this system is lighter weight, solid state reliability, reprogrammability, and low power requirements. A derivative of this recorder was developed for the T-46A but the program was cancelled. Many recording systems are now available for use on aircraft and have similar capabilities.

4. STANDARDIZATION

In order to prevent a proliferation of solid state microprocessors on aircraft, the F-16 SPO and the Avionics SPO started a program to develop a standard flight data recorder which would replace the MXU 533A on the F-16 and provide a replacement recorder for sixteen other aircraft which currently have obsolete recorders. The original concept was to use F-16 design requirements and a draft Tri Service Specification to design the Flight Recorder. The program turned out to be very complex due to the age of the aircraft involved, local environments for flight recorders, instrumentation available and method of analysis. Two major studies had to be performed to resolve the issues on vibration environments and parameters. The vibration study consisted of surveying the recorder location on the aircraft and comparing the local vibration spectrum to that for the Standard Recorder. One example of this deviation from the standard vibration spec can be found in the A-7F vibration spec at the flight recorder location which necessitated use of shock mounts to reduce the PSD spectrum to design requirements (see Figure 2. Endurance vibration requirement SAU-YA-7F). The A-7F was not an isolated case, so the recorder had to be modified in a similar fashion for the other aircraft. The other issue related to the number of parameters and their associated aircraft instrumentation which was the result of having 16 individual ASIP programs of varying complexity (see Table III. Number of signals versus parameters and Table IV. Parameter errors versus aircraft).

The span of these parameters range from 240 for the C-17 to 7 for the T-43. The tables show some of the technical difficulties present in the old system that had to be compensated for in the new system by software. Also, the data had to be collected in a format suitable for use by the system managers.

5. DATA PRESENTATION

The collection of data is very important, but the presentation of the data allows the managers instant access to the status of the particular Mission Design Series aircraft he is responsible for. Two reports are generally available, the L/ESS and the IATP report or a single report called a Service Life Monitoring report. The IATP is a technical document which describes the areas where cracks will occur, describes the analysis approach used to mathematically grow the cracks and contains crack lengths for various location. The L/ESS is a useful to the managers because it shows how the aircraft are being used. It has numerous forms of usage data (see Figures 3 through 5). It allows him to assess usage by air base and mission and make recommendations for on aircraft inspection, rotation, maintenance, and operational usage. The data in IATP quarterly reports can be tailored to meet operational and maintenance needs. In reviewing US Air Force reports, you will see variations in the data presented in these reports based on the command and mission of the aircraft. The format for the reports are developed during the early production phase of the aircraft. It is a coordinated effort between Aeronautical Systems Division (ASD), AFPLC, and the contractor to develop the format, analysis, data storage and data retrieval for the reports. Even though the data has historical value, it is used for other purposes than loads monitoring.

6. DESIGN

Loads data is not only used to evaluate current aircraft, but can be used to develop design criteria for new aircraft. Generally, data is provided in AFCS-87221A for the various type aircraft, e.g., cargo, fighter, bomber and trainer. The information available in this guide is based on compilation of generic data obtained on aircraft type and may not be suited for a specific system. The criteria may not be severe enough, or too severe depending on mission requirements. In the case where the mission is rather mundane, the files are searched for data, e.g., design handbook, or actual

L/ESS reports generated for a particular system. The data is reviewed and compiled to generate usage data for the new system. As an example of how this information is applied, just recently in a Class II mod for the KC-135 the contractor needed loads/service usage data. The University of Dayton Research Institute reports vintage 1968 had the appropriate usage data but lacked landing touch and go data. We recommended the contractor use C-141 landing touch and go data on a similar mission provided in the Design Handbook to fill this data gap. This satisfied the immediate need to establish requirements. Based on actual usage, it will be determined later through L/ESS and IAT how correct the original design requirements were. Another case where aircraft requirements exceed previous technology, the loads data has to be projected beyond current design limits, e.g., going from 7.3g fighter like an F-4 to a 9g fighter like an F-16. Fighter usage data (see Figures 6 through 8) is reviewed from such exercises like Red Flag, or southeast Asia combat data or the most severe usage data from air bases known to have aggressive missions. This data is tabulated and extrapolated to project future usage of such a high performance aircraft for design purposes. This data has to be verified by actual L/ESS and IATP data. Generally, it is easier to use the set criteria in the specs because it's use through the years has been proven and there is a general acceptance of the data. The use of more severe environment from a loads standpoint pushes the state of art which affects fatigue life of airframe, functioning of internal systems, moving parts, bearing lubrication, etc. and even pilot physiology. Detailed data is necessary to understand this environment and how it affects the man machine interface.

7. FUTURE RECORDING SYSTEMS

The transition from mechanical tape recorders has been made to the use of solid state microprocessors. These have many advantages over the tape recorder but will suffer the same fate as it's counterpart, e.g. becoming obsolete through technology advances and excessive replacement costs due to lost manufacturing technology. The current preferred approach is to integrate the Flight Data recorder function into the newer multifunctional computers found in modern avionics. Instead of a separate piece of avionics, it becomes a software function on one or more of an aircraft on board computers. The advantages to this approach is elimination of the need to procure a separate piece of avionics and the attendant impact on Environmental Control System, tech orders, maintenance and training. As part of flight avionics, reliability will be greater because it will be located in the more desirable location in the aircraft. The system, when integrated with maintainee data collection, will be current because the dates will have to be collected everyday and will be updated every time the avionics suite is changed.

TABLE I
USAF aircraft structural integrity program tasks

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANALYSIS AND DEVELOPMENT TESTS	FULL SCALE TESTING	FORCE MANAGEMENT DATA PACKAGE	FORCE MANAGEMENT
ASIP MASTER PLAN	MATERIALS AND JOINT ALLOWABLES	STATIC TESTS	FINAL ANALYSES	LOADS/ENVIRONMENT SPECTRA SURVEY
STRUCTURAL DESIGN CRITERIA	LOAD ANALYSIS	DURABILITY TESTS	STRENGTH SUMMARY	INDIVIDUAL AIRPLANE TRACKING DATA
DAMAGE TOLERANCE & DURABILITY CONTROL PLANS	DESIGN SERVICE LOADS SPECTRA	DAMAGE TOLERANCE TESTS	FORCE STRUCTURAL MAINTENANCE PLAN	INDIVIDUAL AIRPLANE MAINTENANCE TIMES
SELECTION OF MAT'L'S, PROCESSES, & JOINING METHODS	DESIGN CHEMICAL/THERMAL ENVIRONMENT SPECTRA	FLIGHT & GROUND OPERATIONS TESTS	LOADS/ENVIRONMENT SPECTRA SURVEY	STRUCTURAL MAINTENANCE RECORDS
DESIGN SERVICE LIFE AND DESIGN USAGE	STRESS ANALYSIS	FLIGHT VIBRATION TESTS	INDIVIDUAL AIRPLANE TRACKING PROGRAM	
	DAMAGE TOLERANCE ANALYSIS	FLUTTER TESTS		
	DURABILITY ANALYSIS	INTERPRETATION & EVALUATION OF TEST RESULTS		
	SONIC ANALYSIS			
	VIBRATION ANALYSIS			
	FLUTTER ANALYSIS			
	NUCLEAR WEAPONS EFFECTS ANALYSIS			
	NON-NUCLEAR WEAPONS EFFECTS ANALYSIS			
	DESIGN DEVELOPMENT TESTS			

TABLE II
MULTISTAGE IMPROVEMENT PROGRAM
UNIT COST TRACK

	F-16A BLOCK 15	F-16A BLOCK 30	F-16C BLOCK 40	F-16C BLOCK 50	
TY \$	\$8.8 FY 82	\$10.6 FY 86	\$13.0 FY 89	\$15.6 FY 93	
FY 90\$	\$11.3	\$11.5	\$12.4	\$13.0	
BY 75\$	\$4.1	\$4.2	\$4.5	\$4.7	
	COST OF IMPROVEMENT				
					COST OF INFLATION

TABLE III
Numbers of signals versus aircraft

Aircraft	Signal Type			
	Digital	Analog	Discrete	Total
YA-7F	22	11	3	36
A-37	0	12	11	23
B-52	2	26	12	40
C-17	233	7	0	240
C-130	0	18	4	22
C-135	13	14	7	34
C-141	0	23	6	29
E-3	0	6	7	13
F-5	0	14	8	22
F-5N	0	8	1	9
F-15A-D	34	24	27	85
F-15E	89	11	9	109
F-16A/B	39	13	38	90
F-111	55	18	18	91
T-37	0	13	5	18
T-38	0	12	3	15
T-43	0	6	1	7

TABLE IV
Parameter errors versus aircraft

Parameter	Aircraft							
	C-130	C-135	C-141	E-3	F-15	F-16	T-37	T-38
Altitude	A-2,1	A-2	A-2,1	A-1,2	A-2,1	A-2,1		A-1
Airspeed	A-2,1	A-2,1	A-2,1	A-1,2	A-2,1	A-1		A-3,2
Taxi speed	A-2		A-2					
Long acceleration					A-1,3	A-3,2		
Lateral acceleration				A-1	A-3	A-2	A	A-2
Vertical acceleration	C-2	B-2,3	C-2	B-3	B-1,2	A-2	B	B-3
Roll rate						A-2	A	A-1
Pitch rate							A	A-1
Yaw rate							A	A-1
Roll acceleration					C	A-2		
Angle of attack					A-4			
Power lever angle						A-3,2,1		
Aileron position	B-1		B-1		C-1,2,5		C-2,1,3	C
Flap position	B-1		B-1			C-1,3	C-2,1,3	
Elevator position						B-1,3		C
Hor. tail position						C-3,1	C-2,1,3	C
Rudder position	B-1		B-1		C-1,2,5			
Flap position	B-1		B-1	A-2				
Fuel quantity		A			A-2,4	A		
Strain	C-1,2	C-2,3	C-1,2	C-3,1,2				
Weapon count					A-3			
Speedbrake position					A			
Landing gear position					A			
Autopilot switch		B-2						
Weight on wheels		A						

Notes

The letter code denotes the frequency of failure:

A-Uncommon, B-Occasional, C-Common

The number codes denote the failure modes:

1-Biased, 2-Frozen, 3-Eratic, 4-Out of Range, 5-Reversed

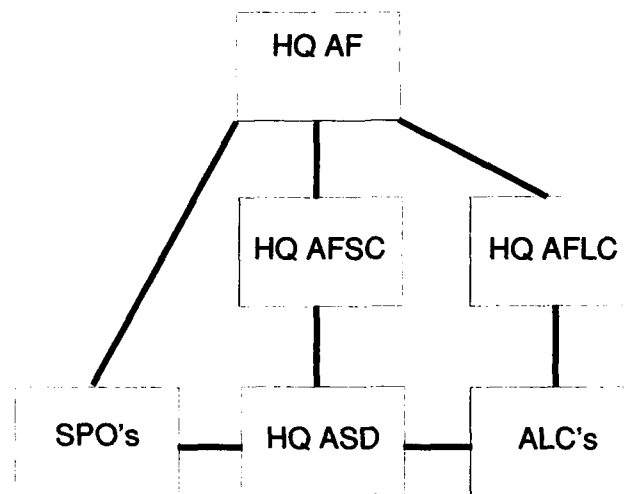


Figure 1. Chain of Command

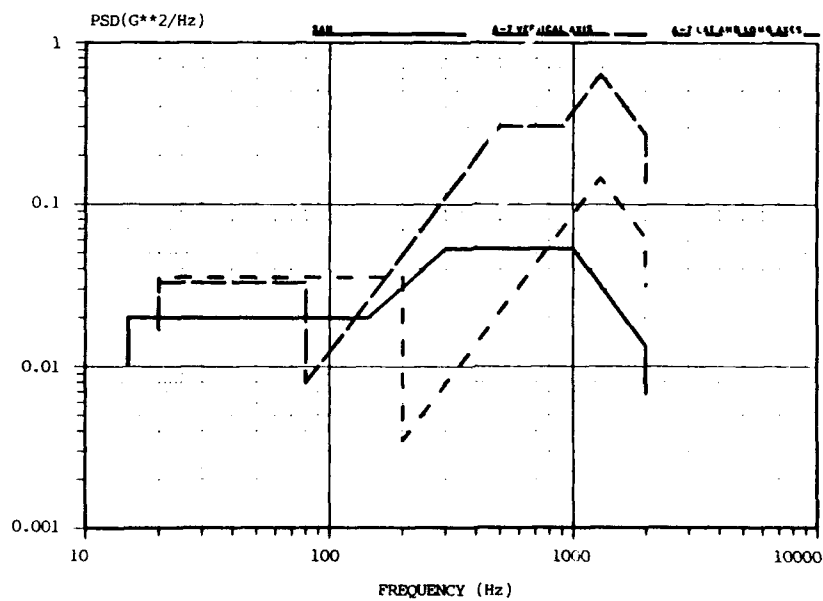


Figure 2. Endurance vibration requirement W-7F

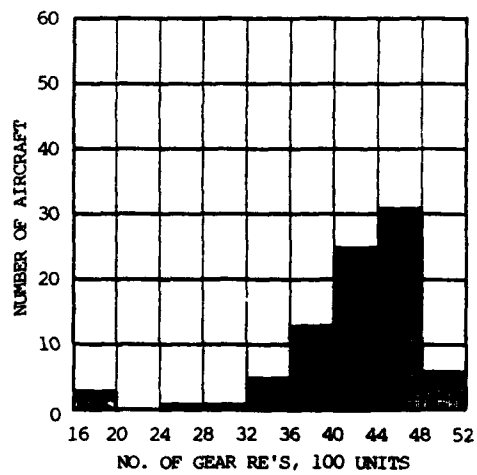
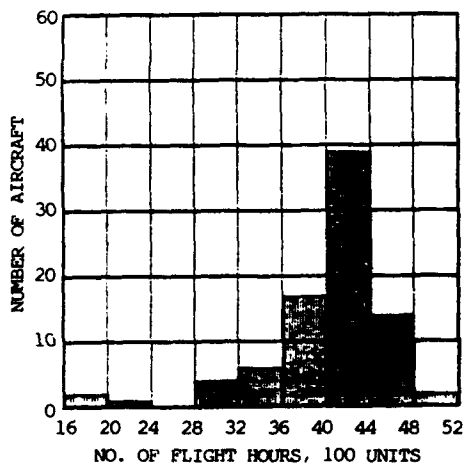
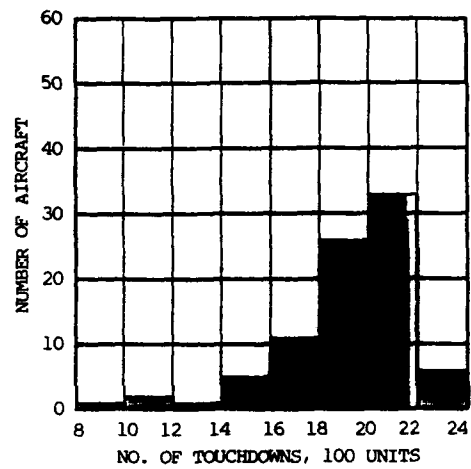
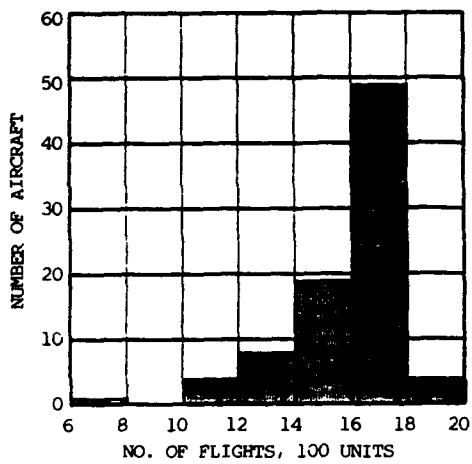


Figure 3. Histograms of usage

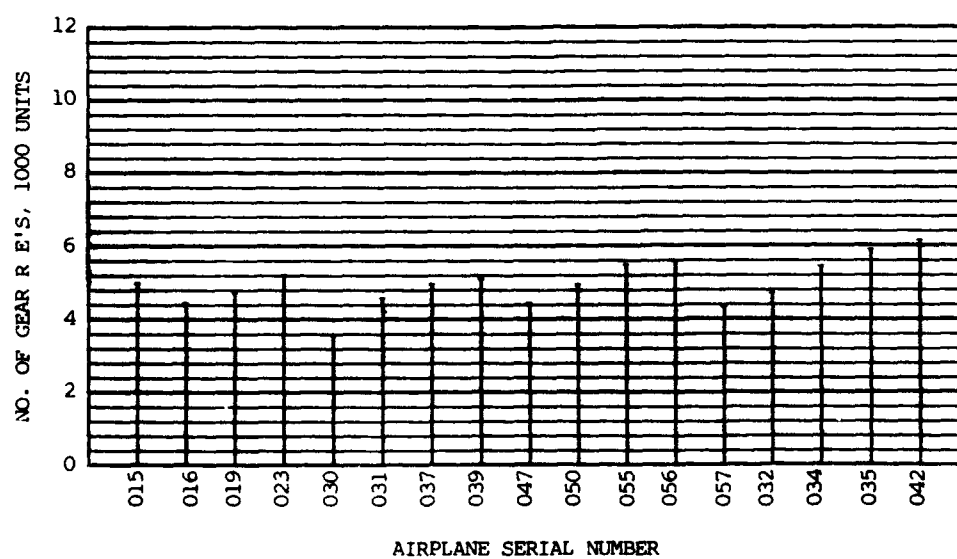
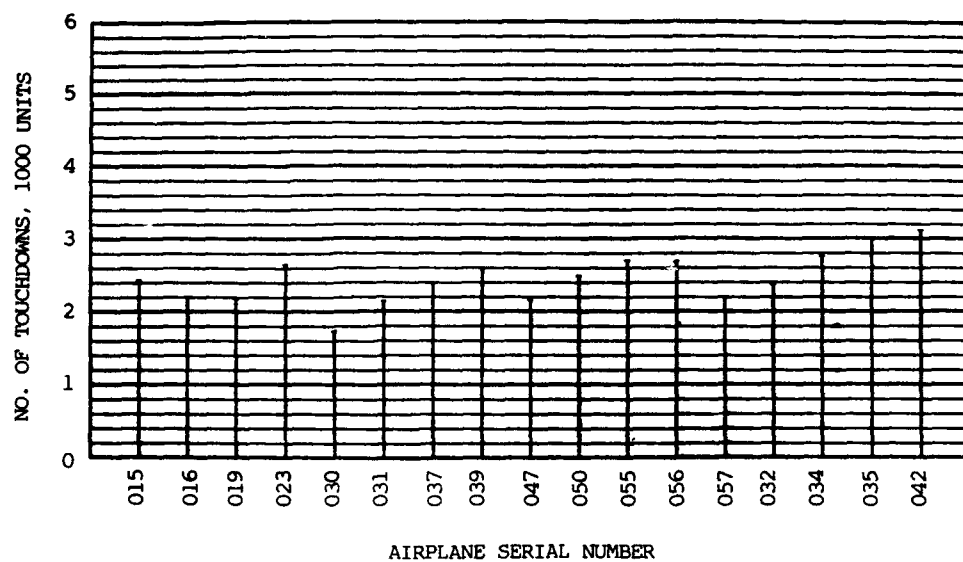
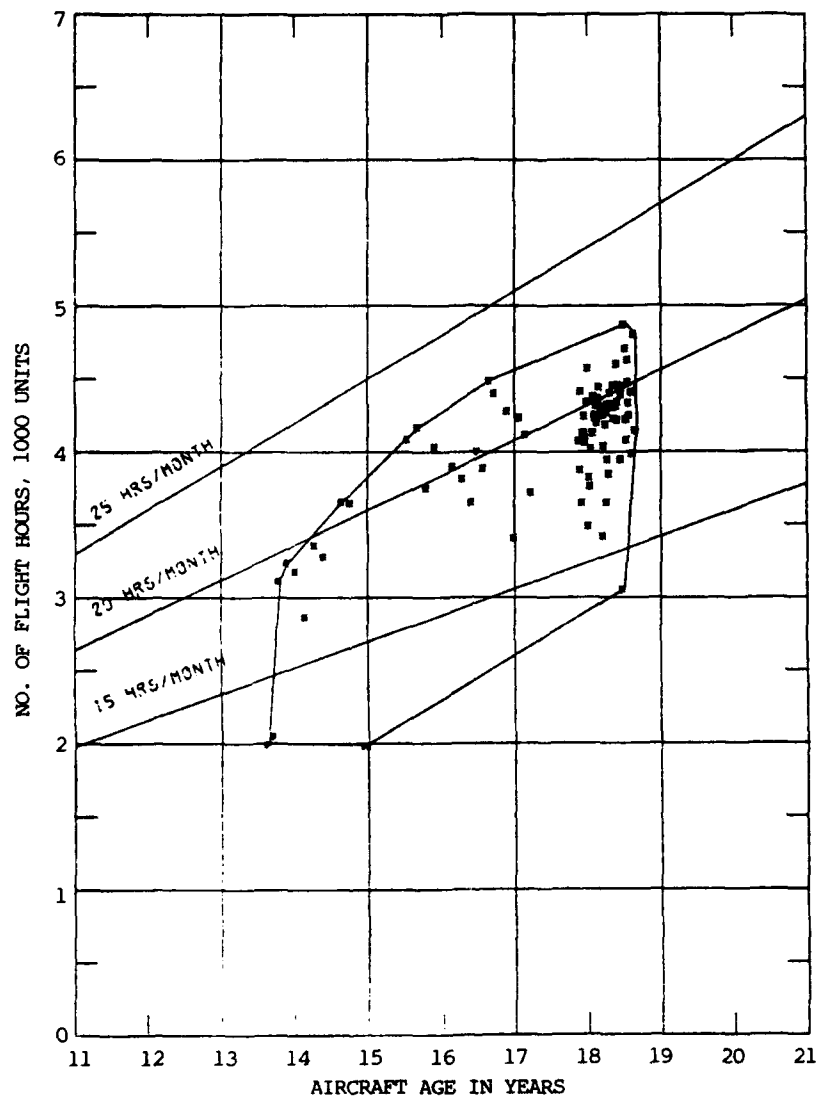


Figure 4. Aircraft bargraphs of usage thru date organization location



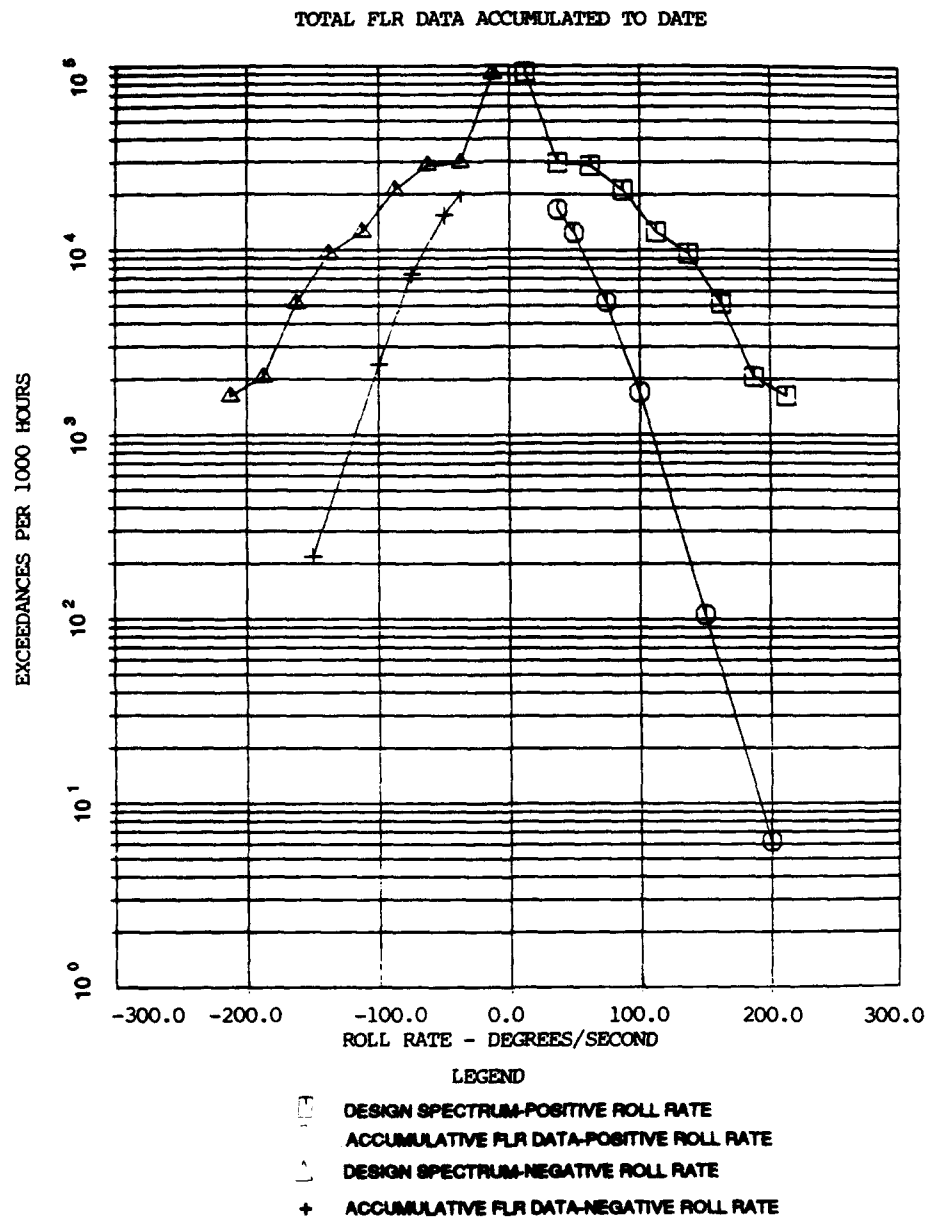


Figure 6. Comparison of roll rate peak exceedances design spectrum versus flr data for 1000 flight hours

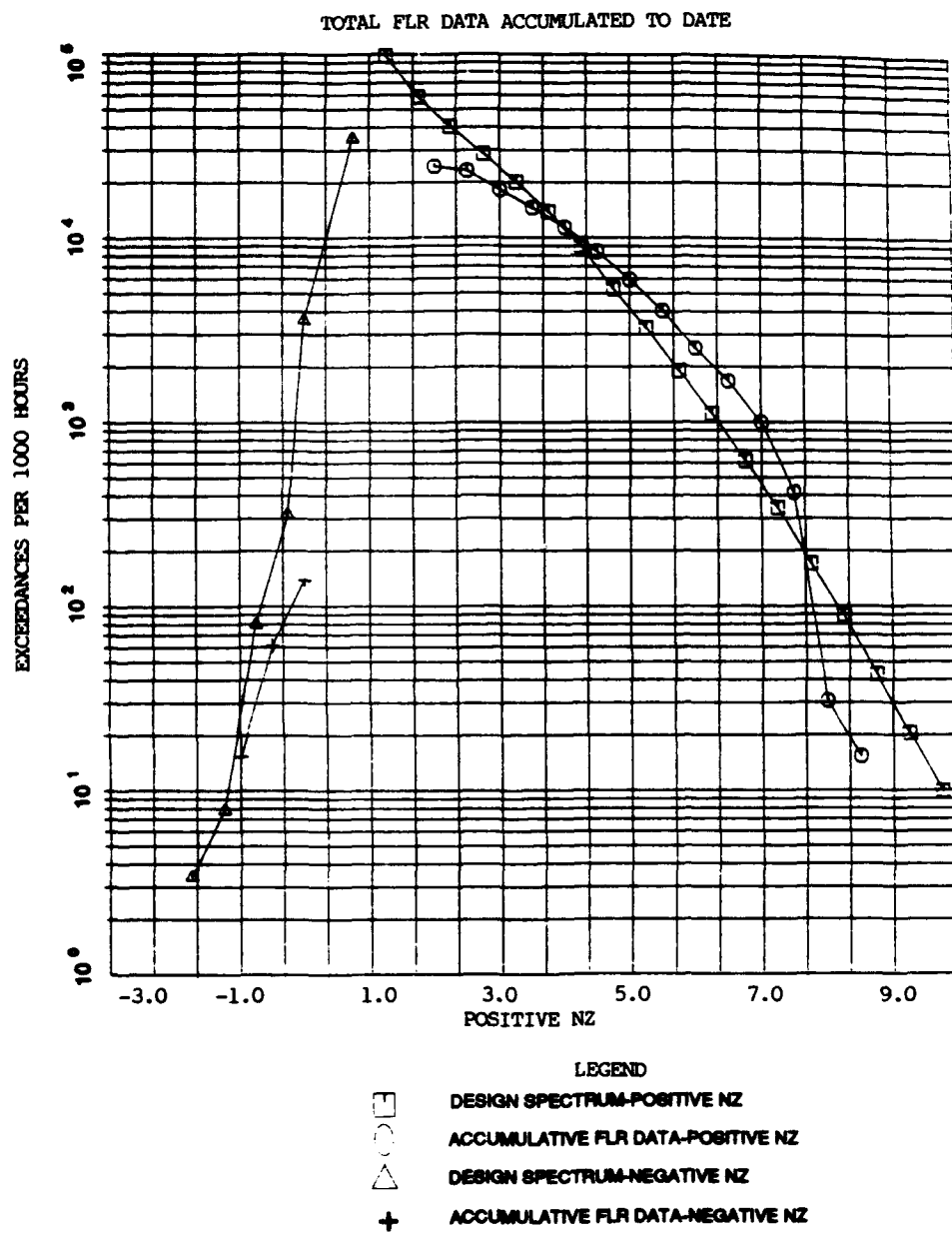


Figure 7. Comparison of nz peak exceedances design spectrum versus flr data for 1000 flight hours

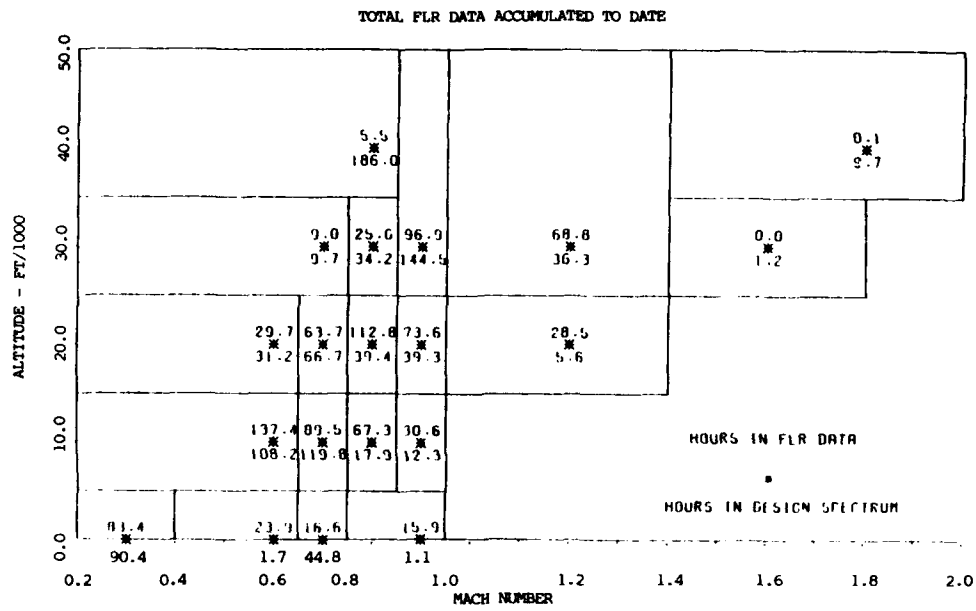


Figure 8. Comparison of flight hours by mach/altitude block design spectrum versus flr data for 1000 flight hours

**APPROACH TO CREW TRAINING IN SUPPORT OF THE USAF
AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)**

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SUMMARY

Maintaining the safety and strength of an aircraft is dependent upon the capability of appropriate Air Force commands to perform maintenance and inspections throughout the service life of the aircraft. One of the maintenance action involves the collection and reporting of operational usage data to support the Loads/Environment Spectra Survey (L/ESS) and Individual Aircraft Tracking (IAT) programs. This paper presents the Air Force Approach to training operational flight and ground crews about their responsibilities and the importance of this task which is an integral part of the Aircraft Structural Integrity Program (ASIP) required by AFR-80-13 and MIL-STD-1530A.

1. INTRODUCTION

One of the major reasons of an airplane system affecting its operational readiness is the unforeseen structural fatigue problems encountered throughout the airframe service life. The objective of ASIP is to prevent these problems by diagnosing potential structural failures in the design phase, providing a basis for corrective action, and predicting operational life expectancy of the airframe on individual aircraft. The program consists of five tasks (table 1) which are: a. Task I - Design Information, b. Task II - Design Analysis and Development Tests, c. Task III - Full Scale Testing, d. Task IV - Force Management Data Package, and e. Task V - Force Management. To accomplish task IV the effect of changes in usage on aircraft service life must be assessed. Therefore, the following actions are required:

a. Obtain time history records of parameters necessary to define the actual stress spectra for critical areas of the airframe. These data are processed and analyzed to assess the applicability of the design service life and usage spectra.

b. Develop an individual aircraft tracking program to establish and adjust inspections and repair intervals for each critical area of the airframe based on the individual aircraft usage data. These data are utilized in tracking analysis methods to predict the service life of each individual aircraft.

Dedicated instrumentation and recording equipment must be installed in the fleet aircraft to acquire these data. In the structures community, the recording equipment is typically referred to as the flight loads data recorder. Even though these recorders are designed toward a unique weapon system, most of them require some kind of interface with the flight and ground crew to be able to accomplish its intended function.

Establishing proper methods of training and emphasizing the crew responsibilities is an important task towards the goal of a successful integrity program.

2. DOCUMENTATION

One of the methods the Air Force uses to educate the crews is through suitable documentation. A technical manual/technical order is written for each weapon system to provide the necessary guidance in regards to the responsibilities, objectives, data requirements, and general procedures for collecting and reporting operational usage data in accordance with the ASIP requirements. This manual typically includes but is not limited to the following:

a. Program Description -- Includes a brief description of the program background, objectives, overall tasks associated with ASIP, interaction with other programs, and applicable references.

b. Force Management Operational Programs -- Includes all the programs that utilize the usage data, such as the Individual Aircraft Tracking (IAT) program and the Loads/Environment Spectra Survey (L/ESS) program. Also systems/data bases that support and update these programs should be defined, such as maintenance and status accounting systems.

c. Hardware Description -- Includes a description of the flight loads data recorder, all instrumentation dedicated to the recorder (e.g., strain gages, accelerometers, etc.), support equipment such as data extraction and processing equipment, and all parameters monitored by the recorder, including analysis methods, sampling rates and data sources (table 2).

d. Data Flow -- Includes a description of the various elements of the force management program such as input and data collection, data processing, data analysis,

and force management reporting.

e. **Functional/Management Responsibilities** -- Includes a brief description of the organizations and offices interfacing with the program, and their responsibilities. Includes organizations with overall ASIP responsibilities during and after acquisition of the system, air commands, operating bases, depot maintenance organizations, and associated contractors.

f. **General Procedures** -- Includes the criteria and procedures for extracting/downloading data, data transcription, disposition of data, data retrieval and reporting from maintenance and status accounting systems, and maintenance of ASIP hardware.

3. AUDIO VISUAL

Audio visual presentations are another method used to instruct and prepare the crews towards their responsibilities and proper procedures. Training films have been produced depicting crews performing the step by step tasks associated with the collection and processing of flight loads data. These videos could also be used to describe the following:

- a. Summary of the technical manual/documentation contents
- b. New equipment and procedures
- c. Benefits of new systems
- d. How is the loads data going to be used and how it relates to maintenance

An advantage of creating video presentations is the re-education factor. Since crews are often rotated to perform different maintenance tasks, personnel unfamiliar with their new responsibilities can be brought up to speed relatively easy by showing them the training films.

4. MOTIVATION

Maintaining a positive attitude within the aircrews towards the collection of flight loads data is another important factor that needs to be considered. Maintenance personnel are often discouraged by some of the problems that tend to plague some force management tasks, such as the reliability of the recorders, inoperative sensors/instrumentation, and unavailability of supplies (e.g., blank tapes/disks, forms, etc.)

An approach used by the Air Force to motivate and maintain the crews informed is through feedback. A status report is provided regularly to each operating base on items of importance found during processing of the flight loads recorder data such as data collection statistics, recorders malfunction, and faulty sensors/instrumentation. Figure 1 and table 3 are examples of how these data are presented. The report can also be used to provide information about flight envelope excursions (fig. 2), and requests for maintenance actions on recorders and sensors.

Another technique used to increase crew interest and get them involved is through active participation. Periodic force management meetings are held to discuss topics such as hardware and software status, data collection performance, program modifications, and documentation status and changes. Field personnel are encouraged to attend these meetings in which they can provide valuable feedback in regards to the procedure and potential problem areas. These meetings can help them realize the importance of their work to the purpose and overall objectives of ASIP.

5. RECOMMENDATION

The collection and reporting of operational usage data is one of the most important tasks towards the validation of the aircraft design service life and usage spectra, and prognosticating the remaining structural life of major components of each aircraft. Training methods and motivational techniques should be established and available at the time of operational readiness and should be continuously applied throughout the aircraft service life.

6. REFERENCES

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2. Aircraft Structural Integrity Program Technical Order (T.O. 1B-1B-38), USAF/Rockwell International, 1985
3. B-1B Motivation/Training Videotape, USAF, 1985
4. ASIP Status Reports, Air Force Logistic Center (AFLC/LABR), 1990
5. F-5 ASIP Video Tape, USAF, 1974

USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP) TASKS

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANALYSIS AND DEVELOPMENT PLAN	FULL SCALE TESTING	FORCE MANAGEMENT DATA PACKAGE	FORCE MANAGEMENT
ASIP MASTER PLAN ----- STRUCTURAL DESIGN CRITERIA ----- DAMAGE TOLERANCE & DURABILITY CONTROL PLANS ----- SELECTION OF MAT'L'S, PROCESSES, & JOINING METHODS ----- DESIGN SERVICE LIFE AND DESIGN USAGE	MATERIALS AND JOINT ALLOWABLES ----- LOAD ANALYSIS ----- DESIGN SERVICE LOADS SPECTRA ----- DESIGN CHEMICAL/ THERMAL ENVIRONMENT SPECTRA ----- STRESS ANALYSIS ----- DAMAGE TOLERANCE ANALYSIS ----- DURABILITY ANALYSIS ----- SONIC ANALYSIS ----- VIBRATION ANALYSIS ----- FLUTTER ANALYSIS ----- NUCLEAR WEAPONS EFFECTS ANALYSIS ----- NON-NUCLEAR WEAPONS EFFECTS ANALYSIS ----- DESIGN DEVELOPMENT TESTS	STATIC TESTS ----- DURABILITY TESTS ----- FLIGHT & GROUND OPERATIONS TESTS ----- SONIC TESTS ----- FLIGHT VIBRATION TESTS ----- FLUTTER TESTS ----- EVALUATION OF TESTS RESULTS	FINAL ANALYSIS ----- STRENGTH SUMMARY ----- FORCE STRUCTURAL MAINTENANCE PLAN ----- LOADS/ ENVIRONMENT SPECTRA SURVEY ----- INDIVIDUAL AIRPLANE TRACKING PROGRAM	LOADS/ ENVIRONMENT SPECTRA SURVEY ----- INDIVIDUAL AIRPLANE TRACKING DATA ----- INDIVIDUAL AIRPLANE MAINTENANCE TIMES ----- STRUCTURAL MAINTENANCE RECORDS

TABLE 1

PARAMETER MODES

PARAMETER	MODE		ANALYSIS METHOD	SAMPLE RATE/SEC	DATA SOURCE
	LESS	IAT			
GROSS WEIGHT	X	X	TH	2	AC1
CENTER OF GRAVITY	X		TH	2	AC1
ALTITUDE	X	X	TH	2	AC1
AIR SPEED	X		TH	2	AC1
MACH NUMBER	X	X	TH	2	AC1
PITCH RATE	X		PV	10	AC1
ROLL RATE	X		PV	10	AC1
YAW RATE	X		PV	10	AC1
PITCH ACCELERATION	X	X	PV	10	CALC
ROLL ACCELERATION	X		PV	10	CALC
YAW ACCELERATION	X		PV	10	CALC
LEFT HORIZONTAL STAB. POSITION	X		TH	25	AC2
RIGHT HORIZONTAL STAB. POSITION	X		TH	25	AC2
LEFT INBOARD SPOILER POSITION	X		TH	25	AC2
RIGHT INBOARD SPOILER POSITION	X		TH	25	AC2
RUDDER POSITION	X		TH	25	AC2
LEFT HORIZONTAL STAB. STRESS	X	X	PV	30	AC2
RIGHT HORIZONTAL STAB. STRESS	X	X	PV	30	AC2
WING STATION 321 STRESS	X	X	PV	30	AC2
WING STATION 123 STRESS	X	X	PV	30	AC2
FUSELAGE STATION 234 STRESS	X	X	PV	30	AC2
FUSELAGE STATION 345 STRESS	X	X	PV	30	AC2
WEIGHT ON WHEELS	X	X	DSCR	1	AC1
AIR REFUEL DOOR	X	X	DSCR	1	AC1

AC1 AIRCRAFT COMPUTER 1
 AC2 AIRCRAFT COMPUTER 2
 CALC INFLIGHT CALCULATED
 DSCR DISCRETE
 PV PEAK VALLEY COMPRESSION
 TH TIME HISTORY COMPRESSION

TABLE 2

SENSOR STATUS

TAIL NO.	BASE	ACCELEROMETERS			STRAIN GAGES				
		Nz	Ny	Nx	D1	D2	D3	D4	D5
1	A	*	*	*	*	*	*	*	*
2	A	*	*	*	*	*	*	*	*
3	A	*	*	*	*	*	*	*	*
4	A								
5	A	*	*	*	+	*	*	*	*
6	A								
7	A	*	*	*	*	*	*	-	*
8	B	*	*	*	*	B	*	*	*
9	B	*	*	*	*	*	*	*	+
10	B	*	*	*	*	*	*	*	-
11	B	*	*	*	*	*	*	*	*
12	B								
13	B								
14	B								
15	C	*	*	*	*	*	*	*	*
16	C	*	*	*	*	*	B	*	*
17	C								
18	C								
19	C								
20	C	*	*	*	*	-	*	*	*
21	C								

* sensors operating normally
 B inoperative sensor
 + sensor biased, high
 - sensor biased, low

TABLE 3

USABLE DATA RETURN - FLIGHT HOURS BASE X

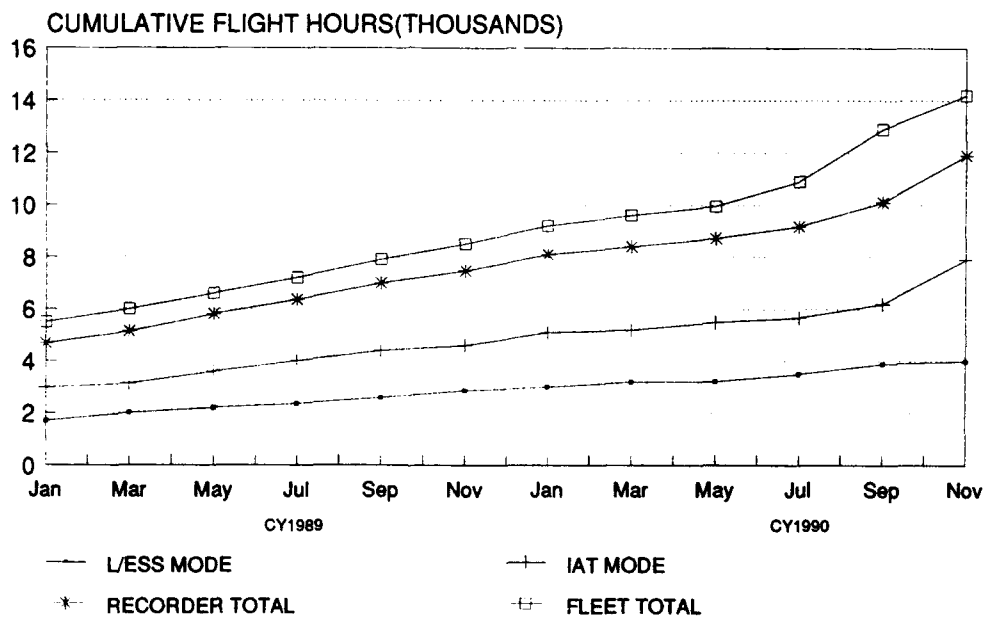
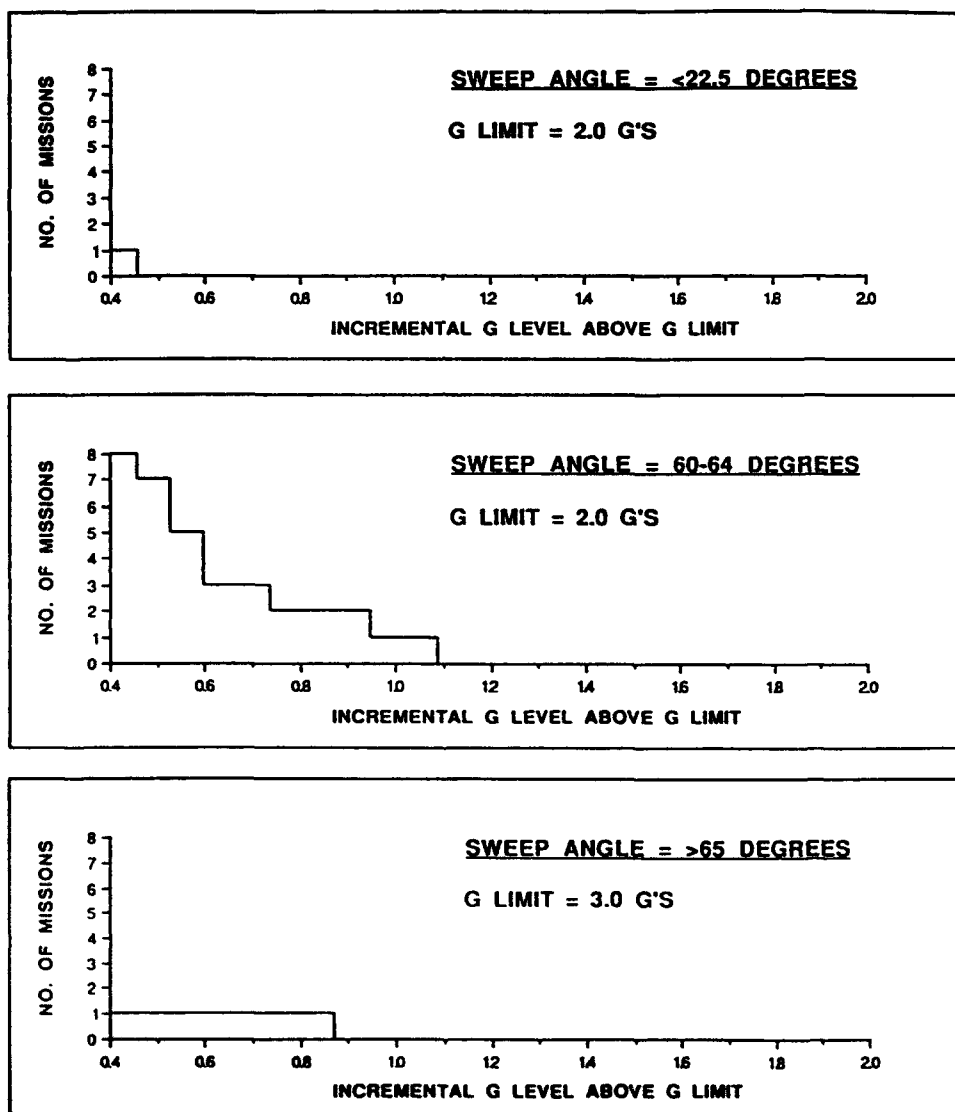


FIGURE 1

SIGNIFICANT EVENTS IN EXCESS OF G LIMIT**FIGURE 2**

ROUND TABLE DISCUSSION - ISSUES AND RECOMMENDATIONS

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1. INTRODUCTION

Following the presentation of the formal papers there followed an active discussion period devoted to Issues and Recommendations. A summary of the discussions is presented in the following.

2. ACKNOWLEDGEMENTS

The Round Table discussion period was led by the author with assistance from the five Session Chairmen, Mr P D Adams (UK), Mr J B de Jonge (NL), Mr C L Petrin Jr (US), Prof. A Salvetti (IT) and Mr J Thompson (CA). Contributors from the audience, were Mr A G Aponte (US), Mr M A Bullock (UK), Mr S R Hall (CA), Dr C E Harris (US), Mrs D M Holford (UK), Dr L E Jarfall (SW), Dipl Ing V Ladda (GE), Dr J W Lincoln (US), Dipl Ing R Neunaber (GE), Dr M K Nygard (NO), Mr J G Reichel (US), Sqn Ldr M E J Render (UK), Dr Ing W Schulz (GE), Mr D L Simpson (CA), Ir D J Spiekhout (NL) and Major M B Zgela (CA).

3. STRAIN GAUGES

3.1 Aircraft-to-aircraft repeatability

It is possible for there to be different responses per unit load for strain gauges located at identical locations on different aircraft in the fleet. An example was given of outputs from strain gauges attached to each face of a simple rectangular section pin-jointed link that had been used on an airframe fatigue test. Differences in strain response were of the order of 10%, this equating to about 30% on life.

The general opinion was that a form of check calibration was necessary and that special care was required when locating the gauges. The need to

locate gauges away from joints was noted.

One study had shown measured strains in a theoretically uniform stress field to be 60% greater than predictions, with very large variations in stress gradient. Such studies reinforced the need to ensure that gauges must be precisely located (to within 1mm in that particular area).

There was extensive discussion on "calibration" procedures, all of which related to "in-flight" calibration.

Techniques referred to were :

(i) to monitor gauges and parameters and to establish a relationship using regression techniques. If there were significant deviations (ie. with low correlation) then there may be calibration problems and further studies were then necessary to establish software changes to account for these potential variations. This technique was not always successful.

(ii) to perform specific manoeuvres on each aircraft to provide a calibration. To a question about the repeatability of such manoeuvres, one response was to suggest using the whole flight and then perform a regression, as in (i) above. However another view was that such an approach may lead to too many complexities and that specific manoeuvres should be chosen to reduce the quantity of data.

(iii) to examine all strain-gauge responses from high percentage fleet fits and identify which gauges are biased high, and which appear to be low. From such an examination correction factors could be established.

3.2 Change of monitoring location

Two problems were discussed. The first was that associated with physically locating the new gauges and checking their repeatability. This task was similar to that necessary at new build, and there may be need for new parametric work to establish in-flight "calibrations".

The second related to the need to assess the life already used at such a location. The complexity of this task was dependent on the type of information already stored for other areas (processed fatigue consumption data, or raw data for example) and hence changes in monitoring locations should be borne in mind during the design of the system.

3.3 Long term reliability

A number of papers commented on reliability and maintainability of strain gauge systems, however it was difficult to establish statistics.

A false, bad, impression could arise because of poor application of gauges. It was essential to have a very carefully controlled process which could be aided by making use of a jig both to locate the gauge and to apply a controlled pressure during curing. There should be a controlled temperature for the mix of the adhesives and there should be very precise instructions for operators who were to embody the gauges.

In general users were happy with strain gauge systems, with some reporting over 10 years in service with few problems. A typical comment was that if there was a problem then it was obvious and action could be taken to correct it.

Like any sensor strain gauge systems required maintenance. Typical problems, particularly where there had been retrospective embodiment, related to wiring being damaged, problems with power supplies and water ingress to conditioning boxes.

For new designs such problems should

be reduced significantly by good instrumentation and installation design. Such systems should not be installed in areas where there was high maintenance activity.

The Royal Aerospace Establishment at Farnborough offered to produce a report covering experiences of strain gauge systems in use on UK aircraft.

3.4 Corrosion

To a question about the possibility of corrosion occurring at the location of the strain gauge one commentator referred to such problems on an undercarriage loads measurement programme. However the general opinion was that correct protection after embodiment of the strain gauge should overcome this problem.

4. PARAMETERS

Parametric systems can be classified as:

(i) systems to determine loads by interrogating on-board parameters using predictor equations set up from deterministic models.

(ii) systems set up to give loads predictions from parametric equations that have been derived by regression analysis using strain gauge data.

Some concern was expressed about the potential complexity of parametric systems for areas such as the rear fuselage of modern combat aircraft with active control. It was suggested that in such cases a strain gauge approach was possibly more viable.

There was also concern about statistical parametric equations set up from regression analyses if they did not respect the fundamental physical process and if they contained functions of a number of dependent parameters. A warning was given about the dangers associated with setting up equations from a limited number of manoeuvres, the equations potentially not being sufficiently robust to cater for new types of manoeuvres in different parts of the flight envelope.

It was stated that the key element in setting up parametric systems was to have a good flight load survey at the outset. If buffet was present then this should be addressed separately as a parametric systems could not hope to cater for such a phenomenon directly. An understanding of when it occurred could be obtained from the flight load programme and action could then be taken to account for it separately.

Should new monitoring locations be necessary there were conflicting views on potential problems. One commentator believed that a strain gauge should be installed at the new location and that new algorithms should then be determined using regression techniques. Another thought that there would be a lot of work as the original development aircraft might not be available. New strain gauging might be necessary to support the development of new algorithms. The same commentator reflected on the problems of analysing past flying, referred to in 3.2 above. Another mentioned the large amount of data that might be stored on the ground, but a fourth supported the need to retain the data, it being easier to reanalyse this than repeat a measurement programme.

Finally there was a comment that if good accuracy could be demonstrated from parametric systems these should be used rather than strain-gauge systems as there would be less of a maintenance problem.

5. GENERAL

5.1 Methods

Both strain-gauge and parametric systems have potential advantages and disadvantages. There is no "right" system to use; the particular problem should be considered and a selection made (strain gauges, parameters, or a combination of both).

5.2 Monitoring port and starboard

Following the observation that there were some who thought it was necessary to monitor only one side of the aircraft, because of symmetry, a

number of observers noted that they had seen differences between port and starboard results. On larger aircraft these differences could be due simply to the fact that the pilot seating was offset and that circuits were usually performed in one direction.

One commentator believed the differences were small and that there might be a stronger need to monitor other locations, bearing in mind the limited number of available channels.

5.3 Loads/stress/damage

Many systems were set up to generate outputs of fatigue damage, life consumed or fatigue index. One questioner asked whether it would be better simply to measure load spectra at various locations on the airframe and to compare these with design spectra; provided the cumulative usage spectrum was less severe than design then there were no problems. This approach was suggested because of possible difficulties in arriving at accurate damage assessments (see 6). To a question about dealing with significant differences in spectrum shape it was suggested that more tests might be necessary.

5.4 Quality

There was a strong plea that there should be a total quality management system in place for the whole process of data collection, processing and delivery of management information to the user. For every step of the procedure there should be a process to ensure data quality and integrity.

5.5 Sample Rates

It was noted that pure sine wave peaks can be detected using sample rates that were only a few times higher than the frequency. For a complex load, however, with superimposed minor loads a sample rate of at least 20 times the bandwidth was necessary to detect peaks with 1% accuracy.

6. LIFE CALCULATION

During both the formal presentations and the discussion period the life (damage) calculation process was

raised as a problem area. One view was that "Miner" calculations were usually wrong but conservative. However when used for life tracking in a relative sense to compare load spectra then serious discrepancies could result. Another view was that Miner calculations were always unconservative and that, if the design spectrum was less severe than the service spectrum, there would be problems. On the other hand if it was more severe there would be no problem.

A further comment related to comparisons of spectrum severity using cycle-by-cycle local stress-strain approaches and, alternatively, simple Miner type calculations from exceedance data. Totally reversed relative severities of spectra could result.

It was important to build up confidence in using a particular method. Reliance should not be placed on a single method until this was well proven. It should be possible to go to relatively simple basic procedures and obtain the same trends as those being generated by more complex methods.

The main weak link in the whole fatigue monitoring process was the life calculation procedure. There was general agreement that there was a need to devote time to debate this matter, perhaps through an AGARD Workshop. Such a Workshop should review available procedures for estimating damage, fatigue life, crack growth or durability whether applied to Safe Life or Damage Tolerant structures. The need to retain historical data rather than to use processed (rainflow) time histories should also be covered.

7. ADVICE TO OPERATORS

There were a number of questions that had not been addressed in the formal papers. From the enormous quantity of data generated by fatigue monitoring systems it was essential that useful information was extracted and presented as advice to the operator in a simple form that enabled him to realise his fatigue budget and management requirements.

Designers of fatigue monitoring and management systems must give serious consideration, at the planning stage, to the way in which results are to be presented. Apart from knowing the life consumed, operators require information on ways in which consumption rates can be reduced (change of configuration in specific roles, change of operation). Serious consideration should be given to the interface problem - simple graphical presentations were likely to be more useful than large tabulations of data, for example.

For systems that were embodied on only a limited number of aircraft in the fleet then it was necessary to ensure these aircraft were allocated in a way that would give a representative cross-section of the fleet usage.

In providing advice to operators it would also be helpful if indications could be given of the reliability of predictions.

8. PROBABILISTIC PROCEDURES

It was noted that a number of industries with large, expensive and critical structures used probabilistic approaches in their design and clearance procedures. Such topics had also been covered during the meeting.

Procedures could be set-up to include a crack growth model, randomisation of load spectra, uncertainties in inspection methods, material properties, load, stress analysis, failure criteria, stress intensity factors and so on. In addition cost functions for inspection and repair might also be included.

Such procedures enable the risk of failure to be quantified. They also allow an assessment to be made of the importance of a particular variable. In carrying out sensitivity studies it is not necessary to be highly accurate when quantifying uncertainties of individual variables in order to see how a particular variable influences the overall safety. Some parameters may be very uncertain yet have little influence on the overall risk. If cost functions are included then it is also possible to optimise inspection

plans that satisfy both safety and economic requirements.

Procedures may also permit the user to update predictions based on the results of inspections and there should be a long term aim to use these methods at the design stage to optimise life cycle costs.

Despite the developments and aims referred to above it was noted that such methods are intended to be used not as an alternative to deterministic methods but as an addition.

One commentator confirmed the belief that such procedures had their place, but offered a word of caution with regard to the difficulty of accounting for rogue flaws in crack distributions, and in allowing for in-service damage. His advice to operators had always been that safety was eroded to some extent when probabilistic procedures were involved.

They were not yet ready for use as a design tool and the benefit from the reduction in the cost of inspections was not a big enough driver over the use of deterministic methods.

It was noted that, for the study of multi-site damage problems, statistical approaches were essential.

9. RELIANCE ON INSPECTION

In following inspection procedures, for a given probability of detection for a particular technique, there were other risks to be considered. These related to the need to be sure that the right piece of structure was being examined and that the right frequency had been chosen, the latter being dependent on the fatigue monitoring system generating the correct information.

A comment was made that, for typical military aircraft structures, whenever inspection procedures are relied on to maintain integrity then safety is eroded to some degree. The USAF damage tolerant approach had now been in existence for 15 years and, for structures designed to carry initial flaws for two lifetimes without

inspection, there had never been a failure in the first lifetime. There was a desire not to require inspections during the aircraft lifetime and, from experience to date, there was a satisfactory degree of confidence in the damage tolerant approach.

To a request to see if all present believed that inspection needs had been satisfied and that there was no further need for R & D in this area there was no response!

10. ESTIMATE OF USAGE

Some discussion took place on the fact that invariably actual usage was quite different from (and usually more severe than) design usage, and that most operators wanted to keep their aircraft longer than originally envisaged. It was also pointed out that there were potentially major problems as structures were more fully optimised at the design stage and were therefore less tolerant to change. Usage was also much more complex with state-of-the-art developments and the use of active controls.

To minimise potential problems it was essential that there was a good, continuing relationship between designers, qualification organisations, operators and pilots.

One observer was struck by the large amounts of money spent on monitoring usage of military aircraft when there were no such activities in the civil field. His experience was that, for transport aircraft, the assumed spectrum was usually more severe than the actual usage. Military people should be more realistic in the first place, using a severe spectrum, and accept the small mass penalty but save cost in the long term.

This latter approach received support from another commentator but he also pointed out that, in the early stages of a project, designers were often driven by aircraft performance to satisfy customer demands.

11. RECOMMENDATIONS

In general any recommendations resulting from the discussions are included under the appropriate section heading in the text. There is however one important recommendation for a follow-on AGARD activity.

In Section 6 it was highlighted that no matter what monitoring system was used results were critically dependent upon the life calculation process. It is recommended that the AGARD SMP should organise a Workshop to review available procedures for estimating life consumed for both Safe Life and Damage Tolerant design philosophies. The link to "design" and "test" must be included, as must cycle identification and counting techniques.

REPORT DOCUMENTATION PAGE									
1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document						
	AGARD-CP-506	ISBN 92-835-0642-1	UNCLASSIFIED						
5. Originator	Advisory Group for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France								
6. Title	FATIGUE MANAGEMENT								
7. Presented at	the 72nd Meeting of the AGARD Structures and Materials Panel, held in Bath, United Kingdom 29th April to 3rd May 1991.								
8. Author(s)/Editor(s)	Various		9. Date December 1991						
10. Author's/Editor's Address	Various		11. Pages 278						
12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the back covers of all AGARD publications.								
13. Keywords/Descriptors									
<table border="0"> <tr> <td>Aircraft</td> <td>Monitors</td> </tr> <tr> <td>Airframes</td> <td>Structures</td> </tr> <tr> <td>Fatigue — Materials</td> <td>Maintenance management</td> </tr> </table>				Aircraft	Monitors	Airframes	Structures	Fatigue — Materials	Maintenance management
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