RTO-MP-7
AC/323(AVT)TP/4

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION

BP 25, 7 RUE ANCELLE, F-92201 NEUILLY-SUR-SEINE CEDEX, FRANCE

RTO MEETING PROCEEDINGS 7

Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs

(Exploitation des données relatives aux efforts structuraux et à l'intégrité des structures en vue de la diminution des coûts globaux de possession)

Papers presented at the Specialists' Meeting of the RTO Applied Vehicle Technology Panel (AVT) (organised by the former AGARD Structures and Materials Panel) held in Brussels, Belgium, 11-12 May 1998.

The Research and Technology Organization (RTO) of NATO

RTO is the single focus in NATO for Defence Research and Technology activities. Its mission is to conduct and promote cooperative research and information exchange. The objective is to support the development and effective use of national defence research and technology and to meet the military needs of the Alliance, to maintain a technological lead, and to provide advice to NATO and national decision makers. The RTO performs its mission with the support of an extensive network of national experts. It also ensures effective coordination with other NATO bodies involved in R&T activities.

RTO reports both to the Military Committee of NATO and to the Conference of National Armament Directors. It comprises a Research and Technology Board (RTB) as the highest level of national representation and the Research and Technology Agency (RTA), a dedicated staff with its headquarters in Neuilly, near Paris, France. In order to facilitate contacts with the military users and other NATO activities, a small part of the RTA staff is located in NATO Headquarters in Brussels. The Brussels staff also coordinates RTO's cooperation with nations in Middle and Eastern Europe, to which RTO attaches particular importance especially as working together in the field of research is one of the more promising areas of initial cooperation.

The total spectrum of R&T activities is covered by 6 Panels, dealing with:

- SAS Studies, Analysis and Simulation
- SCI Systems Concepts and Integration
- SET Sensors and Electronics Technology
- IST Information Systems Technology
- AVT Applied Vehicle Technology
- HFM Human Factors and Medicine

These Panels are made up of national representatives as well as generally recognised 'world class' scientists. The Panels also provide a communication link to military users and other NATO bodies. RTO's scientific and technological work is carried out by Technical Teams, created for specific activities and with a specific duration. Such Technical Teams can organise workshops, symposia, field trials, lecture series and training courses. An important function of these Technical Teams is to ensure the continuity of the expert networks.

RTO builds upon earlier cooperation in defence research and technology as set-up under the Advisory Group for Aerospace Research and Development (AGARD) and the Defence Research Group (DRG). AGARD and the DRG share common roots in that they were both established at the initiative of Dr Theodore von Kármán, a leading aerospace scientist, who early on recognised the importance of scientific support for the Allied Armed Forces. RTO is capitalising on these common roots in order to provide the Alliance and the NATO nations with a strong scientific and technological basis that will guarantee a solid base for the future.

The content of this publication has been reproduced
directly from material supplied by RTO or the authors.

Printed on recycled paper

Published November 1998

ISBN 92-837-1005-3

Printed by Canada Communication Group Inc.
(A St. Joseph Corporation Company)
45 Sacré-Cœur Blvd., Hull (Québec), Canada K1A 0S7

Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs

(RTO MP-7)

Executive Summary

With the increasing sophistication of monitoring systems, the operators of current and next generation military aircraft have the means available to understand the actual loading environment and the on-going health of their weapons platforms. This Applied Vehicle Technology Panel Specialists' Meeting brought together the designers and operators of monitoring systems for fixed-wing aircraft, helicopters and engines to review the potential benefits from the exploitation of existing and emerging systems.

The meeting covered a wide range of tracking and monitoring techniques, ranging from the complex Eurofighter Structural Health Monitoring System to the simple event counting technique for 3 NATO 707s used as AWACS Trainer Cargo Aircraft. The papers highlighted the reliance of most fixed-wing fleets on direct monitoring methods whereas helicopters and engines rely on more indirect methods. One emerging theme was the need for the integration of the knowledge derived from monitoring systems with maintenance activities so that the benefits could be exploited at an earlier stage, rather than wait for life extension programmes.

Once the designers of weapon systems also include fully integrated monitoring systems (for fatigue, health and usage), it was felt that the design safety factors, used in part to account for usage variability, should be reviewed. Any reduction in such factors would reduce life cycle costs by extending lives without undermining current airworthiness standards.

Exploitation des données relatives aux efforts structuraux et à l'intégrité des structures en vue de la diminution des coûts globaux de possession

(RTO MP-7)

Synthèse

La sophistication grandissante des systèmes de surveillance permet aux exploitants de la génération actuelle ainsi qu'à ceux de la prochaine génération d'avions militaires de connaître l'état de fonctionnement et le chargement réel de leurs plates-formes d'armes. Cette réunion de spécialistes organisée par la commission technologies appliquées aux véhicules a rassemblé concepteurs et exploitants de systèmes de surveillance pour aéronefs à voilure fixe, hélicoptères et propulseurs pour un examen des avantages pouvant découler de l'exploitation des systèmes existants et à venir.

Le programme de la réunion a couvert un large éventail de techniques de surveillance et de localisation de pannes, allant du système complexe EUROFIGHTER de contrôle de l'état de la structure, à la technique de simple comptage des évènements utilisée pour les 3 BOEING 707 de l'OTAN, utilisés comme avions cargo pour entraînement AWACS. Les communications présentées ont mis en relief la dépendance de la majorité des flottes d'avions à voilure fixe vis à vis des méthodes de surveillance directe, comparée aux hélicoptères et aux propulseurs pour lesquels il est plutôt fait appel aux méthodes indirectes. La réunion a souligné l'importance de l'intégration des connaissances dérivées des systèmes de surveillance dans les activités de maintenance de façon à pouvoir les exploiter plus tôt et ne pas avoir à attendre les programmes de prolongation du cycle de vie des appareils.

La réunion a conclu que dès que les concepteurs de systèmes d'armes mettent en place des systèmes de surveillance intégrés (pour le contrôle de la fatigue, de l'état de fonctionnement et de l'aptitude de l'appareil vis à vis de la mission), il semble qu'il faille réexaminer les coefficients de sécurité qui couvrent actuellement une très large gamme d'opérations. La réduction de ces coefficients aurait pour effet de diminuer les coûts globaux de possession, car le cycle de vie des appareils serait prolongé sans compromettre les normes de navigabilité.

Contents

Preface

With the increasing sophistication of monitoring systems, the operators of current and next generation military aircraft have the means available to understand the actual loading environment and the on-going health and usage of their weapons platforms. The objective of this Specialists' Meeting was to examine the state of the art in loads measurement and health monitoring systems to see how these were contributing to the management of life cycle costs of both fixed-wing aircraft and helicopters fleets.

The Specialists' Meeting took place in Brussels, Belgium in May 1998, and 19 papers were presented from industry, academia, military operators, and research institutions, covering airframe and engine monitoring systems. This broad range of topics provided an excellent forum for the exchange of ideas between practitioners working with current systems and designers preparing future systems. Therefore, the meeting fulfilled its objective of providing an open exchange of scientific and technical information between all the NATO countries.

S. WELBURN
Wg Cdr, RAF
Chairman of Sub-Committee

Sub-Committee Members

Chairman

Wing Commander S. WELBURN
HQ Logistics Command
ADSM 17
Room, W113, Swales Block
RAF Wynton
HUNTINGTON, Cambs PE17 2DL

D. Chaumette	-	FR
J. Coyle	-	US
H. Goncalo	-	PO
G. Günther	-	GE
P. Heuler	-	GE
A. Humble	-	UK
G.N. Livings	-	UK
H.H. Ottens	-	NE
D. Paul	-	US
C. Perron	-	CA
A. Salvetti	-	IT
S. Sampath	-	US
O. Sensburg	-	GE
D. Simpson	-	CA
J. Waldman	-	US

Panel Executive

Dr. J.M. CARBALLAL, SP

Mail from Europe:

RTA-OTAN/AVT
BP 25
7, rue Ancelle
F-92201 Neuilly-sur-Seine Cedex
France

Mail from US and Canada:

RTA-NATO/AVT
PSC 116
APO AE 09777

Tel. 33 (0)1 55 61 22 90 & 92
Telefax 33 (0)1 55 61 22 99 & 98

TECHNICAL EVALUATION REPORT ON 1998 SPECIALISTS' MEETING ON EXPLOITATION OF STRUCUTURAL LOADS/HEALTH DATA FOR REDUCED LIFE CYCLE COSTS

Held on 11-12 May 1998 in Brussels, Belgium

J. D. Cronkhite, Bell Helicopter Textron Inc.
PO Box 482
Fort Worth, Texas 76101 USA

L. Gill, British Aerospace
Military Aircraft & Aerostructures Warton Aerodrome
Preston, Lancashire PR4 1AX

SUMMARY

With the increasing sophistication of monitoring systems, the operators of current and next generation military aircraft have the means available to understand the actual loading environment and on-going health of their weapons platforms. This Specialists' Meeting was designed to bring together the designers and operators of monitoring systems for airframes and engines to review techniques for monitoring structural life consumption for both safelife and damage tolerance fatigue methodologies and to highlight the potential benefits that might be gained from the exploitation of the information derived from modern and future systems in terms of improved airworthiness and reduced maintenance costs.

The meeting, which covered aspects of both fixed and rotary wing aircraft, included contributions from aerospace designers, research establishments and the military staff. In particular, the meeting concentrated on the collection, analysis and use of loads/health data by the military for fleet maintenance and logistics planning. Experiences with monitoring systems and techniques for automated analysis and data gathering were described by the authors from six NATO nations that provided valuable insight into how those systems can contribute to reducing the life cycle costs of military aircraft. In this report, summaries of each of the 19 papers presented at the meeting are followed by conclusions and comments drawn from the presentations and the roundtable discussion at the end of the meeting.

TECHNICAL ABBREVIATIONS

ASIP = Aircraft Structural Integrity Program
ENSIP = Engine Structural Integrity Program
HCF = High Cycle Fatigue
HUMS = Health and Usage Monitoring Systems
IAT = Individual Aircraft Tracking
LCF = Low Cycle Fatigue
OLM = Operational Loads Monitoring
SAT = Selected Aircraft Tracking
SHM = Structural Health Monitoring

INTRODUCTION

For many years, fixed wing military aircraft have been monitoring airframe fatigue consumption using either simple g-meters or, more recently, strain gage systems. Engine health monitoring is also becoming increasingly sophisticated. In addition, technology improvements are bringing comprehensive health and usage monitoring systems (HUMS) to helicopters in both military and civilian fleets. Consequently, there is an increasing awareness in the aerospace community of the potential benefits that can be derived from monitoring to achieve an optimum balance between, cost, safety and availability.

Operational Load Measurement (OLM) programs are currently used periodically on UK military fixed-wing aircraft to capture in-flight parameters and strain data from critical locations on a sample of the airframes in a particular fleet. The captured data is then used to amend fatigue meter formula predictions and to check that representative loads are being applied to full-scale airframe fatigue tests. However, due to significant advances in airborne data capture and storage devices the next generation of European military combat aircraft, such as EF2000, will be permanently fitted with individual aircraft tracking (IAT) structural health monitoring (SHM) systems. With the capital investment in new aircraft fleets usually amounting to many billions of dollars, the maximum utilization of component lives is a key objective for military operators in NATO. Furthermore, UK studies have shown that the cost of each 1% of fleet fatigue life can be in excess of £100 million. Structural health monitoring of individual airframes could play a significant part in the improved utilization of fatigue lives and thus lead to very significant savings for NATO airforces, particularly if fleet lives could be extended.

In the US and Canada, the current generation of combat aircraft already have onboard monitoring systems but there is a real concern that, despite the technological advances, the great volume of usage data exceeds the means to fully exploit the information available from these systems. Although onboard computing devices now offer the means to have processed strain data and fatigue usage available at the end of every sortie, the subsequent use of this data requires careful consideration for planning fleet maintenance and modification programs. Thus the need for automation in monitoring analysis becomes vital.

Although HUMS are being embodied into new aircraft, existing fixed and rotary wing fleets are prime candidates for retrospective system fit that could be exploited to extend their service lives. However, investment appraisals are required to assess the benefits that a HUMS modification could bring.

In his opening remarks, the conference chairman, Wing Commander Stephen Welburn (UK-RAF), stated that the presentations at the two-day meeting would cover a wide range of monitoring systems and analysis techniques applied to fixed wing aircraft, helicopters and engines. His hope was that the presenters would discuss how these systems and techniques were being used and could be exploited to the benefit of the NATO airforces.

SUMMARIES OF PAPERS PRESENTED

1. "Requirements on Future Structural Health Monitoring Systems"
   Presenter: Mr. M. Neumair, IABG , GE

This presentation summarized the systems used for the F-4F Phantom and the Tornado aircraft. The Tornado system, referred to as OLMOS (Onboard Life Monitoring System), is more comprehensive than the older F-4F system. The monitoring approach for both aircraft is to use simplified Individual Aircraft Tracking (IAT) and more complex selected aircraft tracking (SAT). Due to the cost of maintenance it is important to get the full service life available for aircraft components and there is a need to monitor fatigue consumption on individual aircraft at each of the identified fatigue critical areas (FCA). The point was made that actual service usage is often different than the aircraft design assumptions and even if not, life extensions are often required. For the future, a combination of limited onboard data processing and more comprehensive ground based processing and analysis is recommended. Also, the use of parameter-based (indirect) monitoring is recommended, but with a limited fit of strain gages (direct monitoring) on a few aircraft for checking and read-across purposes.

2. "Future Fatigue Monitoring Systems"
   Presenter: Sqn Ldr. S. R. Armitage, RAF, UK

This presentation was from an end user viewpoint and summarized the existing RAF systems for usage and fatigue monitoring. Most aircraft are safe life designs subject to full-scale fatigue test. Although current monitoring techniques are adequate, there is room for improvement. In-service usage is being monitored by fatigue meters and some limited flight parameters are recorded on hard copy flying log and fatigue data sheets filled in by the captain and/or tradesman. Simple fatigue meters have been used in the RAF since the 1950's and the shortcomings were discussed (e.g., center fuselage and inner wing monitoring only, low frequency effects only, no buffet taken into account). For emerging systems when considering direct (strain gages) versus indirect (parameters), the UK RAF prefers direct with some parameters. Also, when considering airborne calculation versus ground based, the preference is for ground based for the main fatigue calculations, but exceedance and system malfunction data is required immediately post flight. Storing the time history data on the ground based system also enables retrospective analysis work to be undertaken.

The current Harrier II Fatigue Monitoring and Computing System (FMCS) uses a fleet sample (SAT approach) for operational loads monitoring (OLM). The Eurofighter 2000's structural Health Monitoring (SHM) system uses a fleetwide (IAT approach) OLM fit. Both of these systems have onboard processing and are down loadable to ground stations but cannot provide time histories as a matter of routine. For the future, more commonality of equipment is needed and the RAF intends to use VC10, Jaguar and Tucano aircraft as demonstrators. Of paramount importance is the ability to produce readily usable data to the agencies that need it.

For Helicopters the current policy is to make worst case assumptions and not rely on fatigue monitoring. HUMS is being fitted to the helicopter fleet and there is the possible use of neural networks for fatigue monitoring in the future but this will require training using strain gage data.

3. "Technical Data Management – An Essential Tool for the Effective Life Cycle Management"
   Presenter: Mr. Steve R. Hall, Celeris Aerospace, CA

This presentation focused on data management considerations for an OLM program. Effective data management is often neglected at the outset of an OLM program. Large amounts of data are collected during an OLM program with emphasis on the engineering aspects (e.g. OLM recorder, sensors, and system integration) but data management is typically poorly addressed. The incremental costs of adding additional channels/sensors may be small at the onset of a program, but the impact on analysis and acquisition times can be significant. Therefore, do not simply add extra channels etc., just because it's easy without considering the implications on data management. A total Data Integrity Initiative (TDI^2) was referred to as a systematic approach to provide cost effective Life Cycle Management of the aircraft fleet. Key elements of effective data management are: (1) define the objectives and let the requirements dictate the software, (2) detect errors in the data and fix as close to the source as possible, (3) the more data you have the more robust the structure of the system needs to be, (4) use the internet, it can be secure, and (5) a disciplined approach is essential to Technical data management. Celeris plans to have central locations for OLM programs on the internet that is to be launched in the Fall 1998.

4. "CF-188 Fatigue Life Management Program"
   Presenter: Capt. Y. Caron, DND, CA

This presentation discussed the experiences of the Canadian CF-188 Fatigue Life Management Program (FLMP). The FLMP is an onboard monitoring system with OLM and IAT. In the early days of the CF-188, there were very high fatigue consumption rates and an FLMP was implemented to maximize the fatigue life and maintenance periods and enable service life extensions to occur. The information obtained from the OLM has enabled the fatigue consumption rates to be reduced. The accuracy of the system is critical. Among the problems encountered were how to use uncalibrated strain gage data (in flight cals were used which might be questionable). Eighteen important lessons learned were summarized at the end paper. Examples included: (a) consider parameter-based loads tracking in lieu of a strain-based system, (b) plan for calibrations of strain sensors, and (c) provide continued feedback of fatigue tracking data to the operational community.

5. "Recommended Practices for Monitoring Gas Turbine Engine Life Consumption"
   Presenter: Mr. G. Harrison, DERA, UK

This presentation summarized the findings of the PEP-WG28 engine monitoring working group that examined methodologies used to predict engine life and life consumption rates. Damage mechanisms that were addressed included low/high cycle fatigue and thermo-mechanical fatigue and both civil and military practices were considered. It was noted that ENSIP classifies components as either safety, mission or durability critical and that safety cannot be compromised. The group is investigating various types of usage monitoring equipment for either onboard or ground based analysis. If onboard processing was used then a data reduction algorithm would be required, which would need to be checked for accuracy. It was also noted that for onboard processing historical data would not be available. Monitoring on an individual aircraft basis can lead to extended life of components. To ensure +/- 10% accuracy, monitoring approximately 85% of the engines was required. It was noted that a safe life approach generally offers lower maintenance and is more attractive for smaller fleets. A damage tolerance approach gives improved airworthiness and offers more use of available life.

6. "Methods of Modern Lifing Concepts Implemented in Onboard Life Usage Monitoring Systems"
   Presenter: Dr. M. Kohl, MTU, GE.

In this presentation, the author discussed two lifing concepts used for engine applications - safe crack initiation (0.4mm crack) and safe crack propagation (fracture mechanics approach). The latter approach is being used more and more to extend the life of components beyond the life predicted by the former method. It is essential that all engines involved are individually monitored. As an example, a turbine disc showed an increase in life of 40%
by using algorithms developed to monitor life consumption in the crack propagation regime.

7. "Contribution of HUMS to Calculation of Damage State and Future Life of Helicopter Components under Safe Life and Damage Tolerant Designs"
   Presenter: Prof. P. Irving, Cranfield Univ., UK

This presentation looked at the potential of using loads monitoring techniques for service life extensions on helicopter components. Currently flight hours are used to life components. Measured loads data from flights of a Westand Lynx helicopter were used to calculate fatigue damage and showed a wide variation in fatigue damage between different flight regimes. The conservatism used by the manufacturers in the use of a 1.2 factor applied to the measured loads to account for scatter and the conservative assumptions applied to the S-N fatigue data was questioned. The presenter felt that there was a potential for eliminating, or at least reducing, the 1.2 factor using loads monitoring.

8. "SH-60Helicopter Integrated Diagnostic System (HIDS) Program Experience and Results of Seeded Fault Testing"
Presenter: A. Hess, NAWC, USA

This presentation discussed the results of an extensive program by the U.S. Navy to develop a library of vibration data (32 channels at 100 kHz) for good and seeded fault components using an SH-60 helicopter testbed. In addition, steady state data from the power plant and airframe is also being recorded. The results of the analysis of several faults were discussed including an inflight bearing fault. It is intended that the HIDS diagnostic system will contain alarms that could be set to alert the maintainer and potentially the pilot of the need for maintenance or other action.

9. "Development and Validation of Algorithms for Engine Usage Monitoring Systems"
Presenter: Dr. M. Henderson DERA, UK

Currently a safe life fatigue methodology approach is used, and the individual usage of each engine component is not tracked. Safety issues are paramount. The adoption of individual aircraft usage monitoring systems would provide a better exploitation of the available life. An example of usage on the Red Arrows formation-flying team was given highlighting that formation position has a significant effect on fatigue consumption. Position 8 (aft and right position) consumes up to 30 times more fatigue life than position 1 (lead position) while performing the same maneuvers. Fleetwide monitoring (IAT approach) that accounts for individual aircraft usage will enable a fuller utilization of the critical components and so reduce life cycle costs.

10. "HUMS Loads Monitoring and Damage Tolerance: An Operational Evaluation"
Presenter: M. Basehore, FAA, USA

This paper presents the results of comparisons of the fatigue lives of four critical High Cycle Fatigue (HCF) helicopter components for different mission profiles and fatigue methodologies. Under an FAA program with Bell Helicopter and Petroleum Helicopters (PHI), a HUMS-equipped helicopter was operated in service under two different mission profiles. The damage rates for the four components were determined from the usage monitoring data and compared between the two mission profiles and the certification profile using both safe life and damage tolerance fatigue methodologies. The comparisons showed that usage monitoring provides significant benefits in service life extensions using a safe life approach and extended inspection intervals for a damage tolerance approach. It was noted that a study was done to investigate a simplified or "mini-HUMS" approach that obtained the greatest benefit for the least cost. It was found that simply monitoring the altitude alone resulted in significant life extensions since altitude has a major effect on loads and conservative worst-case assumptions had to be used for certification.

11. "CP-140 (P3) Structural Data Recording System"
Presenter: Mr. M. Oore, IMP, CA

This presentation described the Canadian Forces (CF) Structural Data Recording System (SDRS) used on the CP140A Acturus aircraft (a variant of the Lockheed P-3C) to facilitate Individual Aircraft Tracking (IAT). The SDRS fits were completed in 1997 and enable the CF to quantify individual aircraft fatigue usage. This allows calculation of optimum inspection times to reduce the frequency and cost of inspections required under an ASIP program while maintaining aircraft safety. Strain data was recorded at 4 critical locations, on the wing, and 1 on the empennage. A "rise/fall" criteria was used to minimize the volume of data saved while keeping the significant data. The SDRS zeros the strain gages at the start of each flight using a method that was substantiated by finite element analysis (FEA) of the structure. The CF has demonstrated that the SDRS has accuracy adequate for fatigue and crack growth analysis.

12. "NATO TCA Cycle Counting Study and Its Application"
Presenter: Ir. E. Moyson, Sabena Technics, BE

This presentation addresses the analysis of aging aircraft in a review of the results of a fatigue cycle counting study conducted on the NATO Trainer Cargo Aircraft (TCA) that were formally commercial B707's and transferred to NATO to be used as AWACS trainers and transport aircraft. The TCA aircraft accumulate significantly more damaging cycles due to touch-and-go landings, one engine out landings, and aircraft refueling training. The military KC135, although similar to the commercial B707, was designed to a <u>safe-life</u> requirement using 7178 aluminum. The B707 was designed to a <u>fail-safe</u> requirement using the more damage tolerant 2024 aluminum. Following the Osaka B707 mishap in 1997 involving failure of a horizontal stabilizer, a <u>damage tolerance</u> requirement was adopted in 1978.

The TCA aircraft are being operated well beyond the original design objective by 20+ years with continued operation planned until 2020. The highest time TCA will then have 2.6 times the aircraft design service objective. An effective Corrosion Preventive and Control Program (CPCP) is needed to control corrosion to level 1 or better (does not exceed allowable limits). Composite crack growth factors were used to indicate crack growth rates for TCA GAG cycles relative to the commercial B707 standard flight cycles. Inspections were adjusted based on calculations from pilot logbook data. It was noted that for aging aircraft there is no limit to the service life using damage tolerance provided that necessary inspections are done along with timely repairs and replacement of damaged structure.

13. "F16 Loads/Usage Monitoring"
    Presenter: Mr. D. J. Spiekhout, NLR, NE

This presentation described the NLR loads and usage monitoring program for the RNLAF F-16 fleet ongoing since 1990. In the past, fleet sample load monitoring was used. NLR and RADA Electronic industries in Israel have now developed a direct loads monitoring system called "FACE" (Fatigue loads monitoring and Air Combat Evaluation) that monitors 5 strain gages and is being fitted fleetwide by RNLAF. In 1996 fatigue load monitoring along with engine and avionics bus monitoring was added to the "ACE" system that had been used as a pilot debriefing tool. NLR uses a damage indicator they developed called the Crack Severity Index (CSI). Lockheed Martin Tactical Aircraft System (LMTAS) supported implementation of the FACE system. To date, about one half of the F-16 fleet has been fitted. Ground stations for data management are located at the squadrons and NLR. Results are reported to RNLAF on a weekly basis. In response to questions from the audience, it was noted that no major problems are expected with the reliability of the strain gage fit and an acceptable calibration procedure has been developed.

14. "CC-130 Data Analysis Systems for NLR, NE
    Presenter: Capt. A.M. van den Hoeven, NDHQ, CA

This presentation reviewed the OLM/IAT system for the Canadian Forces (CF) CC130 fleet that is used to assign usage severity and support optimized maintenance. The CF CC130 fleet is the highest time fleet in the world and is managed using ASIP. The CF moved to damage tolerance in the 1980's with the Durability and Damage Tolerance Analysis (DADTA) having been done by the OEM (Lockheed). The CF uses their C130 fleet in a broad range of missions and continued airworthiness depends on reliable measurements of operational loads under highly variable mission severities. The OLM/IAT system is fitted fleetwide and has been in operation and collecting data since 1996. A Data Analysis System (DAS) has been developed to use parametric data to assign usage severity and derive stress time histories on all aircraft. The derived stresses are checked against direct strain measurements recorded on selected aircraft of about 1/3 of the fleet. Thus both methodologies, parameter-based and direct measurement, are applied by the CF in their fleet load monitoring approach. The load monitoring system incorporates fracture mechanics models and fleet management tools and displays "red flags" for questionable data. An "engineer-in-the-loop approach is used for confidence building, continuous validation and special analysis of individual flights. The OLM/IAT system is planned to enable the CF to fully optimize maintenance in the near future and manage the damage tolerance program on the C130 fleet into the aging aircraft stages of its life.

15. "Service Life Monitoring of the B-1B and Impact on Flight Operations and Structural Maintenance"
    Presenter: Mr. A. G. Denyer, Boeing, USA

This presentation reviewed the B-1B Bomber fleet monitoring program. The B-1 structural design and qualification requirements included ASIP and a slow crack growth damage tolerance approach. Each of the 100 aircraft fleet is equipped with a solid state recorder called the Structural Data Collector (SDC) that collects loads and flight parameter data to support the ASIP analysis. The SDC collects data for 39 parameters (6 strain gages, 3 accelerometers, and 30 flight and mission parameters). The system has routines for data validation, data dropouts and automatic warnings. The strain gages were not calibrated before entering service. Calibration is done using strain gage records taken before and after flight. The ASIP analysis system has four elements: (1) a Loads Environment Spectrum System (LESS) to accumulate fleet usage data from the SDC data, (2) a Durability and Damage Tolerance Assessment (DADTA) to establish structure life under the operational conditions using the LESS data and (3) an IAT program to monitor each aircraft for damage accumulation based on the SDC data, and (4) a Force Structural Maintenance Plan (FSMP) to define inspections and maintenance requirements based on the DADTA and IAT information. The large quantities of data collected by the SDC required automation to process which had some problems early in the program but is now operating much better. The B-1 missions are quite severe, such as terrain following at high subsonic speeds. Actual usage was found to be much more severe than the design estimations and required maintenance adjustments to maintain structural integrity. The IAT data can be used to optimize the maintenance schedules to reduce cost and also support life extensions in the future.

16. "Eurofighter 2000: An Integrated Approach to Structural Health and Usage Monitoring"
    Presenter: Steve R. Hunt, BAE, UK

This presentation outlined the Structural Health Monitoring System (SHM) for the Eurofighter (EF) 2000 aircraft. The EF2000 is a 4-nation collaboration between Germany, Italy, Spain and UK. The SHM is installed on each aircraft and monitors fatigue consumption and significant structural events. The system is flexible and allows either parameter-based calculations or strain gage-based calculations or combinations of both. The SHM onboard system is configurable using upload data. It was noted that the amount of data retrieved per flight was very large (approximately 350 times the Tornado). The SHM data is analyzed and processed mostly on the ground based system and supports effective maintenance planning and management.

17. "Optical Fiber Sensing Techniques for Health and Usage Monitoring"
    Presenter: Dr. P. Foote, BAE, UK

This presentation described the applications of optical fibers as strain sensors for loads monitoring. The fibers employing Bragg gratings can be imbedded in composite structures and incorporated into an onboard SHM system. A considerable amount of processing is required for strain/load monitoring with fiber optics but the costs and complexity may go down as the technology matures. Although there are some advantages of optical fibers over strain gages (e.g. EMC interference), there were some practical concerns raised by the audience about embedded optical sensors in composite structures. For example, how can the sensor be repaired or replaced if damaged, how much do the sensors degrade the structure, and how can the sensor be inspected to determine if a change in response is due to the sensor or structural damage. Nevertheless, this technology is maturing and bears watching as it develops.

18. "Corrosion Modeling and Monitoring: Neural Networks and Neural Network-Based Sensor Systems"
    Presenter: Prof. W. F. Bogaerts, University of Leuven, BE

(No copies available) This presentation was a tutorial on Neural Networks (NN) followed by a discussion of the application of NN's to automated corrosion risk prediction. An intelligent stress corrosion crack detection sensor concept was discussed along with a NN inspection failure analysis for pits and cracks.

19. "Structural Health Monitoring of Fullscale Components using Acoustic Emission and Fiber-Optic Sensors"
    Presenter: Capt. S. Mays, WPAFB, USA

This presentation discussed the results of investigations on the application of advanced sensing systems (acoustic emission (AE) and fiber optic sensors) for structural health monitoring (SHM) systems. A joint US Air Force/Navy Smart Metallic Structures (SMS) program supported the investigations conducted at Northrop Grumman Corporation. The sensors augmented an onboard system capable of monitoring loads, detecting faults, and assessing the overall health of the aircraft. Damage detection of impacts and delamination in composite structures was considered a challenge for the future. Testing of AE and fiber optic sensors on specimens from coupon level to a full-scale wing attach bulkhead were conducted to verify the techniques. The next step will be creation of a flight-qualified system.

ROUNDTABLE DISCUSSIONS

The roundtable discussion was chaired by Professor Phil Irving. To kickoff the discussions, Jim Cronkhite presented conclusions developed by the Recorders during the meeting. It was pointed out that benefits in maintenance extensions can be achieved with structural loads monitoring whether one uses a safe life fatigue methodology (retirement extension benefit) or a damage tolerance fatigue methodology (inspection extension benefit). It was also noted that if the aircraft is operated <u>less severely</u> than design assumptions, monitoring provides benefits in <u>maintenance extensions</u>, but military aircraft may be operated <u>more severely</u> than the design assumptions and the benefits from monitoring would be in risk reduction or <u>enhanced airworthiness</u>, as shown in figure 1.

REQUIREMENTS ON FUTURE STRUCTURAL HEALTH MONITORING SYSTEMS

M. Neumair
IABG
Einsteinstraße 20, 85521 Ottobrunn
Germany

Summary

Due to the higher structural complexity of combat aircraft the time and costs for inspections and modifications are increasing. For economic and operational reasons, it is more and more important to carry through a modern aircraft maintenance and life extension program, which directly depends on the individual in service usage of combat aircraft.

As a result of the rapid progress in the development and production of new powerful data aquisition units, the use of complex onboard structural health monitoring systems for each individual aircraft, which directly monitors the point of time for inspections and modifications for life extension programs, is the future of economic maintenance.

This paper provides an overview of the experiencies with previous and existing in-service life monitoring systems, i.e. the OLMOS system of TORNADO. Based on this experience the hard- and software requirements for a new in-service structural health monitoring system, which is necessary for an economic usage of combat aircraft, will be discussed within the scope of the actual technical and operational limits.

Due to such complex structural health monitoring systems, rearrangements are necessary in planning and carrying out depot maintenance. This paper will point out some important alterations.

List of Abbreviations

A/C	Aircraft
ACI	Analytical Condition Inspection
FCA	Fatigue Critical Areas
GAF	German Air Force
IAT	Individual Aircraft Tracking
OLMOS	Onboard Life Monitoring System
SAT	Selected Aircraft Tracking
SHMS	Structural Health Monitoring System
TAT	Temporary Aircraft Tracking
WS	Weapon System

1. Introduction

The costs and time for military aircraft maintenance become a more and more important factor for the in-service availability of a weapon system. Due to the lower budgets of the air forces and to the increasing complexity of new A/C components it is necessary to make use of the whole life of an aircraft component in each individual aircraft.

Because of the high investments for the development and production of a new A/C - type, the in-service time of a military A/C weapon system is nearly twice that of former years. But to save time and money after the development, the series production of an A/C begins often before all results of the fatigue tests are known and all necessary modifications are implemented into the A/C series.

Another relevant reason for modification is the life extension of A/C due to longer or harder operational usage as supposed in the design phase. Due to the results of the fatigue test, the FCA's were defined with an individual in-service life. This in-service life depends on the operational usage of each individual A/C. As consequence of the political necessity of world wide operations of the air forces with different missions and different environmental parameters the operational usage of the individual A/C gets a wider spectrum.

The calculation of used life can not be done as in the past with an average flight parameter usage spectrum for the whole fleet and a vertical acceleration spectrum for the individual A/C. Now it is necessary to know the individual flight parameter and operational stress spectrum of the various FCA's with an extended IAT to exploit the life of each fatigue critical component individually for each A/C.

So it is necessary for the A/C users to have a complex and intelligent instrument for the individual A/C structural fatigue tracking. But for a cost effective usage with a high in-service availability with a minimum of restrictions on operational usage, the planning of maintenance actions is a main point of a modern structural health concept. Thus, not only an intelligent and highly developed individual aircraft fatigue life monitoring system but also a flexibel and intelligent maintenance planning tool is necessary.

The basis for monitoring of FCA's, which were defined during full scale fatigue tests or component

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

tests, is that the design load distribution at each structural area is identical with the in-service load distribution. The past has shown, that due to changing operational usage, which was not implemented into the fatigue tests, and due to simplifications in the fatigue tests, which have to be made at the load initiation, at some areas in flight measurements have been necessary. Thus, an important third part of a cost effective and modern structural health monitoring system is in-service measurement of complex load distributions at identified or potential FCA's which are included in the TAT (Fig. 1).

The following chapters will provide a short overview of existing concepts for fatigue life monitoring in the GAF and then the requirements of a modern SHMS and a proposal for a possible future SHMS are outlined.

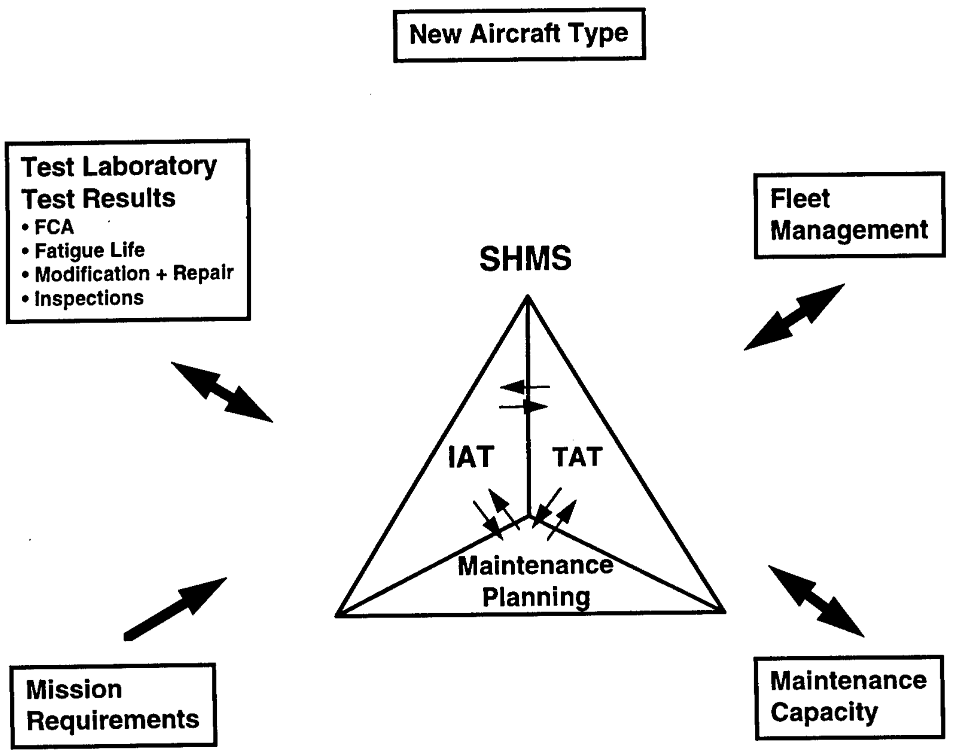


Fig. 1: Components of SHMS

2. Overview of existing Fatigue Life Monitoring Systems in the GAF

The two important WS in the GAF are the F-4F Phantom and the TORNADO. The basic concept for the fatigue life monitoring is similar for these two WS. Both have simplified IAT and a complexer SAT.

The influence of different operational usages were calculated with the SAT data acquisition system and then imported into the fatigue life calculation of each individual A/C.

But due to the long time scale between the development of the F-4 and TORNADO life monitoring concepts, there are many differences.

WS F-4F

Hardware:
- G-counter in each individual A/C
- VgH-recorder distributed in 10% - 15% of the fleet on a statistically representative basis with the measurement of:
  - Altitude
  - Velocity
  - Vertical Acceleration
  - Flap-Position
  - Slat-Position
  - Main Landing Gear Position

Monitoring concept:
Only one damage index for the A/C

SAT:
Evaluation of the VgH-recorder data and calculation of S-N-data for the fleet dependent on the actual operational usage with a crack growth model according to the US-ASIP.

IAT:
Evaluation of the g-counter data and calculation of the individual damage index for each individual aircraft with the above calculated S-N-data.

FCA's:
The FCA's and the respective inspection and fracture limits were provided by the manufacturer, dependent on the results of Full Scale Fatigue Tests.

Maintenance and Modifications:
Many FCA's have to be inspected with fixed maintenance interval, because they cannot be monitored with this g-counting-based monitoring concept.
Calculation of the modification point of time for the FCA's which are dependent on the vertical acceleration and relating this modification to the nearest maintenance.

Advantages:
- Low costs of the data acquisition devices
- Low costs of the fatigue life monitoring and maintenance planning

Disadvantages:
- Only FCA's, where the damage index depends on the vertical acceleration, can be monitored.
- Long time for the response of system failures, because of magnetic tape data storage for about 40 flights ⇒ high data loss rate
- High maintenance costs because of:
  - Fixed maintenance intervalls ⇒ loss of useable fatigue life
  - Higher safety factor because of the fatigue life monitoring concept with only one flight parameter

WS TORNADO

In the WS TORNADO the powerful OLMOS system is integrated [1].

Hardware:
- Various sensors for a large number of flight parameters
- Data Acquisition unit for the various flight parameters
- Maintenance Recorder for the data storage
- Strain Gauge Amplifier for the measurement of stress or loads at FCA's

Monitoring concept:
The monitoring concept is devided into three sectors
IAT Individual Aircraft Tracking
SAT Selected Aircraft Tracking
TAT Temporary Aircraft Tracking
Each main component of the aircraft is monitored separately

IAT:
Evaluation of the pilot parameter set (simplified flight parameter data acquisition of Nz, weight, wing sweep angle and stores) and calculation of the individual damage index with the S-N-data calculated with the results of SAT and TAT

SAT:
Recording of all relevant flight parameters
Calculation of:
- Flight hour related damage
- Usage spectra survey
- Damage comparison of the parameter sets dependent on the results of TAT

TAT:
Recording of the relevant flight parameters and special strain gauge signals
Calculation of:
- Strain based damage calculation
- Full parameter set / stress correlation
- Pilot parameter set / stress correlation

FCA's:
The FCA's and the relevant operational limits were calculated from the results of the fatigue test (MAFT)

Maintenance and Modifications:
Large depot maintenance intervals.
FCA's with a high damage index were inspected on the damage index fleet leader within the scope of an Analytical Condicition Inspection
The modifications were installed dependent on the consumed fatigue life.

Advantages:
- Low costs for the IAT
- Measurement of real strain with strain gauges
- Measurement of a complex flight parameter set
- A/C with FCA's with high damage index were only inspected at ACI

Disadvantages:
- High effort for SAT and TAT and a higher risk of data loss
- High effort for special measurements of complex load distributions
- Individual planning of modifications only dependent on the pilot parameter set.

- Fixed maintenance intervall necessary for inspections.

3. Requirements on future SHMS

Future military A/C have a higher structural complexity because of the large integral components. The repair and modification of fatigue critical areas of such structural components produce higher maintenance costs and longer maintenance time. To compensate for such higher costs, which is necessary for an economic maintenance of military aircraft, the installation time for modification has to be planned more accuratly to use the whole fatigue life of the components. This also produces a higher in-service availability of the A/C because of the longer time interval between maintenance tasks.

Due to the lower hardware costs of intelligent on-board data acquisition units the integration of a complex, intelligent and extensionable fatigue life monitoring system in each individual aircraft is a main part of future fatigue life monitoring systems. But not only the individual aircraft fatigue life monitoring system is a main part of future structural health monitoring systems. Also an intelligent and flexible maintenance planning and logistic system is necessary for an economic fleet management.
A third important part for the SHMS is the temporary monitoring of FCA's with special in-service load measurements.

So the main requirements can be divided into the following three sectors

Individual Aircraft Tracking:
The IAT should consist of:
- an intelligent and programable data acquisition unit in the A/C
- a Ground Station unit for simplified data checking, data evaluation and damage index calculation
- a Fatigue Analysis Center for extensive data checking, data evaluation, damage index calculation and fatigue parameter calculation
- a Maintenance Coordination Center for maintenance orders

Temporary Aircraft Tracking:
The TAT should consist of
- an additional intelligent data acquisition unit installable in each A/C for special load / stress measurements, which is integratable in the IAT data acquisition unit
- a Fatigue Analysis Center for the extensive data checking, data evaluation and fatigue calculation for the special load /stress measurements

Maintenance Planning:
The Maintenance Planning should consist of
- a Ground Station unit for the import of enforced maintenance tasks
- a Maintenance Coordination Center for the economic management of modification and inspection time dependent on operational usage and maintenance capacity

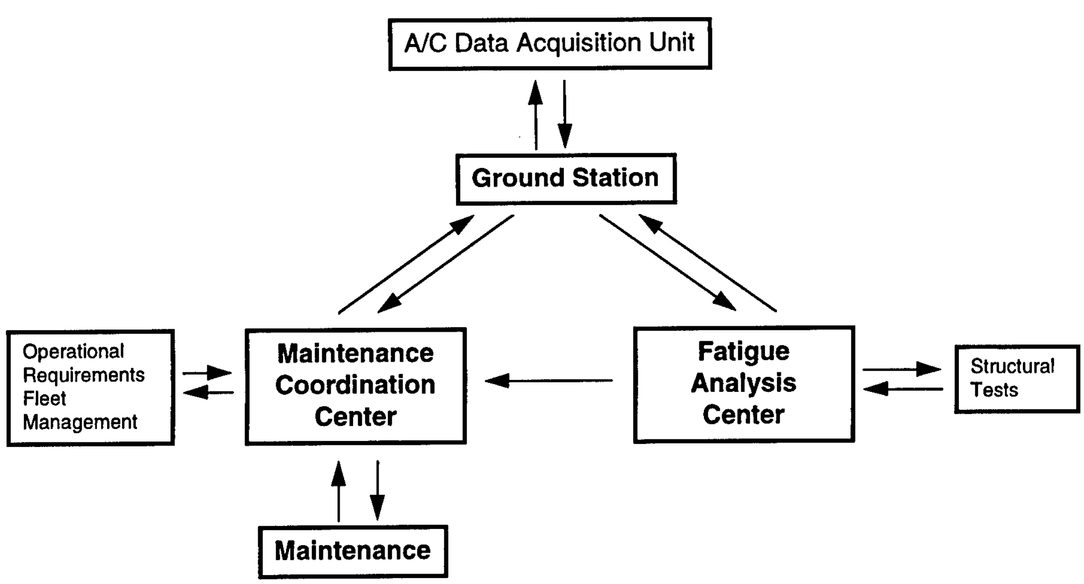


Fig. 2. Proposal for a modern SHMS

4. Proposal for a future SHMS (Fig. 2)

The primary part of a SHMS is the concept of the flight parameter monitoring system. The next chapter makes an assessment of the following main monitoring concepts:

- Onboard data evaluation and fatigue calculation
- Onboard data reduction and on ground data evaluation and fatigue calculation

Both concepts have positive and negative aspects which are shown in Tab. 1.
As conclusion of the advantages and disadvantages of both systems a combination of the two systems would be the best solution for future flight parameter monitoring systems, which is shown in Fig. 3.
In the following a proposal for a modern SHMS concept is given by taking the above requirements as basis (Fig. 2). This SHMS concept is also divided in the three sectors

- IAT
- TAT
- Maintenance Planning

The sectors are not self-contained. Each sector needs inputs from the other sector for an optimum of structural health monitoring and structural fatigue fleet management.

Individual Aircraft Tracking IAT

A/C:

- Recording of the relevant flight parameters
  Onboard data reduction software like a Sequential Peak with Time software with Master-Slave function between all flight parameters and the weight and store information
- Recording of relevant real stress / loads
  For example a Sequential Peak with Time software with Master-Slave function between the load / stress sensors and additionally with the flight parameters and weight and store information
- Recording of the weight and stores
  Only slave function
- Recording flight by flight
- Sensor check before and after each flight
- Logbook for malfunctions and flight information
- Large electronic data storage
- Onboard damage index calculation for main components

Onboard data evaluation and fatigue calculation		Onboard data reduction and on ground data evaluation and fatigue calculation	
Advantages	Disadvantages	Advantages	Disadvantages
• Low onboard data storage	• No information about the flight parameter versus time	• All flight parameter available in Ground Station	• Large onboard data storage and data transfer
• Low effort in data management	• Fixed damage index calculation algorithm	• Recalculation of damage index for new FCA is possible	• High effort in data management
• Damage index needs no on ground calculation	• Fatigue parameter have to be implemented at individual A/C	• Fatigue parameters have only to be changed at a few Ground Stations	• Damage index needs on-ground calculation
	• High effort for onboard hard- and software	• Easy update of Ground Station hard- and software	
	• Short interval for onboard hard or software update	• Large interval for onboard hard- or software update	
	• Recalculation of damage index for new FCA is not possible		

Tab. 1 Comparison of different onboard flight parameter monitoring systems

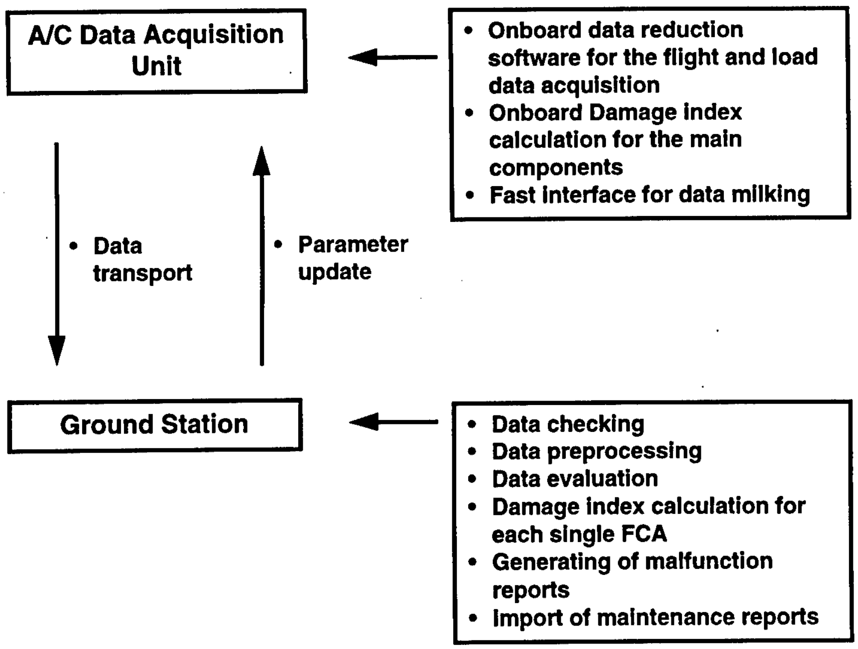


Fig.: 3 Flight parameter monitoring system

Ground Station (Fig. 4):
The Ground Station is part of the individual A/C SHMS were all relevant A/C fatigue data have to be calculated or accumulated. The main features are:

- Data milking from the A/C and A/C IAT-parameter update
  Use of hand held terminals as interface between A/C and Ground Station
- Simplified data evaluation and data checking
- Sensor checking and creation of malfunction reports
- Calculation of the individual damage index of each FCA for the individual A/C
- Export of the flight data to the Fatigue Analysis Center
  The fastest way would be in future the online data exchange
- Export of the damage index to the Maintenance Coordination Center
  (Online data exchange)

Fatigue Analysis Center (Fig 4):
The Fatigue Analysis Center is the headquarter of the Fatigue Monitoring. All structural information like the fatigue test results or the maintenance results meets in this center. The main features are:

- Data evaluation and data checking
- Calculation of the fatigue parameters dependent on the new operational usage
  The fatigue parameters were calculated for each FCA
- Update of the actual fatigue parameters and damage index for existing and new FCA and for each A/C and export to the Ground Station and Maintenance Coordination Center
- Calculation of damage index rates for each A/C and for squadrons and fleet and export to the Maintenance Coordination Center
- Export of malfunction reports to the Ground Station
  Failures at sensors or equipment were reported
- Creation of software changes and update of the software at the Ground Station
  The main fatigue calculation software with the fatigue parameters is implemented in the Ground Station and not in the A/C. Necessary software changes or parameter changes can be made with low effort. Recalculation of data with a newer software is possible
- Definition of TAT-tasks, fatigue tests and new FCA's

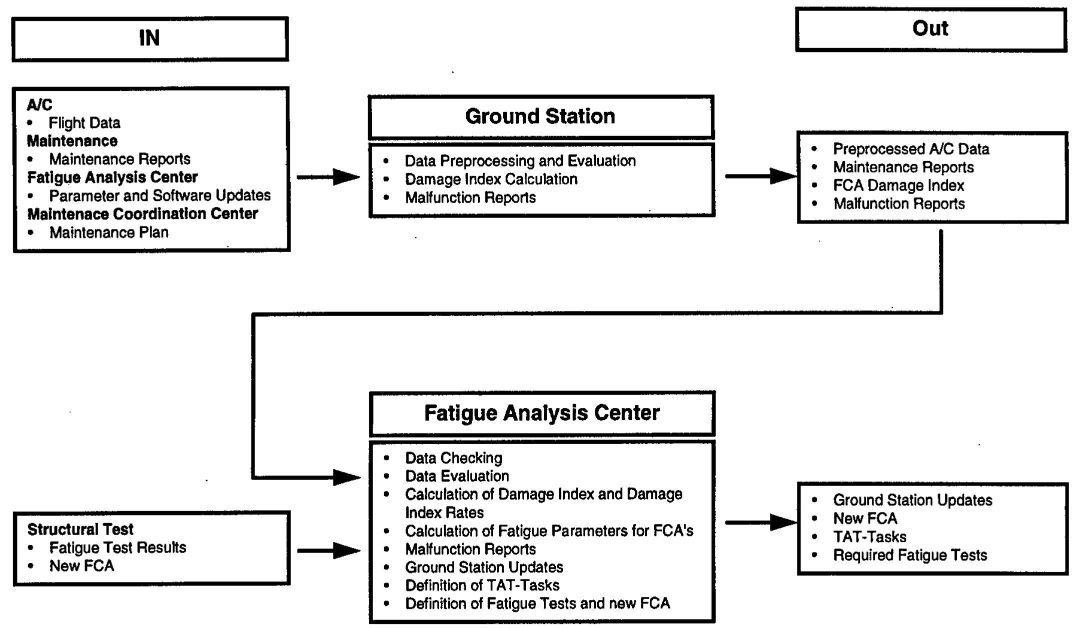


Fig. 4: Components of Fatigue Monitoring and Analysis

Maintenance Coordination Center (Fig. 5):
The Maintenance Coordination Center is the headquarter of all maintenance activities with regard to the requirements of maintenance and fleet management.

- Calculation of the FCA modification or inspection time
  Implementation of all relevant data of future operational usage to project the expected consumation of fatigue life.
- Creation of a maintenance plan for the individual aircraft
  Fatigue maintenance activities are only dependent on consumed fatigue life.

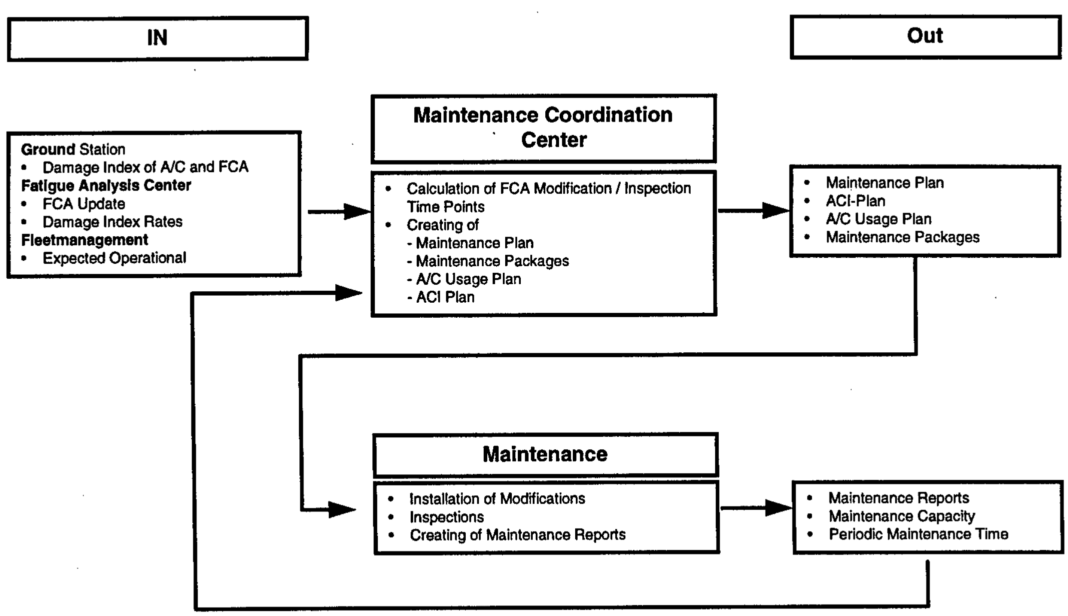


Fig. 5: Components of Maintenance

- Creation of inspection or modification packages for A/C components
- Creation of ACI-Plans
- Creation of A/C usage plans as basis for the A/C rotation between squadrons and for the operational usage plan

Temporary Aircraft Tracking TAT (Fig. 6)

The TAT is only necessary for new FCA's or, if the fatigue test loads are not relevant, for the in-service flight loads
In the following only the tasks are shown, which are not part of the IAT

A/C:
- Recording of the special TAT load data onboard data reduction software like a Sequential Peak with Time software with Master-Slave function between the load / stress sensors and additional with the flight parameters and weight and store information
- TAT-sensor checking before and after each flight

Ground Station:
- Export of the TAT and load parameters to the A/C via a hand held terminal
- Simplified TAT-data evaluation and data checking
  Short reaction time in case of malfunctions
- TAT-sensor checking and creation of malfunction reports
- Export of the IAT and TAT flight data to the Fatigue Analysis Center
  (Online data exchange)

Fatigue Analysis Center:
The Fatigue Analysis Center is the headquarter of the TAT
- Definition of TAT-tasks
- TAT data evaluation and data checking
- Update of TAT and load parameters and export to the Ground Station
- Calculation of new FCA fatigue limits
- Update of the fatigue parameters of each individual A/C
- Export of the new FCA fatigue limits to the Maintenance Coordination Center

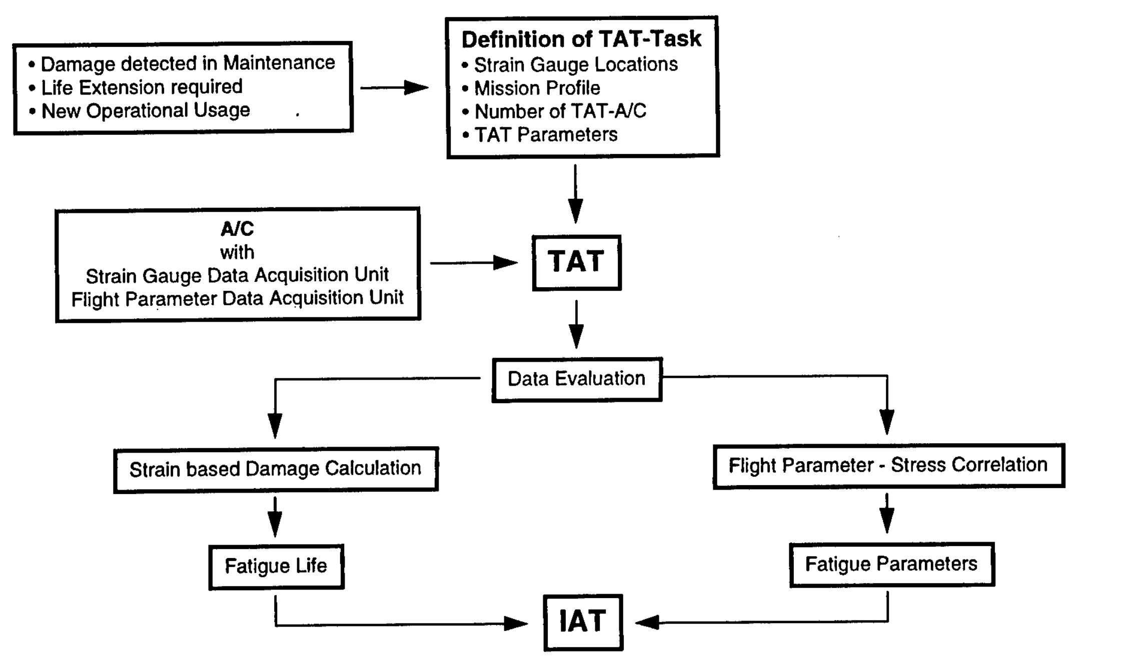


Fig. 6 Temporary Aircraft Tracking TAT

Maintenance Coordination Center (Fig. 5)
- Calculation of the new FCA modification or inspection time and updating of the maintenance plan for the individual aircraft
- Export of the maintenance plan to the Ground Station

Maintenance Planning

Maintenance Coordination Center (Fig. 5):
- Import of the operational requirements from the fleet mangement

- Import of the maintenance capacity from the maintenance center
- Import of the FCA fatigue limits from the Fatigue Analysis Center
- Calculation of the FCA modification or inspection time and creation of a maintenance plan for the individual aircraft
- Creation of inspection and modification packages individual for each A/C under the economic management point of view dependent on the individual damage index
- Export of maintenance plan
- Definition of ACI packages for the damage index fleet leader
  For flight safety it is necessary to compare the in-service fatigue damage rate with the fatigue test results. The best instrument from technical and economical point of view is the accurate inspection of all possible FCA's during a ACI. The results of the ACI inspections are relevant for FCA limits and for the inspection and modification packages of the whole fleet.

5. Conclusions

Due to increasing costs for inspections and the request of more in-service availiability of a WS the previous maintenance concept of periodic inspection of FCA's is no longer applicable to future military aircraft.

Thus, the use of intelligent powerful data acquisition units and a fast and direct monitoring and maintenance planning concept is necessary for a modern SHMS. The main points of this concept are to avoid onboard damage calculation software with a long reaction time for software updates, but also the use of great data storages with the risk of loss of data because of hardware malfunctions and long ways for data storage until data evaluation.

The above article discusses the concept of splitting the main tasks of the IAT in four units.

- The A/C unit
- The Ground Station unit
- The Fatigue Analysis Center unit
- The Maintenance Coordination Center unit

This units have to work together via online data exchange to have all relevant fatigue data of each individual A/C available at any time. This is a main point for an optimum of economic maintenance of FCA's with a high in-service availiability and a minimum of restrictions in the operational usage.

6. References

[1] R. Neunaber:
''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

FUTURE FATIGUE MONITORING SYSTEMS

Squadron Leader Steve R Armitage RAF
SM17b(RAF), SMG, RAF Wyton, Huntingdon, Cambs, PE17 2DL, UK

Dorothy M Holford
Aerostructures Dept, Defence Evaluation & Research Agency, Farnborough, Hants, GU14 6TD, UK

1. INTRODUCTION

The RAF, in common with most of today's air forces, operates a wide range of aircraft, of varying sizes, roles and ages. At the one extreme, the Canberra fleet is now 50 years old and still providing sterling service in the photo-recce role. At the other extreme, Harrier II has been in service for 10 years, and has proven itself to be a highly capable ground-attack fighter. Moreover, EF2000, Nimrod 2000 and C-130J are all due to enter service within the next 5 years or so. The operation of all of our fleets has been underpinned by the close monitoring of fatigue, the aims of which have been to improve the management of structural integrity and to ensure that we can safely operate our aircraft up to their design lives.

Despite advances in aircraft technology, our fatigue monitoring effort continues to rely upon some relatively aged equipment, with correspondingly aged analysis techniques. In recent years, advances in IT and data processing have revolutionised the aircraft designer's ability to install onboard sensors and systems, and the software designer's ability to generate compact and rapid airborne and ground-based processing equipment. The RAF is now looking closely at the procurement of state-of-the-art fatigue monitoring and processing equipment, and the formulation of a strategy by which such equipment can be adapted for use across as many fleets as require it.

This paper will review the RAF's current fatigue monitoring effort and the emerging systems, with particular reference to Harrier II's Fatigue Monitoring and Computing System (FMCS) and EF2000's Structural Health Monitoring (SHM) system. Finally, future system requirements and options will be reviewed.

Although the paper's emphasis is firmly upon fixed-wing aircraft systems, some consideration will be given to emerging helicopter Fatigue Usage Monitoring Systems (FUMS). FUMS is a development of Health and Usage Monitoring Systems (HUMS), currently being introduced into MOD service.

2. CURRENT SYSTEMS

Fixed-wing Systems

The majority of the RAF's aircraft fleets are of safe life design; the aircraft enter service with a quantified fatigue life, underwritten by the manufacturers through a formal structural qualification programme. The qualification, however, together with its acceptance by the Ministry of Defence (MOD), are both conditional upon the aircraft being carefully monitored in service for fatigue consumption. The following paragraphs summarise the UK's use of full-scale fatigue testing, and RAF's traditional method of fatigue monitoring.

Each design should be subjected to 2 Main Airframe Fatigue Tests (MAFTs): a development standard test (DMAFT) and a production standard test (PMAFT). The former would be subjected to a design loads spectrum, and would identify those areas for which major design shortcomings would lead to early in-service problems. The latter would be expected to be subjected to a refined spectrum, taking into consideration in-flight data recorded by prototype/development aircraft, and by early production-standard aircraft. The timing of the PMAFT is, therefore, significant; on the one hand it is tempting to start the PMAFT as early as possible, to maximise its lead on the fleet. However there is also merit in delaying start until in-service operations are well-established, and the loads database is

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

representative. A compromise must, therefore, be sought.

Interpretation of the results of both MAFTs must be supported by Operational Loads Measurement (OLM) activities, preferably continuous, or repeated every few years. The primary aim of OLM programmes is to highlight any differences between actual, in-service structural loads, and the PMAFT spectrum, thereby allowing suitable corrections to be made to the structural lives cleared by full-scale testing. An OLM programme will also allow the structural implications of supplementary sortie profiles and the carriage of new stores to be quantified.

The flying undertaken by the in-service fleet is carefully recorded. Each sortie is logged against standard profiles, and relevant sortie data (start-up and shut-down masses, landings, sorties carried/released, air-to-air refuelling etc) is recorded at the end of the flight by the aircraft captain. Finally, and for the vast majority of our in-service fleets, "g-counts" are also recorded. The g-counts are provided by the aircraft's on-board counting accelerometer, or "Fatigue Meter". Figure 1 is an illustration of one such meter; the device entered service in 1954 and remains in service on $^{2}/_{3}$ of the RAF's active fleets (the remaining fleets rely upon a variety of alternative systems, of varying effectiveness). It contains a series of lock-and-release mechanisms that allow it to register exceedances of pre-defined levels of normal acceleration, N_Z. If mounted somewhere near the aircraft's centre of gravity, the fatigue meter gives the user an overall view of the N_Z spectrum to which the airframe has been subjected during any given sortie. But of what value is this in terms of aircraft fatigue? The majority of fatigue-critical locations in an airframe are located around the wing attachment/centre-fuselage, where structural loading is governed, to a significant degree, by N_Z. Hence, N_Z can be used as a reliable means of estimating stresses and, therefore, fatigue damage. The flight-by-flight recording of g (N_Z) counts, with the supporting sortie information outlined above, can be processed via a Fatigue Meter Formula (FMF), such as the one at Figure 2, to give a meaningful measure of fatigue consumption, usually measured as a proportion of a cleared fatigue life. The entire process is supported by careful monitoring of a fleet's in-service loads environment, typically through an OLM, or equivalent, programme.

A FMF will be generated for a specific structural component or feature thereof. The locations for which FMFs are generated tend to be those that a designer may expect to be subject to fatigue damage, supplemented by other areas that have been found to be vulnerable during full-scale testing. It is not uncommon for a mature fleet, well into its design life, to be monitored by a suite of 6 or 8 FMFs. Cumulative fatigue figures, for each aircraft and location, are published on a monthly basis for the purposes of fleet management. Processing is, for most fleets, performed by a centralised agency, remote from the front-line. For some fleets, however, local management aids enable squadron level calculations to be performed. This transfer of analysis effort, from central location to squadron, is a theme to which this paper will return in later discussions.

The above paragraphs summarise the RAF's traditional fatigue monitoring technique, as employed for most of its fleets; a simple and relatively cheap system that has been largely unchanged for the past 30 or 40 years. FMFs have become more refined in their generation, and experience has allowed us to identify more rationally those structural items that may need to be monitored. Moreover, for the majority of monitored locations, the RAF's current system has been found, on the whole, not to be significantly in error; in general, any over estimate of fatigue for one sortie tends to be balanced by an underestimate for another sortie and, over time, the fatigue damage consumed at a given location can be adequately monitored.

However, it has long been recognised that considerable room for improvement exists, and that the current system is woefully inadequate in a number of areas.

a. The fatigue meter cannot address the monitoring of pressurizations.

Unless a dedicated pressure sensor can be fitted, the usage of the pressure cabin relies entirely upon a captain's ability to retrospectively count and record pressurization cycles.

b. The fatigue meter is unable to record usage information. Usage of flying controls, airbrakes, spoilers etc can only be estimated to varying degrees of accuracy. The structural attachments of flying controls have been found in the past to be vulnerable to fatigue damage and our inability to generate usage spectra hampers our ability to predict lives and refine designs.

c. The fatigue meter records N_Z exceedances, but not when the exceedance took place. All counts are assumed to have occurred at the aircraft's notional mid-sortie mass, and the fatigue implication associated with those counts accumulated at the early part of a sortie, at higher mass, can be significant.

d. The fatigue meter is incapable of quantifying by how much the maximum N_Z level has been exceeded. Occurrences of over-stresses are only, therefore, brought to the attention of engineers if the aircraft captain is aware of the exceedance.

e. The monitoring of fatigue is near-impossible at any structural location remote from the aircraft's centre of gravity (outer wing or empennage for example), for which stresses may be related more to roll or pitch acceleration, or control usage than N_Z.

f. The fatigue meter is accurate to with +/-0.1g, and the maintenance of even this less-than-ideal accuracy depends heavily upon a programme of scheduled meter maintenance and calibration. An unwillingness to invest, service-wide, in more modern and reliable versions has been due primarily to cost, and an inability to quantify the savings that would be made using a more reliable g counter.

g. The generation of the configuration and profile coefficients used in the FMFs is based on a "point in the sky", a single combination of key flight parameters that represents a typical flying condition. The point is based upon measured usage, with a host of supporting assumptions. The "point in the sky" approach has been found to be adequate over time, but on a sortie by sortie basis, its accuracy is undermined by the assumptions that underpin it.

h. Buffet and other high frequency modes of structural loading cannot be measured. Allowance can only be made by manually recording the length of time that the captain considers that he was flying in a buffet regime.

Identifying the weaknesses of the RAF's current monitoring strategy has always been relatively easy. It is only in recent years that we have been given the opportunity to effect meaningful improvements. The development of the digital databus, and its ability to offer the continuous recording of a wide range of flight parameters seemed to offer a step-change in our ability to monitor many in-flight phenomena, including structural loading. But it is only through recent advances in solid-state digital data recording/processing that we have been given the opportunity to review, and possibly supersede altogether, our fatigue meter dependency.

Operational Loads Measurement

The following paragraph will consider the subject of OLM, the RAF's experience in which will have a bearing on later discussions regarding FMS selection. Each in-service aircraft type is required to be subjected to OLM at least every 5 years, and ideally on a continuous basis; such OLM activity allows an operator to monitor fleet usage, and the effects of changes in usage parameters. Our Tornado fleets are in the advantageous position of having a permanent, part-fleet fit of OLM equipment; strain gauges are affixed to structural items that are known, from testing or in-service failures, to be vulnerable to fatigue, or for which a more comprehensive loads appreciation is required. By continually monitoring in-service loads/stresses/fatigue, and by comparing reality with the results of the fatigue meter-based monitoring of key critical components, annual correction factors can be produced for fleetwide application. This approach has proven invaluable, particularly given the uncertainties associated with the effects of wing-sweep and some of the assumptions made during the aircraft's design; the approach also forms the basis of our use of the Harrier II's FMCS, to be discussed in detail later.

Without exception, OLM programmes performed on RAF aircraft have employed direct strain measurement, via strain gauges, as the means of determining structural stresses. The RAF, supporting industry and Government research establishments have a wealth of experience of strain gauge installation, maintenance and data interpretation. This experience has significantly influenced the 'national' position with regard to emerging FMSs.

Helicopter Systems

Helicopter fatigue monitoring is limited simply to the manual recording of flying hours, and the comparison of these figures with the fatigue lives given to specific components; though underwritten by its manufacturers, a component's fatigue life will be calculated using assumed flight loads data and usage spectra; most helicopter types have not been subjected to full-scale fatigue testing. Historically, there has been no formal system for the monitoring of in-service usage for many of our helicopter fleets; changes in sortie content, sortie frequency or all-up mass, have not routinely been measured. The impact of such changes upon the validity of design spectra and, therefore, fatigue lives, has not been as carefully monitored as is the case for the fixed-wing fleets. Historically, however, airframe and component fatigue lives have been founded upon conservative, worst design case assumptions; they have been able to absorb adverse changes in in-service usage.

The practicalities of installing strain gauges on rotating parts, to determine local loading, have meant that the OLM activities conducted on fixed-wing fleets have not been performed to the same extent on our helicopters. Those programmes that have been performed, known as Operational Data Recording (ODR) exercises, have generally confirmed design assumptions.

EMERGING SYSTEMS

Fixed-wing Systems

Thus far, we have considered the RAF's current system for monitoring aircraft fatigue, and its numerous inadequacies. We have also discussed the Service's considerable OLM experience. It is now worth turning attention to developments in fatigue monitoring over the last 10 years or so, by considering 2 specific systems: Harrier II's FMCS and EF2000's SHM. During the discussion of the systems, the issues of direct versus indirect monitoring, and airborne versus ground-based processing will be introduced. Though neither of the above systems has entered service yet, we are already learning lessons from their design, lessons that are influencing our approach to future FMSs.

Harrier II - FMCS

Harrier II offered a major advance in terms of data gathering in that it is fitted with a Mil Std 1553 databus, over which a wide variety of flight parametric information is available continuously. Harrier II designers were faced with the realistic possibility of being able to extract a wide variety of data to assist in the calculation of fatigue, at a range of structural components.

Direct versus Indirect Monitoring. The calculation of component-level structural fatigue requires a stress spectrum that can be broken down into cycles of stress of given mean and amplitude. This can be achieved, at component level, by installing a strain gauge, converting strain information into stresses, and employing a counting algorithm to further convert the stresses into the cycles required. Alternatively, stresses can be calculated, again at component level, from combinations of known flight parameters (angular accelerations, rates etc). The former is known as the "direct" method, and the latter "indirect"; both methods have advantages and disadvantages. The relative simplicity of the direct method is balanced by the need for continuous husbandry/calibration of gauges and their wiring. The latter offers a more maintenance-free approach, in so far as system health checks can be done automatically. However, parametric-based systems are highly sensitive to sensor errors. Moreover, considerable effort and expense is required in the derivation of the algorithms necessary to calculate stresses from parametric data. In particular, the retrospective adjustment of the algorithms, to allow for new stores configurations or flying patterns, can be extremely costly. Finally, any parametric analysis must be supported by a thorough understanding of the loading actions applicable to the component(s) being monitored.

The FMCS designers had to select the preferred way ahead, and it was the UK's experience in strain gauge usage, as discussed under the earlier OLM heading, that swung the balance in favour of the direct method. As a result, FMCS employs 16 strain gauges, located as per Figure 3A, supported by a range of parametric data channels. The system can be reconfigured to monitor additional locations if required. A schematic of FMCS is at Figure 3B. FMCS was designed around an onboard data processor, the Fatigue Monitoring Computer (FMC), which continually calculates component fatigues through the sortie. Once an aircraft has landed, the FMCS provides component-level fatigue consumption figures, together with supporting parametric data, fatigue profiles, and "Frequency of Occurrence Matrices" (FOOMs); all of these are recorded during the sortie at a rate, or "Snapshot Interval", determined by the severity of the sortie being flown. In addition, the onboard analysis can be supplemented by the installation of a 16Mb Raw Data Recorder (RDR), to store a wide range of parametric and strain data, allowing for ground-based and ad-hoc investigations. FMCS will be fitted to our single-seat Harrier GR7s as a part-fleet fit, and employed in the same way as Tornado's SUMS to generate annual correction factors to be applied to the fleet fatigue figures generated by supporting suites of FMFs. It will, however, be fitted to our entire twin-seat Harrier T10 fleet; FMCS will be the sole fatigue monitor for T10.

FMCS represents the first of the emerging FMSs, and allowed the UK to establish itself as a firm advocate of the direct monitoring method, a position that we have maintained, and which has led to the development of 2 FMSs for EF2000.

EF2000 SHM

EF2000's SHM will enter service in 2 forms: the first, the 'baseline fit' is an indirect system, written into the EF2000's original specification and co-funded by all 4 partner nations. It employs an extensive suite of algorithms to calculate fatigue at 10 monitor locations, illustrated at Figure 4A. The development of the algorithms required has taken many years; the costs associated with their development have been phenomenal and would probably rule out equivalent parametric systems on anything other than multi-national, collaborative projects, with large anticipated production runs. The second system, the 'national fit' is a direct system, funded jointly by UK and Spain. Consistent with the UK's direct method preference, it employs 16 strain gauges, the data from which is used to determined fatigue at 16 monitor locations, illustrated at Figure 4B. As with FMCS, data processing is onboard, but a Bulk Storage Device (BSD) can be fitted to allow for the retrospective and ground-based analysis of data, if required for special investigations or to confirm the integrity of the airborne system. The BSD can also support future OLM activities.

Though FMCS and SHM have yet to enter service, both have proved invaluable in prompting the UK to consider its preferred FMS design, and the constraints within which any future UK FMS would have to be designed, installed and operated. Both FMCS and SHM are, by today's standards, relatively old designs. Technology has advanced apace since their development, and already a number of more advanced alternatives are on the market. We have already considered the direct versus indirect system issue; let's now consider the question of airborne versus ground-based processing.

Airborne versus Ground-based Processing. Both FMCS and SHM were envisaged as airborne processing systems, primarily to avoid the need for large and heavy data recorders. However, in recent years, advances in solid state recording technology have resulted in a range of small, yet high-capacity recorders being available on the open market. The reduced size and weight of such recorders, combined with increased capacity, mean that enormous quantities of data can be stored whilst avoiding the need for data-compression, and the loss of detail associated with it. The one advantage of airborne over ground-based processing, is post-sortie availability of data. It is rare, however, that fleet managers require fatigue information immediately after a flight. Moreover, ground-based processing offers considerable flexibility in terms of analysis of raw data, and the re-analysis of data some considerable time after a sortie. If processing can be done on the ground, the aircraft need only be fitted with onboard data acquisition equipment, with commensurate reduction in cost, weight and maintenance requirements. The post-processing of comprehensive data records also allows for the retrospective analysis of a lifetime's worth of data following any improvement to analysis methods, or the identification of additional 'monitor locations'. Refinements to analysis software, revised algorithms etc, can readily be uploaded into a Ground Support Station (GSS), of which there may be one or 2 per unit, far more readily than into every aircraft within a fleet. It must be stressed, however, that some airborne processing will be required; the ability to log any exceedance and sensor malfunctions will highlight to operators the need for post-sortie exceedance checks, and alert personnel to the corruption or invalidity of data. In recent studies, the UK has carefully considered the relative merits of both airborne and ground-based processing, concluding that the latter offers a more flexible approach.

Helicopter Systems

To reduce the levels of vibration produced during helicopter flight, HUMS are being installed in MOD helicopter fleets. HUMS employs the latest technology; its sensors continually monitor the vibration levels to which an aircraft is being subjected an alert engineers if levels exceed pre-set datum levels. Adjustments can be made to keep vibration levels to a minimum, and hence levels of the high-cycle fatigue to which many helicopter components are vulnerable. HUMS has the added benefit of being able to record flight data, from which airframe and component usage spectra can be derived. Whilst HUMS may not strictly be classed as a fatigue monitoring system, it was considered that its fatigue controlling function deserved mention in this paper.

FUTURE SYSTEMS

Fixed-wing Systems

Thus far we have looked in some detail at the RAF's standard, fatigue meter-based fatigue monitoring system. We have also considered 2 emerging systems which, though they have yet to enter service, represent major advances in monitoring/ processing technology. Looking to the future, a recent study by the MOD's Defence Evaluation and Research Agency (DERA), into the way-ahead for fatigue monitoring, highlighted several key features that should be considered as part of a long-term strategy for the procurement and use of FMSs. We have already considered the UK's preferences for ground-based processing, and the use of strain gauge-based direct monitoring systems. Other key features are summarised as follows:

Commonality. The benefits of commonality across fleets has long been recognised, and a firm FMS development/procurement strategy will attempt to standardise wherever possible. Such standardisation offers benefits in terms of costs, procurement timescales and configuration control. It also allows ground stations to be similarly standardised.

Data Acquisition. Any review of modern monitoring systems, as used especially in flight testing and loads survey work, will show a wide variety of weights, costs, sizes and, of course, performance. Before consideration can be given to any optimum solution, some thought has to be given to our over-arching requirements. These can be summarised as follows:

a. Modularity. As part of the overall flexibility requirement of any system, modularity in construction is essential. Modularity can be physical, in terms of the ease with which a systems size/volume can be minimised, and then expanded to suit additional monitoring tasks. Also, modularity can apply to the recording elements of the system, and the mixture of analogue, discrete and strain gauge data channels offered by the FMS.

b. Input Channels. The ability to tailor the type and number of input channels is important. Assuming modular construction, a data acquisition module may offer, typically, 48 channels of discrete data, 32 channels of 'high level' data (conditioned, read from an independent and powered data source, such as a databus), or 8 channels of strain gauge or other 'low level' data (unconditioned and taken from a sensor that is powered by the FMS). The reduction in channels from 32 to 8 reflects the need to provide power supplies to the sensors, from within the FMS.

c. Memory. 16 Mb of solid state recording per card module is a typical figure. Much less than this would severely limit an FMS's ability to store raw data of sufficient resolution to provide levels of detail required to offer full and complete pictures of a sortie's content.

d. Acquisition Programming. A system that allows bandwidth, sample rates, datum levels etc to be varied is essential. Today's technology would allow the set-up of a FMS to be programmed via a ground-based PC, and uploaded through a portable data transfer unit.

e. Frequency Coverage. The influence of buffet and acoustic loading on airframe structure has attracted considerable attention in recent years, and any future FMS must have the facility to deal with high frequency modes of loading. Today, typical data acquisition systems have been found to be able to offer maximum sample rates of 100 000 samples/sec per channel, and an overall capacity of 500 000 samples/sec. This is sufficient to allow for the monitoring of high frequency modes of loading.

f. Exceedance Monitoring. The issue of exceedance monitoring has already been mentioned with regard to the associated airborne monitoring requirements. The operation of a non-"carefree handling" aircraft type beyond its structural limit is always possible; the facility to alert an operator, post-sortie, to excessive levels of N_Z, roll-rate, sideslip or landing sink rates, for example, is essential.

Usage Monitoring. It has long been recognised that confident operation of a fleet up to, and potentially beyond its initial service life, require a comprehensive understanding of its usage. Any future FMS would, therefore, have to offer a usage monitoring function, including a facility to measure/record control surface operation, landing gear usage and cabin pressurizations. Such data would allow the effects of role changes, stores carriage and mass increases to be quantified.

OLM. Programmes of OLM activity are needed to ensure the continued understanding of in-service loads, and the applicability of the FMS; the requirement for OLM is independent of aircraft type or the FMS employed. Any future FMS must, therefore, fill the OLM function or be sufficiently flexible to allow for expansion to cover the OLM role. The use of FMS as the embedded OLM capability offer additional flexibility in terms of timescales, and the number of aircraft potentially participating in any OLM programme.

Helicopter Systems

Recent studies into fixed-wing FMS design have concluded that direct monitoring techniques are most appropriate. However, due to the practicalities of strain gauge use on rotating components, the helicopter world has been forced to adopt more indirect methods in its search for a suitable FUMS. Artificial Neural Networks (ANNs) have been identified as one suitable technique, and it is envisaged that ANNs could be used to synthesise loads or fatigue damage using flight parametric data recorded by HUMS. Significant operating costs savings could be made through the service-aide fit of FUMS. Moreover, some of the conservative design assumptions that manufacturers have historically, been forced to make, may not be necessary.

It should be noted, however, that ANNs require training and validating data, which can only be measured using direct techniques. The use of ANNs cannot therefore be a viable option until equally viable direct monitoring methods have been developed for rotating helicopter components. Strain gauging technology is improving, and a hybrid of direct and indirect techniques may prove to be basis of any future FUMS.

MIGRATION AND DEMONSTRATION

In many ways, the work undertaken to formulate a future FMS strategy, and to identify the range of equipments capable of meeting our needs, has been relatively straightforward. The Service is now faced with the more difficult practicalities of demonstrating the effectiveness of the proposed direct monitoring, ground-based processing, Service-wide solution. The UK MOD is under continuous pressure to demonstrate quantifiable savings for all technical initiatives and programmes; FMS development is no exception. We are in the fortunate position, however, of having a number of ongoing OLM programmes that are, in themselves, evaluating individual aspects of our perceived Service-wide 'way ahead', and which will serve as effective 'demonstrator programmes'. OLM programmes on VC-10, Jaguar and Tucano should allow us to evaluate equipments, installations, data acquisition and processing techniques that would be combined under the future FMS strategy.

DATA HANDLING AND ANALYSIS

Regardless of the type of FMS that the RAF selects for service-wide use, the quantity of both processed and unprocessed data that will be generated will be very large. Hence, and in addition to the more technical aspects of system selection, some consideration has also been given to the ways in which the RAF see fatigue-related data being distributed and where/how analysis is likely to be done.

Historically, fatigue analysis has always been performed by a centralised agency and distributed to our operating station, on a monthly basis. Supplementary ad-hoc analyses have been possible, but time-consuming.

The emerging use of ground-based processing facilities has allowed the RAF to specify its ideal GSS and how it might be used. Figure 5 illustrates the data handling and processing arrangements likely to be seen when EF2000 enters service. The development of unit-level Health and Usage Centres (HUCs) will allow our flying stations to take a more active role in the monitoring of the fatigue consumption of their aircraft. Fatigue data will be more readily accessible to unit and squadron personnel. Moreover, the very presence of the HUC will emphasis the importance of fatigue monitoring, and encourage personnel involved in the support of SHM to treat the onboard system, and the data produced, with respect.

It is likely that the arrangements illustrated at Figure 5 will not be acceptable for all aircraft types/units, but it represents an ideal solution upon which we can build.

CONCLUSIONS

The RAF has long recognised the need for careful fatigue monitoring, across all fixed-wing aircraft types. Technical limitations and costs, however, have resulted in the Service's widespread dependency upon ageing equipment, supported by basic FMFs and OLM programmes; the results generated are not grossly in error, but it has been recognised that there is considerable scope for improvement.

For helicopter fleets, fatigue consumption has not been monitored. Fatigue-critical components have been designed using conservative loads and usage assumptions, and replaced as they reach a pre-set flying hour life.

A number of emerging systems have given us a taste for the capabilities that future FMSs could have. Neither FMCS nor SHM represent ideal solution to our FMS requirements, but they are capable systems from which we can hope to learn valuable lessons in terms of equipment/installation support and data handling/analysis.

HUMS offers helicopter operators the ability to manage levels of structural vibration, thus reducing fatigue consumption.

Recent studies by DERA have identified a suitable, standard FMS that could be employed for all generic aircraft types. A number of ongoing OLM programmes will serve as suitable technology demonstrators.

An equivalent helicopter FUMS is being developed, but depends upon advances in both direct and indirect monitoring techniques.

Care needs to be taken in defining analysis and handling techniques for the enormous quantities of data that will be generated by whichever FMS is selected. The need to present readily usable data to the agencies that really need it is paramount.

FIGURE 1 - NEGRETTI & ZAMBRA COUNTING ACCELEROMETER

JAGUAR GR MK 1 - FATIGUE METER FORMULA FOR A/C WITH EITHER PHIMAT OR CBLS ON WING O/B, CLEAN I/B - B STANDARD

Fatigue Index (FI) = $K_2 * S_1 * (2.31A+0.03B+0.001C+0.001D+0.28E+3.43F+10.36G+8.63H+1.16WL)$

K_2 = MSM Coefficient
S_1 = Stores Configuration Coefficient.
WL = Landing Coefficient.
A to H = Fatigue Meter Readings (A = -1.5g window, H = 8g window).

FIGURE 2 - EXAMPLE FATIGUE METER FORMULA

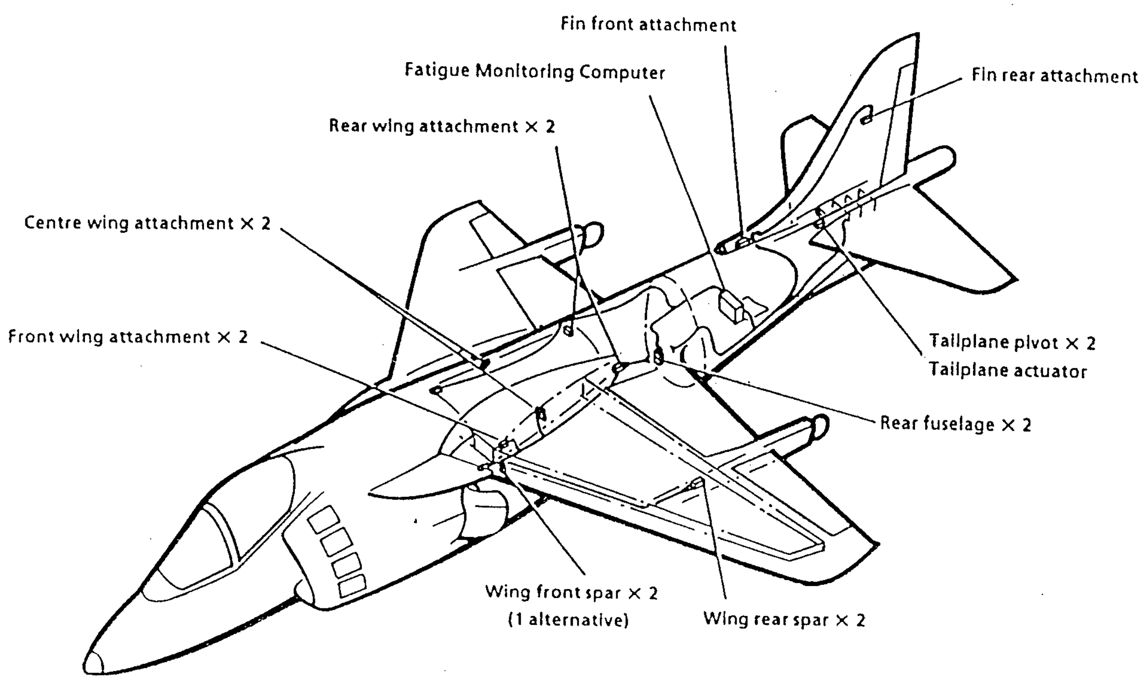


FIGURE 3A - FMCS STRAIN GAUGE LAYOUT

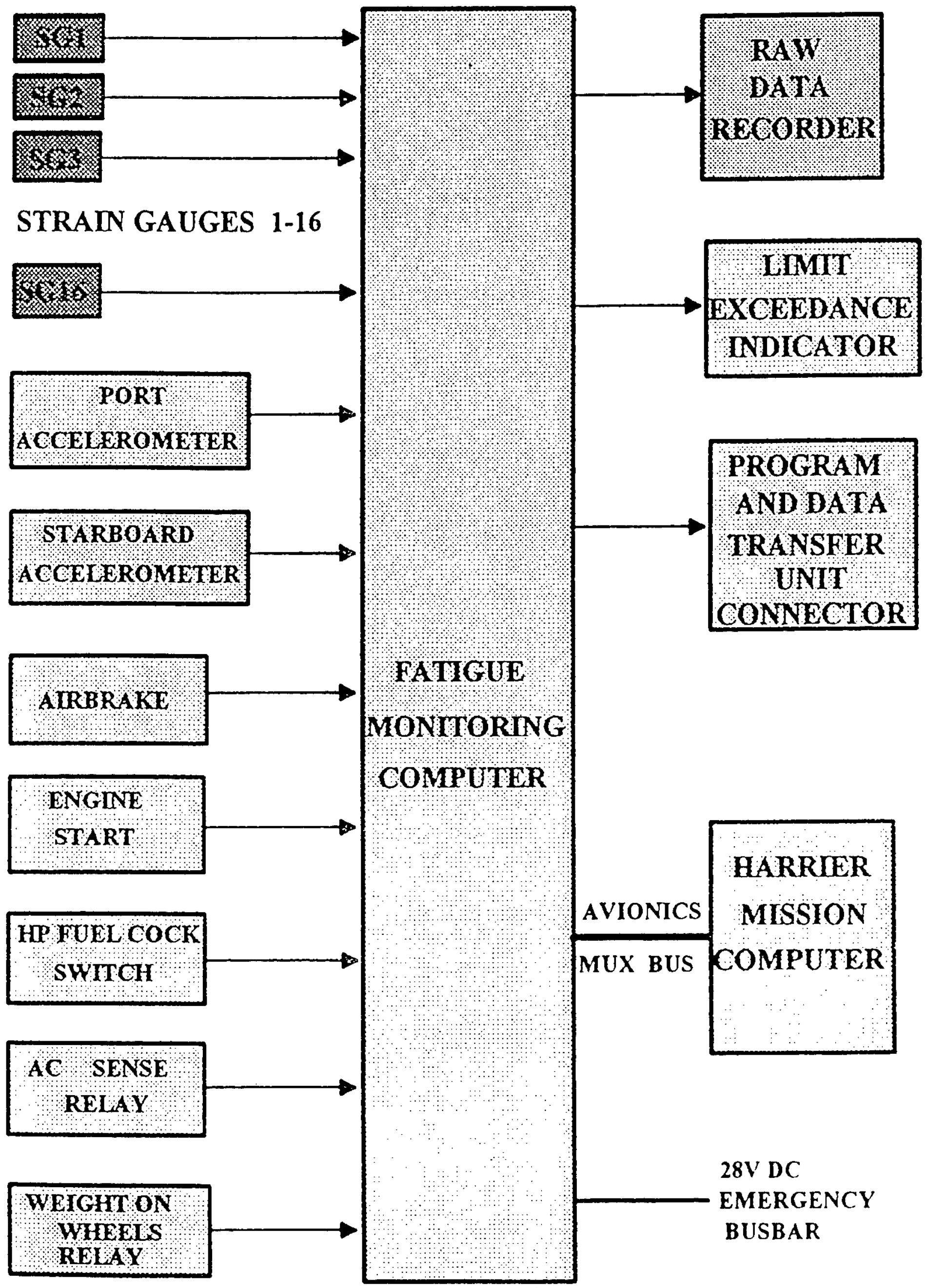


FIGURE 3B - FMCS SCHEMATIC

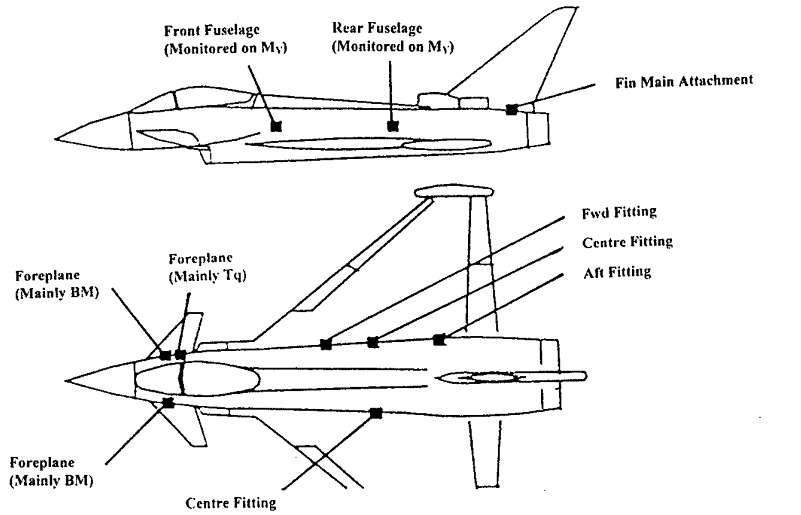


FIGURE 4A - SHM MONITORED LOCATIONS - BASELINE FIT

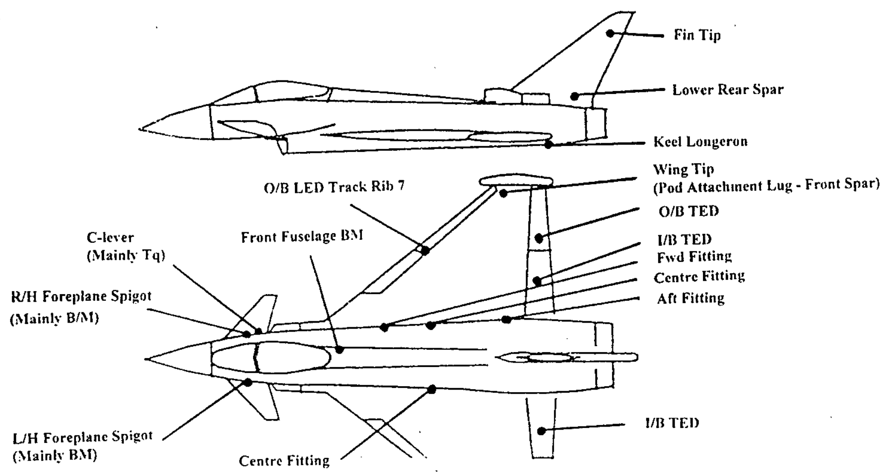


FIGURE 4B - SHM MONITORED LOCATIONS - NATIONAL FIT

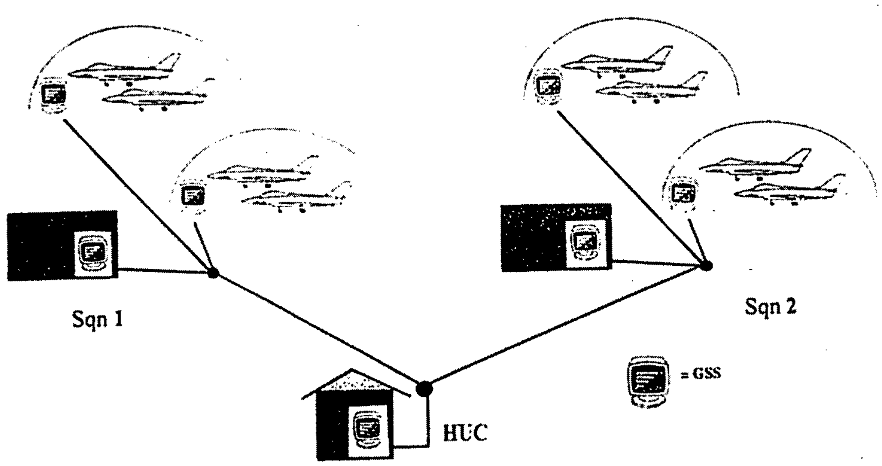


FIGURE 5 - EF2000 - GSS AND HUC DISPOSITION

CELERIS AEROSPACE CANADA INC.

TECHNICAL DATA MANAGEMENT - AN ESSENTIAL TOOL FOR EFFECTIVE LIFE CYCLE MANAGEMENT

Stephen R. Hall and John. W. R. Miner
Celeris Aerospace Canada Inc.
Suite 300, 255 Centrum Blvd
Ottawa, Ontario K1E 3V8, Canada

Integrated Aerospace & Informatics Consulting

CELERIS AEROSPACE CANADA INC.

OVERVIEW

- Economic Realities
  - The "Need for Exploitation"
- Tools of Exploitation
  - OLM Programs: Their Successes and Failures
- The Neglected Partner: Technical Data Management
  - The Costs of Neglect
  - Internet/Intranet: Friend or Foe?
- The Solution: An Integrated Approach
  - Total Data Integrity Initiative (TDI^2)
  - Loads Monitoring Central
- Conclusions

Integrated Aerospace & Informatics Consulting

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

NEED FOR EXPLOITATION

- Aging Aircraft Fleets
  - Need to extend the life far beyond the originally anticipated design life
  - Longer lives will invariably result in aircraft being used in a role for which they were not designed
- New Aircraft
  - "Day-One" efficient management of an expensive resource
- Focus
  - Safe and Economic Life-Cycle Management
    - Technology is a tool to achieve this

CELERIS AEROSPACE CANADA INC.

TOOLS OF EXPLOITATION

- Multi-Channel OLM Recorders
  - High Sampling and Data Storage Capabilities
    - Obtain lots of data
  - Ability to extract data from the Aircraft Bus
    - Obtain lots of data
  - Ability to Monitor a variety of stand-alone sensors
    - Obtain lots of data
  - Additional boards (Channels) can be easily added
    - Obtain lots of data
  - Interface with high speed data download techniques
    - Obtain lots of data

Integrated Aerospace & Informatics Consulting

ELERIS
AEROSPACE
ANADA INC.

SUCCESS OR FAILURE

- Engineering Aspects are generally well addressed
  - OLM Recorder
  - Sensors
  - Integration with "Aircraft System
- Data-Management Aspects are generally poorly addressed
  - Treated as an adjunct to the "technical" parts of the program
    - Assumed that data -management will take care of itself
    - Addressed once the "technical aspects have been resolved

Integrated Aerospace & Informatics Consulting

ELERIS
AEROSPACE
ANADA INC.

CAUSES OF FAILURE

- Review of "Failed Programs"
  - Estimate in access of 90% were due to ill-defined or inadequate data management programs
- Data Management Perspective
  - Poorly understood
  - Not addressed in the context of the objective of the OLM Program
    - Safe and cost-effective management of an expensive resource: Life Cycle Management

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

TECHNICAL DATA MANAGEMENT

- Production of valid data upon which informed decisions can be made in a cost-effective and timely manner
  - For reduced life-cycle costs context has to be in terms of overall Life-Cycle Management
- The successful implementation of the engineering aspects of an OLM program is a necessary, but not sufficient condition for successful aircraft Life-Cycle Management

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

PROGRAM COST DISTRIBUTION

- Greatest Costs associated <u>over the life of a program</u> are the data acquisition and processing costs and the cost of not having or being able to access reliable data when it is needed
  - Cost of acquisition, validation, processing, storage and analysis
- Greatest effort is usually placed in the acquisition and integration of the OLM system
  - Significant up front cost
  - Falls into the Engineers Comfort Zone
    - "See, Touch and Feel Aspect"

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

THE COST OF DATA ACQUISITION

- Data is a valuable commodity, i.e.:
  - It is costly to acquire
  - Significant costs can be incurred if the data has been compromised or is not available when life-cycle management decisions have to be made
- Protection of Investment makes it essential to have robust procedures for:
  - Data validation
  - Data security

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

SYMPTOMS OF FAILURE

- "Let's Monitor Everything Syndrome"
  - Capability to acquire data cannot replace the need for technical understanding or clearly defined objectives
- Data Isolation Syndrome
  - The data obtained is treated in a fragmentary manner during acquisition and processing
    - Downloaded data
    - Validated data
    - Processed data

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

SYMPTOMS OF FAILURE (cont)

- "But it worked on my PC Syndrome"
  - Validation/Simulation of both the engineering and data management aspects is essential
- "But we already have development environment software Syndrome"
  - Let requirements drive the system, not the system the requirements

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

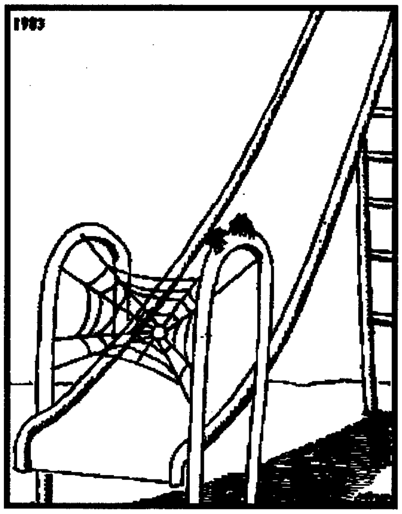
LETS MONITOR EVERYTHING?

- While the incremental costs of adding additional channels/sensors is relatively small, the impact on the data processing costs may be significant
  - Acquisition
  - Validation
  - Processing, Analysis and Reconciliation
  - Storage
- How will the acquisition of this parameter assist with existing or future LCM Objectives?

Integrated Aerospace & Informatics Consulting

CELERIS AEROSPACE CANADA INC.

REASONS FOR MONITORING



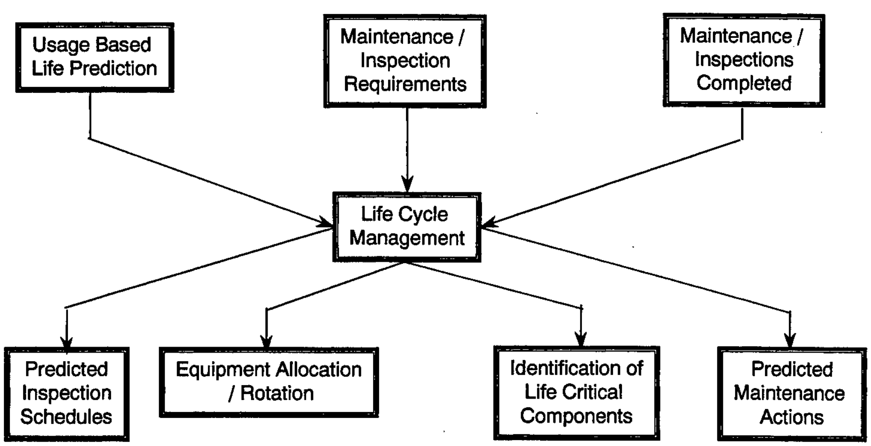
"If we pull this off we will live like kings"

- There are a number of valid reasons for OLM programs
  - Operational Problems
  - Life Extension
  - Fleet Management
- Make sure you know what yours are
  - Clearly defined objectives from day one
- OLM programs will not solve all your problems

Integrated Aerospace & Informatics Consulting

CELERIS AEROSPACE CANADA INC.

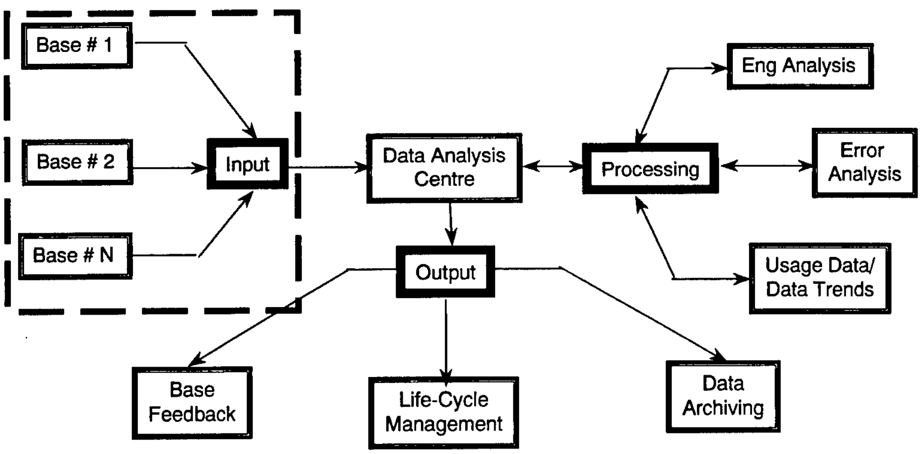
DATA ISOLATION



Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

PROCESSING CONCEPT



Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

AN HOLISTIC APPROACH

- Provides an understanding of both the engineering and data management aspects program
  - Isolates areas of risk
  - Smooth transition of data between facilities
- Decisions are made in the context of the ultimate aims
  - Life-Cycle Management
  - Make informed decisions
- Data-Management considerations have to be integrated with engineering considerations
  - Combined development teams

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

DATA-MODELING TECHNIQUES

- Used by the software industry for years
  - As a means of defining data interaction, processing and storage requirements
- They are also a good engineering validation tool and facilitate
  - Development of well defined system objectives
  - Identification of critical components
  - The development of a phased approach to system development
- Remove the “Black-Art” from the development of effective data management systems

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

SOURCES OF DATA ERROR

- Hardware
  - Faulty Recorders and or Sensors
  - Sensor installation problems
  - Incorrect recorder initialization procedures
- Software
  - Incorrect data downloading and/or transcription
  - Incorrect configuration tracking
    - Universal implementation of fleet-wide modifications
    - Inappropriate application of Fill-in data

Integrated Aerospace & Informatics Consulting

TYPES OF ERROR

- Two general classifications
  - Logical Errors - Errors that can easily be identified as right or wrong
    - Range checks
    - Event response frequency
  - Potential Errors - Errors which only become apparent over time and/or require detailed analysis by skilled personnel
    - Strain gauge drift

CELERIS AEROSPACE CANADA INC.

PRINCIPLES OF DATA PROTECTION

- Regular data down-loading (approx monthly), even if the recorder is not full
- Based on the assumption that the majority of errors are Logical Errors
  - Logical Errors should be detected as close to source as possible
  - Appropriate remedial action should be implemented as soon as possible
- Aim is to minimize the risk of the acquisition of substantial amounts of invalid data

Integrated Aerospace & Informatics Consulting

CELERIS AEROSPACE CANADA INC.

DATA PROTECTION CONSTRAINTS

- OLM acquisition is generally a secondary duty and therefore should have minimal impact on day-to-day operations
- Clear rectification instructions should be provided to operational personnel
- Data entry associated with rectification actions and error tracking should be minimal

Integrated Aerospace & Informatics Consulting

CELERIS AEROSPACE CANADA INC.

DATA PROCESSING TRADE-OFFS

- Onboard versus Post-Flight Processing
  - "Onboard" is processing of the data by the recorder such that the raw data is discarded
  - "Post-Flight" is any processing which is based on raw data to which there is still access.
- Issue is Data Transparency
  - Solution is somewhere between total onboard processing and no onboard processing
  - The decision is a trade-off: reduced data-volume versus:
    - impact of not being able to reconstruct anomalous events
    - risk of failure to identify errors

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

"BUT IT WORKED ON MY PC"

- Prototype programs for two/three aircraft)
  - Can be implemented on a stand-alone PC
  - Use "standard" PC Spreadsheet / Database software tools (Lotus, Excel, QuatroPro, Paradox etc..) environment
  - Emphasis on Graphic Display of individual flights and individual parameters
    - Useful but again staying in the engineering comfort zone

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

BUT IT WORKED ON MY PC (cont.)

- "Data-volume" transition where these tools are no longer adequate
  - Data management/manipulation problems
  - Throughput slow
  - Need discriminators to identify where to look
    - Even limited hand-cranking becomes impractical
  - Data-security /integrity becomes a significant problem
- Bottom-Line:
  - System that has been "proven" based only on a limited trial basis (two/three aircraft) may well prove to be inadequate for larger fleets

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

APPLICATION SOFTWARE

- Many 4GL languages and Database systems
  - Even comprehensive and expensive development environments such as ORACLE cost in the range of C$10K - C$20K
  - Context of implementing a program to effectively manage fleets of aircraft worth millions dollars, these costs are negligible
- Let the requirements dictate the application software not the other way around!
  - Use of existing resources is a consideration not a driver

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

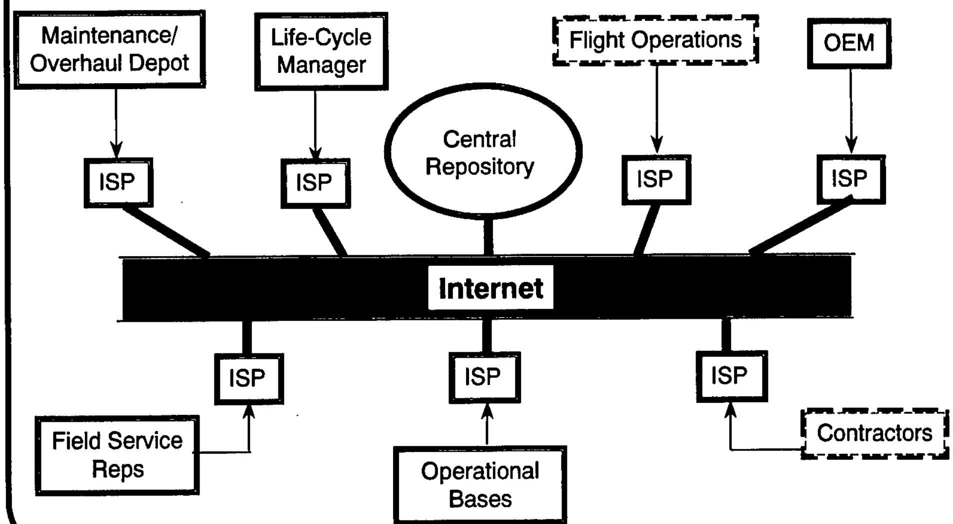
USE OF THE INTERNET

- Key factor in effective data-management is the ability of different users to readily access appropriate views of the data
  - Data itself may be distributed or centralized
  - User Community may be distributed or centralized
- Use dedicated networks (Intranets)
  - Costly to implement and run
- Internet
  - Readily available, but is it secure?

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

COMMUNICATION POTENTIAL



Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

SECURITY ISSUES

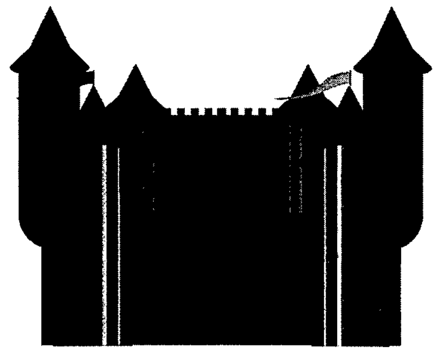
- Security of Data
  - May be a problem
  - Often related to the inconvenience of losing or corrupting the data
    - Back-up
- Security of Access
  - Major problem, particularly regarding access on to secure internal networks (Intranets)
- Virus Proliferation
  - Same as for other transportable media
  - Should have a virus-scan policy

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

INTERNET - HOW SECURE?

- Contrary to common belief can be made very secure
  - Trade Cost of Security vs Risk or Consequences of losing data
  - Graduated Security
    - Analogous to Physical Plant Security
- Zero-use is a solution
  - Costly
  - Significant loss of benefit
  - Is the trade-off worth it?



Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

SOLUTIONS FOR SUCCESS

- Total Data Integrity Initiative (TDI^2)
  - Cradle to grave approach based on the end point being cost effective Life-Cycle Management
- Pooling of knowledge
  - Implementation of Loads Monitoring Central on the World-Wide Web

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

TOTAL DATA INTEGRITY INITIATIVE

- To fully exploit Structural Loads/Health data for reduced Life-Cycle costs , the Total Data Integrity Initiative (TDI^2) has been developed
- Based on an Integrated Approach to OLM
- Programs coupled with an awareness of Life-Cycle Management requirements
  - Cooperative Venture involving
    - Recorder Manufacturers
    - Installation Contractors
    - Engineering and LCM organizations
    - Technical Data Management organizations

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

WHAT DOES TDI^2 PROVIDE ?

- High quality, validated data that can be interfaced with any Data Analysis Centre
  - Validates the data from both a system and engineering perspective.
  - Rapid error isolation and rectification
  - Seamless Interfaces with other Client Life-Cycle Management Applications
    - New Applications
    - Protection of Investment in Existing Applications

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

TDI² IMPLEMENTATION

- Multi-Fleet Application
- Program Definition
- Instrumentation/Installation
- Data Validation, Analysis and Management
- Provision of an interface to overall LCM systems

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

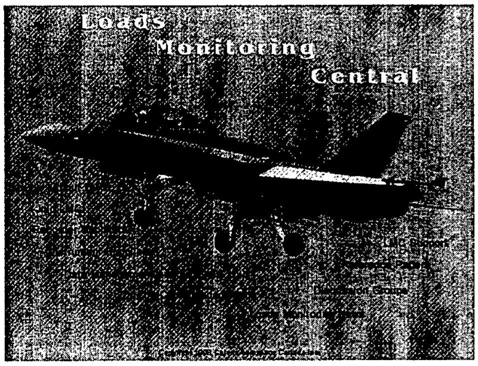
TDI² - RESULTS

- Improved data which can actually be used for LCM decisions
- "One-Stop Shopping" for rectification of problems
  - Single contact to resolve problems
- Rapid, Cost-Effective Support
  - Remote Support and/or Internet
  - Minimal expertise required at base-level
- Access to Independent Validation and Verification (IV&V) if required

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

LOADS MONITORING CENTRAL



- Central location for OLM Programs on the World-Wide Web
  - Overall Program Design
  - Recorder/Instrumentation Issues
  - Engineering Issues
  - Data Management Issues
  - LCM Issues
- To be launched in Fall 1998
  - In conjunction with several partners

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE CANADA INC.

CONCLUSIONS

- A disciplined approach to Technical Data Management is essential if Structural Loads/Health Data is to be fully exploited
  - It cannot be an adjunct or afterthought
- Successful programs have to be implemented in the context of Life-Cycle Management
  - The engineering is a tool to get there
- Initiatives are being implemented to ensure maximum benefit is derived from the data that is collected
  - Total Data Integrity Initiative (TDI^2)
  - Loads Monitoring Central

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

ACKNOWLEDGEMENT

- Encouragement and support to pursue this work has been provided in part by
  - Canadian Department of National Defence
  - Institute for Aerospace Research of National Research Council of Canada
- The Authors would like to particularly note the support and encouragement of:
  - Mr Dave Simpson, NRCC/IAR

Integrated Aerospace & Informatics Consulting

ELERIS AEROSPACE ANADA INC.

CONTACTING CELERIS

- Address

  Suite 300, 255 Centrum Blvd
  Orleans, Ontario
  CANADA, K1E 3V8

  Tel: (613) 837-1161
  FAX: (613) 834-6420
- Internet

  Steve Hall - halls@celeris.ca
  John Miner - minerj@celeris.ca
  Webpage - http://www.celeris.ca

- Download this presentation
  - Anonymous FTP
    - 206.222.85.19
    - /papers
  - Download
    - AGARD02.PPT or
    - AGARD02.ZIP

Integrated Aerospace & Informatics Consulting

CF-188 FATIGUE LIFE MANAGEMENT PROGRAM

by

Capt. Y. Caron
Directorate Technical Airworthness
National Defence Headquarters
Mgen R. Peakes Building
Ottawa, Ontario K1A 0K2 Canada

Yves Richard
ASIP Section Chief
Bombardier Services
10,000 Cargo A-4 Street
Mirabel, Québec J7N 1H3 Canada

SUMMARY

The Canadian CF-188 Fatigue Life Management Program (FLMP) has been in place for over 10 years. This program has been successful in controlling fatigue accumulation on the CF-188 airframe. This had a major impact on the life cycle management cost of the aircraft. The solutions to the challenges encountered operating this program will be discussed. Lessons learned are summarized.

1.0 INTRODUCTION

The Canadian Forces (CF) operate the McDonnell Douglas CF-188 aircraft since 1982. The aircraft structural fatigue life is tracked using data from the onboard recording system. The aircraft fatigue life data is used to:

- Help operators maximize the airframe life;
- Assist in the planning of the inspection, repair and modification programs; and
- Ensure that the fleet life is maximized while maintaining operational effectiveness.

This paper describes how the CF-188 on-board fatigue tracking system is used by the CF for its Fatigue Life Management Program (FLMP), along with the challenges that were faced and how they were handled. Finally, a series of lessons learned are provided for future use.

2.0 BENEFITS OF THE CF-188 FATIGUE TRACKING SYSTEM

The CF-188 onboard fatigue tracking system allows the aircraft life cycle managers and operators to manage the overall airframe life. This is very important for the CF-188, since the airframe life is based on accumulated fatigue rather than flying hours. Adequate use of the fatigue tracking data can allow the operators to better control the fatigue life consumption of each individual aircraft as well as properly estimate the aircraft retirement time. It also permits efficient management of the aircraft maintenance program.

Early in the CF-188 program, the fatigue tracking data helped to identify that the aircraft in-service usage was more severe than anticipated at the design stage. A FLMP was implemented in 1987 and has allowed the CF to overcome this high fatigue life consumption and extend the life of the Canadian fleet by 12 years (See Figure 1). This program is based on usage characterization, fatigue awareness training and guidance and control for the operational community.

The use of the fatigue tracking data also allows the pilots to better understand the impact of different flight techniques on the aircraft fatigue life and make adjustments to reduce the fatigue damage without compromising the operational capabilities. For example, during peacetime, delaying high "g" maneuvers until external fuel tanks are empty can significantly reduce fatigue damage in the long run.

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

As shown in Figure 2, the CF-188 would need to be retired earlier if no FLMP was in place. With the current defense budget restrictions, the need to maximize the CF-188 operational life further reinforces the need for a FLMP. The implementation of the FLMP and an optimum use of the aircraft should allow increasing the fleet life by at least 12 years. Reference 2 mentions that this could represent savings of up to $400 Million for the CF.

In addition, a strong FLMP allows proper control and scheduling of the structural maintenance. All the major modifications to be applied to the CF-188 structure, resulting from both the in-service failures and the IFOSTP (see section 3.2.1.1) and OEM tests, have been given a fatigue threshold and included in a maintenance model. This, combined with the fatigue tracking of every aircraft, allows a prediction of how many aircraft are to be modified and to forecast their introduction into the third line facilities. This gives a complete picture, from now until planned fleet retirement, of the resources required to maintain the fleet (See Figure 3). Using this model, a reduction of 10% of the FLEI expenditures could represent savings of about $20 Million in life cycle costs. This goes to show that the accuracy of the fatigue tracking is critical, and that every effort must be done to eliminate errors and undue conservatism. This can help put in perspective the relative importance of the efforts described below.

3.0 CF-188 FATIGUE TRACKING AND FLEET MANAGEMENT CHALLENGES

The following sections describe the various problems encountered with the CF-188 FLMP (in terms of data acquisition and analysis), and present the solutions that were implemented or/and currently under development. Fleet management issues and operator feedback issues are also reviewed to complete the full cycle of the FLMP.

3.1 CF-188 FATIGUE DATA ACQUISITION

The relevant aircraft flight data for the CF-188 are obtained from its Maintenance Signal Data Recording System (MSDRS). This system is mainly used for maintenance and trouble shooting purposes, but doubles as a flight data recorder for tracking purposes. The tracking data collected includes:

a) Aircraft Weapon Inventory;

b) Aircraft Fuel Consumption;

c) Aircraft Flight Parameters;
   - ⇒ Nz acceleration;
   - ⇒ Altitude;
   - ⇒ Speed;
   - ⇒ Roll Rate;
   - ⇒ Angle of Attack; and
   - ⇒ Control Surface Deflection

a) Strain sensor readings;

b) Mission Data (Date, Mission Code, Pilot Identification Code)

Seven fatigue strain sensors (Figure 4) are strategically located on the aircraft structure. These locations are:

- FS470 Wing Root Attachment lug;
- Wing Fold;
- Stabilator Spindle Block, Left and Right;
- FS598 Vertical Attachment Stub, Left and Right; and
- Nose Landing Gear Drag Brace Support.

In case of primary fatigue sensor, a back-up sensor can be manually connected.

3.1.1 Problems Encountered

Various problems were encountered in the acquisition of CF-188 Fatigue Data. The impacts of these problems on the fatigue life evaluation, as well as the proposed solutions are described below.

3.1.1.1 Strain Sensors Signal Not Calibrated

The use of measured strains in the fatigue damage calculation allows for good accuracy because flight parameters such as weight, configuration, velocity, altitude and others are inherently taken into account. However, it has been shown that the strain sensors installed at the Wing Root, Horizontal and Vertical tails react differently from aircraft to aircraft. Differences of up to 35% from nominal values were observed at the Wing Root sensor. The usage measured at this sensor is compared to the fatigue damage applied on the aircraft full-scale test to obtain the damage accumulated in relation to the test. Considering that it is the primary sensor for fatigue life evaluation, and that a 15% difference in the measured strains can mean a 50% in the fatigue life, it is immediately obvious that the sensors must be calibrated for meaningful tracking. The strain sensors installed at manufacturing were not individually calibrated.

For the Wing Root, the calibration problem is resolved by performing in-flight calibration and analytical evaluation as required. The variation in strain response at the Wing Root location is due to gradual changes in local deformations of the Wing Root lower lug bushing. This changes the friction between the lug bushing and the pin holding the wing in place, resulting in a change of strain response with time. The calibration, either in-flight or analytical, allows correction the strain reading to correct differences between aircraft. Analytical calibration is performed by identifying, through the tracking software, maneuvers that have known wing root bending moment values and comparing the strain outputs to nominal values. Adjustments can then be made to the rest of the strain readings. After this adjustment is done once, it can vary with time. This is verified by periodical in-flight calibrations, every 300 to 600 flight hours or any time the strain sensors are changed.

Calibrations flights are a maintenance burden to the operational community. Flight time and associated maintenance time, not to mention fatigue expended during the calibration flights, are reasons why the CF are currently looking at using a Parametric Load Formulation (PLF) approach to predict the load at the wing root location. It uses data from the flight instruments and other sensors on the aircraft to provide flight parameters and loads formulation established through the International Follow-on Structural Test Program (IFOSTP). This method is not sensitive to local loading problems and should be more reliable, as well as giving comparable accuracy to strain-based tracking. It would eliminate the expensive maintenance of the strain sensors and the calibration flights, since it uses sensors that are already on the aircraft and are calibrated through normal systems maintenance.

3.1.1.2 Strain Sensor Sampling Rate Too Low

The strain sensors on the CF-188 are currently sampled at 10 Hz. This limitation is due to recorder storage capacity. Although the sampling rate is sufficient in the case of maneuver loads, it is totally inadequate to record dynamic and buffet loads. This is the case for the strain sensors at the Vertical tail (dynamic) and wing fold (buffet). It was therefore necessary to establish other methodologies to track those locations. Recent studies and in-service failures on the CF-188 fleet have shown that fatigue usage can be underestimated by a factor of up to 20 times if dynamic loads are ignored.

The ability to use existing data to retrace the history of the fleet is also required since the CF fleet has been in service since 1982. A very accurate tracking system implemented at mid-life is useless if the expended life until now can only be very roughly estimated.

There are various ways to resolve the problem of buffet loads recording. First, a direct (strain-based) approach can be used. This method is theoretically more accurate since there are no errors due to analytical estimates or simplifications. However, in order to apply it, data sampling rates in the order of 600 Hz are required. This approach also requires intensive sensor maintenance (calibration, validation, drift correction, data fill-in when data is lost, initial value, replacing damaged or unserviceable sensors, etc) which can be very expensive. For this reason this method is not being considered for the CF-188.

The second method which could be used is a simple Angle of Attack/Dynamic Pressure Tracking /Buffet Usage Index (BUI). It consists of tracking the time spent in every AOA-Dynamic Pressure "bin" for AOA above 10 degrees. This approach is limited, but it does provide a rough estimate of the relative severity between various aircraft and full-scale tests, allowing aircraft selection for maintenance.

The last method considered using a method based on Parametric Load Formulation to define the Maneuver Loads and a set of Peak-Valley (P-V) Loads (based on AOA and the dynamic pressure) for the Buffet loads. If the AOA is greater than 10°, the dynamic component of the load is read from a database of dynamic cycles associated with given maneuver loads and then superposed upon the maneuver P-V file. The resulting P-V file is then evaluated for fatigue damage using the SLMP software. The method provides a good tool for the operators to control the damage induced on these areas subject to buffeting. It is also compatible with the historical CF-188 data, allowing a determination of the fatigue history of the components. This method has been selected for the CF-188, but some work remains to be done before implementation. Many of its processes were used for full-scale fatigue test spectrum generation but rely on much human intervention. This makes the current process cumbersome and unsuited to continuous fleet tracking. Automatic maneuver recognition and assignment of dynamic damage remains to be implemented. Finally, the database of dynamic damage must be built from the IFOSTP and other test data.

3.1.1.3 Strain Sensor Triggering Based on Maximum Nz Rather Than Load

In the development of the aircraft it was assumed by the aircraft manufacturer that the strain measured (and the wing bending moment) would peak at the same time as the Normal Acceleration (Nz). For this reason, it was decided that the usage at the center fuselage critical location, driven by the Wing Root Bending Moment, could be tracked using only the wing root strain measured at the Nz Peak/Valley triggers.

Recent studies have shown this assumption to be invalid and that on average, this had the effect of underestimating the fatigue life by more than 10%. The difference in the timing of peak Wing Root strain relative to peak Nz is most significant for abrupt maneuvers. For these types of maneuvers, the peak Bending Moment does not occur at the same time as the peak Nz.

This problem was resolved in two ways. First, the data from all seven strain sensors is recorded every time a sensor is triggered (i.e. when the vertical tail sensor triggers, for example, a record is made of the reading of all the other sensors). As an interim fix, it is possible to reconstruct a new Peak and Valley sequence for the Wing Root sensor. However, this method provides accurate results only when sufficient data is available and is therefore intolerant of unserviceable sensors. The second method was to add triggering of the Wing Root sensor based on the strain in addition to the Nz trigger. This last fix has recently been implemented on the CF-188 through a mission computer software change.

3.1.1.4 Strain Sensors Reliability Lower Than Expected

The CF-188 strain sensors were designed to last the full life of the CF-188 (6000 hours). However for various reasons, most of these sensors have a Mean Time Between Failure (MTBF) of no more than 1500 hours. This has the effect, when not changed in a timely manner, to create a significant amount of data loss which until 1995 was very conservatively replaced, leading to lost fatigue potential (as much as 50% for the affected period). Maintenance effort must be expended to limit the time between of the loss of a sensor and its replacement. As zero data loss is not possible, a parametric loads approach was developed to replace the bad sensor data with a more accurate approximation (within 15%) and eliminate the undue conservatism of previous fill-in methods.

The strain sensor reliability problem is mainly related to the sensor bonding process. The aging of the connectors, as well as some of the recording equipment, are also creating maintenance problems. Various improvements have been implemented in the last few years but did not fully resolve the problem. As mentioned in paragraph 3.1.1.1, the CF are currently looking at using an advanced PLF technique when possible to reduce the maintenance costs associated with the strain sensors replacement.

3.1.1.5 Recorder Storage Capacity Too Small

The CF-188 MSDRS system is based on 1970s technology, with limited storage capabilities. The system available on-board the Canadian CF-188 is limited to about 300Kbytes of information, representing about 5 flights. Since the downloading of the data storage cartridge is maintenance intensive, the CF only remove it when it is full. On average, the CF lose 10% of the data due to memory limitations and tape downloading operations. This is important considering that the replacement of this data is always more conservative, even with the improved methods, than the original data.

The limited storage capacity also creates problem in terms of the amount of fatigue data it can capture. Priority, in term of space, is not necessarily given to fatigue tracking data since the MSDRS system is primarily a trouble-shooting system. This has the effect of reducing the quantity and precision of the data available. For example, the current system limits the number of strain sensor readings and the frequency of sensors sampling (10 Hz) and other parameters such as the flight control positions (0.2 Hz).

A two-phase solution is being undertaken to eliminate this problem. The first phase involved downloading the data more often, (which is currently being done) and the second one was to replace the existing storage unit by a solid state storage unit. This unit would have 10 times the storage capacity of the existing one and being solid state, it would have a higher reliability and a less maintenance-intensive downloading process. This is slated for implementation in the next 2 years.

3.1.1.6 Recorder Setting Modifications Time-Consuming and Costly

The CF-188 fatigue tracking system is not very flexible. For example, recorder settings such as the data filtration (strain sensor rise-fall and deadband) parameters and sensor sampling rate can only be modified through a mission computer software upgrade. These changes are time-consuming and costly since the mission computer software is a flight critical item. In addition, the fatigue tracking related mission computer changes have low priority for implementation. An independent, dedicated fatigue tracking system could eliminate this problem. This system has yet to be defined, depending on the tracking options that are finally selected.

3.1.1.7 Inaccuracies in the Mission Data

The mission information attached to each flight is very important in the CF-188 FLMP. Accuracy of information such as flight data, mission type, flight duration and pilot identification is crucial to the operator to properly assess the impact of a mission in terms of fatigue damage. On the CF-188, there are two different ways of providing this information. First, the pilot can enter the information in the aircraft mission computer before each flight. The accuracy of this information is however questionable for the following reasons:

a) Only a two digit field is available to enter each of the following: the date, mission type and pilot id;

b) The pilot often does not remember the information to be entered or will even forget to enter the information due to associated flight preparation workload; and

c) The information entered will be in error due to changing weather conditions or encountered threat, which force a change in mission type after departure from the base.

Second, the information is created electronically after the flight through the maintenance recording system (daily aircraft records). The tracking system information is then available to be reconciled with the proper maintenance records at the time of processing. This data reconciliation is yet to be fully automated and is not 100% successful, but currently provides the most accurate information. Efforts are underway to combine the current automation of daily aircraft records with the fatigue tracking reconciliation process to provide higher accuracy data to the operators.

3.2 CF-188 FATIGUE DATA ANALYSIS

The data acquisition is not the only challenge of the CF-188 fatigue tracking system. After obtaining the data, it must be analyzed properly. The analysis of the fatigue tracking data is done using software originally developed by the manufacturer of the aircraft. The function of this software is to derive the fatigue life of each aircraft based on the tracking data collected on-board the aircraft. A Canadian version of this software called Structural Life Monitoring Program (SLMP) is used to obtain the individual aircraft damage accumulation on a weekly basis. SLMP expresses damage accumulation in terms of Fatigue Life Expended Index (FLEI). The FLEI represents the amount of fatigue damage an individual airframe has accumulated as a fraction of the total life, which is 6000 flying hours of the design usage spectrum. This linear relationship was established using the information during the F/A-18 Full Scale Test conducted by the manufacturer.

Reference 1 provides a detailed explanation of the methodology for the calculation of the FLEI values. However, it is important to briefly discuss the general principles involved in the calculation of this index in order to provide a better insight into the problems that will be presented later. For the purpose of the fatigue calculations, crack initiation was defined as formation of a crack of 0.01 inches. 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.

The damage calculation has to account for the local plastic deformation which will occur around the areas of high stress concentration such as bolt or rivet holes. These areas are considered as material notches for the purposed of the analysis. To obtain the notch stress spectrum from the stress spectrum, Neuber's rule is applied. Figure 5 shows the concept of the determination of the crack initiation life once the notch stress is obtained. The hysteresis curve, which is unique to each material, is used to find the equivalent strain. The equivalent strains are used to obtain the amount of damage per cycle and the linear sum of this damage per cycle gives the crack initiation life.

3.2.1 Problems encountered

3.2.1.1 Reference Stress Level

It is important that design usage used to certify the aircraft be representative of the aircraft operational usage. For the CF-188, the design usage was established by considering the loads at three specific points in the sky, chosen to represent the maximum loading conditions on the aircraft. It was demonstrated early in the life of CF fleet that the CF-188 had a very different type of usage from the design specification. This created problems of interpretation and certification for our fleet.

For those reasons (and others) the CF initiated the International Follow-on Structural Test Program (IFOSTP). This program included the generation of a test spectrum based on representative aircraft usage. Based on this spectrum, it is essential to establish new referencing methodology to obtain FLEIs related to the IFOSTP Test.

The reference stress is the peak spectrum stress required to obtain the test target life. This stress is then used to establish an individual aircraft usage by multiplying it by the ratio of actual over maximum load as derived from each strain readings. This percentage of loads is obtained by dividing the strain by the location reference strain (Strain at 100% of load) (See Figure 6).

The reference stress is a key parameter to derive the FLEI of a critical location. It must be of realistic amplitude in order to have the fatigue model react in a realistic manner. In the case of the CF-188, the only strain sensor available to track the most critical locations on the aircraft was the Y470 wing root sensor, which is installed on a titanium wing lug. Although the most critical locations for the aircraft are made of aluminum, it was decided to use the titanium material properties for tracking purposes. Since that particular part had a very long fatigue life, meeting the intended fatigue target required that a reference Stress of 160 ksi be used. As shown in Figure 7, this made the model react in a very unrealistic manner: the fatigue life becomes longer as the load level increases. For aircraft flying a more benign usage, this could increase their FLEI, forcing unnecessary maintenance. High usage had the reverse effect (see next section). It was therefore decided to use aluminum material properties, which allowed realistic reference stresses to track the center fuselage. Complete reprocessing of the fleet fatigue data was required to rectify this problem. This was a major effort that has been going on for the last 2 years.

3.2.1.2 Effects of Overloads

As discussed earlier, the methodology used for the CF-188 has proved to be very sensitive to high loads. Many aircraft in the fleet experienced this kind of condition and had an unrealistically low FLEI. The methodology was fully reviewed and it was demonstrated that is valid most of the time but when unrealistically high stress levels are encountered, it becomes unconservative. It was also noted that some aircraft were showing unrealistically high stresses due to data spikes in the recorder early in the life of the aircraft. For example, one aircraft recorded strain values, after only 10 hours of operation, that were twice the maximum design values, which the recording software did not reject. This had the effect of totally stopping the damage accumulation for this aircraft, thereby delaying necessary structural maintenance.

The strains are now clipped at 100% of the design values. This eliminates any unrealistic values and is more representative of the design condition. This was also corrected during the fatigue data reprocessing.

3.3 FLEET MANAGEMENT ISSUES

3.3.1 Operator Involvement

The single most significant factor in the success or failure of a fatigue management program is the level of operator involvement. Since the inception of the program at the end of the 1980's the success in reducing the fatigue consumption at the squadrons has been variable. The graph at Figure 8 shows the variation of the FLEI consumption for both CF operating bases since 1990. The data shows significant differences in the fatigue consumption between both bases. Many physical factors can explain this difference such as:

a) terrain surrounding the base and the training areas (for low altitude missions);

b) External stores configurations flown by different units; and

c) Size of training areas and ranges, where tight turns are required to stay within the range boundaries.

However, even after looking at the above, one of the main factors is the active participation of the operators in the program. The official fatigue rate goal of the FLMP program had been set at 0.111/1000 hrs from the beginning. One of the bases was consistently achieving this target, while the other has been well above in the last 5 years.

When faced with fleet life issues, higher command sought further ways to reduce the fatigue consumption and increase the life span of the fleet. Closer examination of the detailed mission data showed that the squadrons that had the lower fatigue rates were not lacking in operational readiness. For this assessment, both the engineers and operators used data recorded and processed for the FLMP. The decision was made in the summer of 1996 to lower the fatigue target rate to 0.090/1000 hrs, a reduction of approx. 20%, and strict guidelines were promulgated to the squadron commanders for the respect of this limit. Squadrons that had the higher rates achieved the target rate within a year.

This is an extreme example, but shows that whatever efforts can be made at the engineering level must be matched by commensurate levels of information, education and involvement of the operational community. Fleet managers who concentrate solely on the technical aspects of fatigue tracking will not realize the full benefits of the program.

3.3.2 Timeliness of data returns

The usage data acquired and processed must be distributed in a timely fashion. The original FLMP called for reports to be generated monthly, quarterly and yearly, due respectively 45, 60 and 120 days following the reporting period. The monthly reports are the main tool for fleet management at the squadron and it can be seen that the original schedules allowed delays of up to 2.5 months for flights done at the beginning of the reporting period. Operators identified this as being too long for fatigue performance monitoring. Through tighter control of the data acquisition and processing times, the delay after the reporting period could be cut down to approximately 20 days, still resulting in some cases in unacceptable delays for the operators.

In order to address the problem, a system of weekly processing has been instituted whereby all the available data is processed at the end of the week and the results sent back to the squadrons by the following week. This allows the operators to have data that is at most 10 days old. This assumes, of course, that the field units have provided the data in time for processing.

3.3.3 Squadron Level Software

In order to keep up with the data coming out of the processing center, data presentation had to be modified. Paper reports are subject to the usual delays of printing and distribution, which creates considerable lag in the data reporting. Another disadvantage is the relative inflexibility of a paper report. It does present data, but any variations must be the object of special processing at the ASIP contractor or tedious data manipulations at the squadron level, with no engineering control over the data which may be created or the process to create it.

Various solutions were proposed to alleviate this problem. First, on-base processing was tried for a short period in the early stages of the FLMP (1990-91). The results yielded by this process were available as soon as the MSDRS tape was stripped, which was very convenient. However, the results did not always correlate with the results of the formal processing, which generated confusion. As more was learned about the fatigue data processing, it was realized that engineering intervention was required to ensure the quality of the data. On-base processing was therefore not successful and was terminated.

In 1996, Squadron Level Software (SLS), used to present and interpret the fatigue data, was put into use at the squadrons. It allows weekly reporting of the squadron fatigue data through network downloads from the ASIP contractor. It complements the monthly reports, which are still being produced at this time, by allowing various query capabilities on all the mission data. Each squadron is provided with a full data set for the last 15 months (for the entire fleet) allowing comparisons and evaluation on a yearly and quarterly basis. Sample output is shown in Figure 9. The role of SLS is strictly to allow better data access, as it does not generate tracking data.

Since the data is now in a database form, various queries can be made to allow the base level operators and maintainers to see exactly the data they wish to see. Improvements over the paper-based reports are as follows:

a) Capability to query by aircraft, pilot, mission type, configuration and fuel consumption for a given time periods; and

b) possibility to review individual mission data for each aircraft or pilot for a specified time period;

This flexibility allows the squadrons to be more independent in their fatigue management by eliminating the need for many of the custom queries directed to the ASIP contractor. This improves the turn-around time for answers while maintaining the integrity of the data which is presented. The software is built so that further queries can be easily added by means of an SQL script.

One of the effects of this greater transparency of the fatigue data is that it forces the engineering community to produce very high quality data. Squadrons are accountable for their fatigue consumption and as such, they scrutinize the data very closely, exposing any discrepancies that may originate from the processing and reporting (see section 3.1.1.7).

3.3.4 Long Term Management

Plots of the aircraft FLEI consumption against flight hours is one of the main tools for long term fleet management (Figure 10). The aircraft are managed using their monthly consumption to try to bring them within the ideal line which represents the "optimal" consumption. This works well for a new fleet, but as the CF fleet ages, scatter between individual aircraft becomes very significant. Even if all the aircraft are brought within the bounds of the optimal consumption, the scatter will cause the fleet to be retired over a long period of time. This in turn causes a reduction in fleet expectancy, since the minimum fleet size cannot be maintained (Figure 11). A new system of management had to be conceived to recuperate this "latent" life potential and reduce the scatter in the expended fatigue lives.

Instead of the monthly rates, the new system uses accumulated level of FLEI for each aircraft. The average of the fleet constitutes the goal to be attained. The aircraft are then assigned to missions known to be more or less damaging to reduce or increase their fatigue consumption and bring them closer to the average. Controls must however be placed on the fatigue rate expended to ensure proper longevity of the fleet, which is done through the 0.090 FLEI/1000 hrs flight limit.

The current goal is to reduce the fleet scatter to approx. 5% at retirement (Figure 12). A scatter of 0% is not possible. Neither is it desirable, as a gradual retirement is required for the transition to a newer fleet. Potential fleet life extension ranges from 3 to 5 years.

4.0 LESSONS LEARNED

Our experience with CF-188 fatigue tracking has taught us a series of lessons. These should be considered for the definition of any future health monitoring or fatigue life management program.

a) Although a strain-based system is efficient, there are many problems in practice. Consider the use of system based on Parametric Loads tracking when possible;

b) Always plan for a calibration (in-flight and analytical) of the strain sensors;

c) Ensure complete reconciliation between the data recorded and the relevant mission information (such as Flight Date, mission type and pilot ID) to allow accurate feedback to the operators;

d) Ensure the sensor sampling rate is sufficient to cover all potential types of loading for the considered locations;

e) Dynamic loading tracking systems are very complex and require intimate knowledge of the loading;

f) Pay particular attention to the bonding process of the strain sensors;

g) When possible, implement a Fatigue Life Management Program early in the life of the aircraft to maximize the fatigue life of the fleet;

h) Pay attention to the scatter in the fatigue life expended of the aircraft as a lot of life can be recuperated by proper management;

i) Use recorders which are easy to maintain and operate;

j) Ensure that the on-board fatigue tracking system is independent of any flight critical items and dedicated to fatigue tracking;

k) Ensure quick and direct access is available to modify any of the fatigue tracking settings parameters on-board the aircraft;

l) Maximize the onboard data storage capability;

m) Ensure that overloads do not have an unrealistic effect on the damage evaluation by selecting reference stresses that are close to actual stress levels and clipping unrealistic strains;

n) When possible, install the strain sensors on material similar to the critical location you intend to track;

o) Use network data transfer to accelerate data exchange and provide complete data access;

p) Secure the support of all levels of the operational community through education and demonstration of the benefits of the program;

q) Provide continued feedback to the operational community on their fatigue tracking data; and

r) Provide the operator with easy, timely and flexible access to the mission related information.

5.0 CONCLUSION

The objective of the CF-188 FLMP is to ensure maximum life of the airframe is achieved by controlling the aircraft fatigue damage accumulation while maintaining operational effectiveness. It was demonstrated that the use of fatigue tracking data through fatigue life management has a major impact on the aircraft life cycle cost and can help control and predict these costs. However, this was not achieved without problems on the CF-188. Although none of these problems was impossible to resolve, it would have been more efficient to address them early in the life of the aircraft. It is believed that the application of the "Lessons Learned" in the development of future health monitoring systems will increase their chance of success.

REFERENCES

1. MDC A5172, Fatigue Analysis Report, 4 December 1979

2. The Cost Effectiveness of ASIP, Capt. A.J.A Appels, November 1993

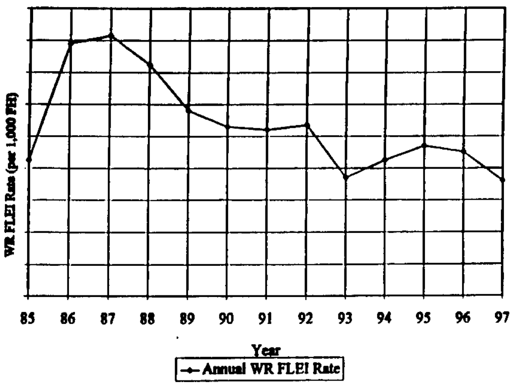


Figure 1 - CF-18 FLEI Rate by Year

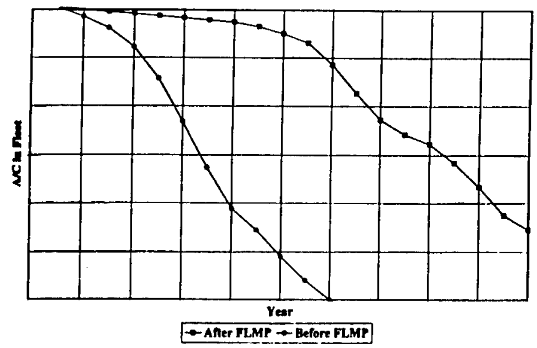


Figure 2 - Fleet Decline

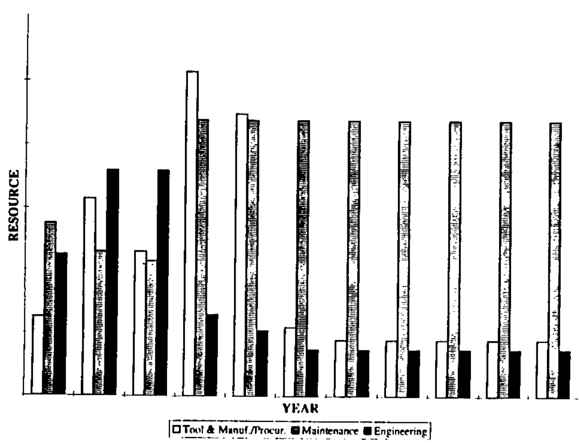


Figure 3 - Aircraft Modification Resource Planning

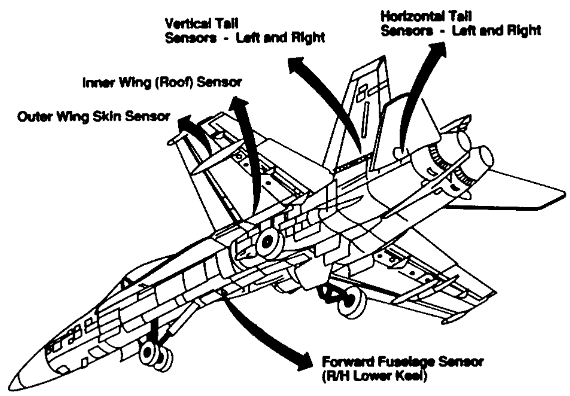


Figure 4 - CF-18 Strain Sensor Locations

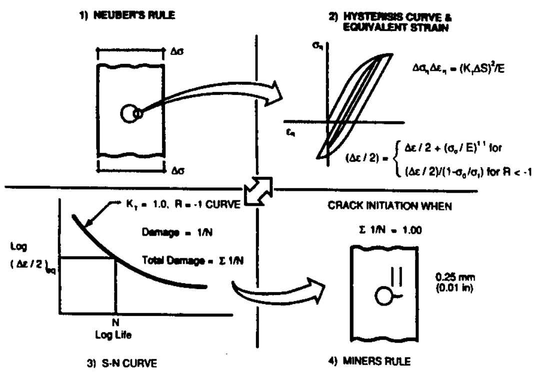


Figure 5 - Crack Initiation Concept

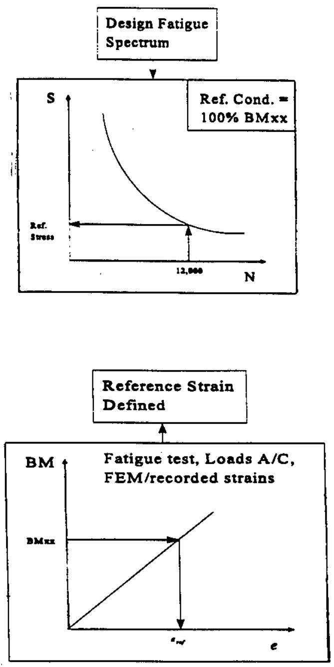


Figure 6 - Reference Stress/Strain

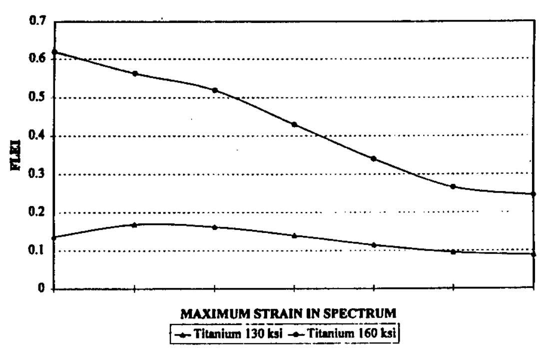


Figure 7 - Reference Stress Impact

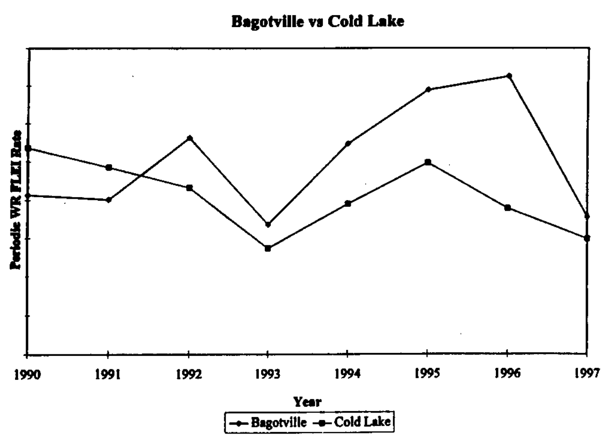


Figure 8 - Cold Lake vs Bagotville

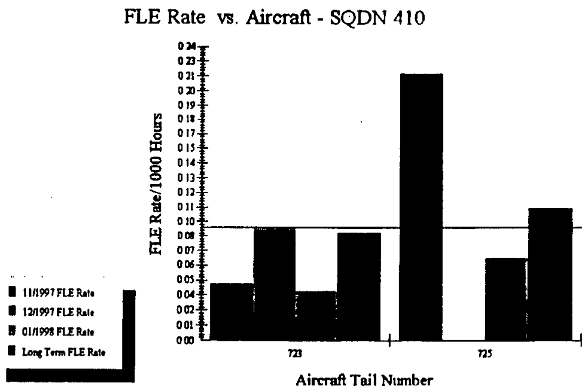


Figure 9 - SLS Sample Graph

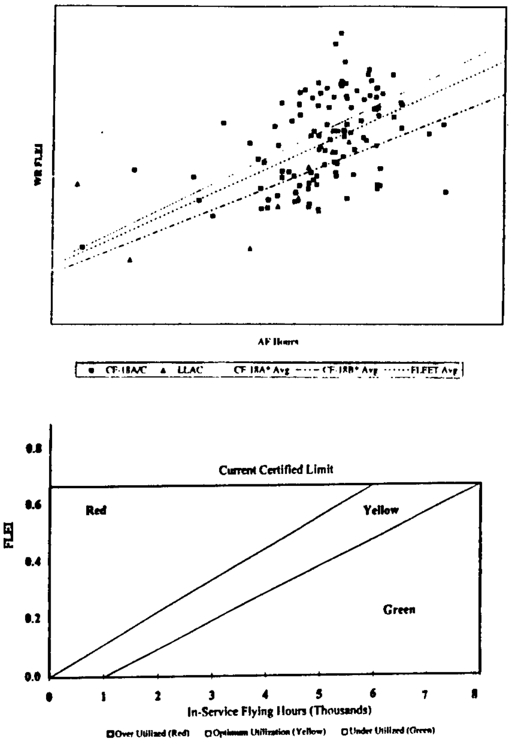


Figure 10 - CF-18 Scatter Plot and Color Code

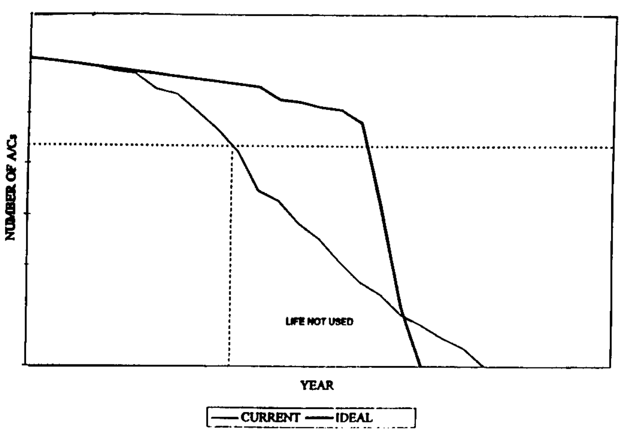


Figure 11 - Latent Fatigue Concept

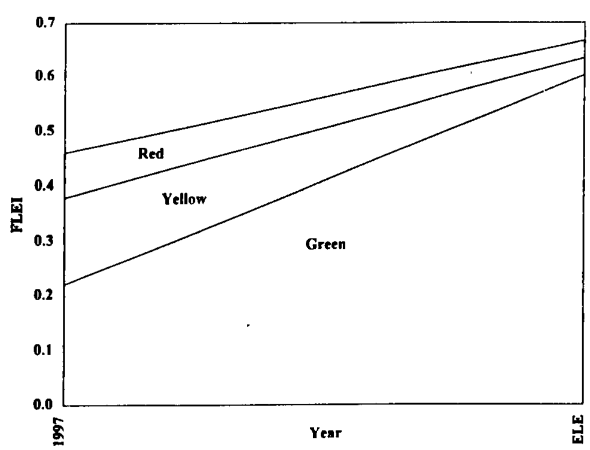


Figure 12 - Proposed Color Code

An Overview of PEP WG28 - Recommended Practices for Monitoring Gas Turbine Engine Life Consumption

G. F Harrison
Structural Materials Centre
DERA
Farnborough Hampshire GU14 0LX
United Kingdom

SUMMARY

The terms of reference for PEP WG28 provide the framework for ongoing activities. The Working Group has been tasked to examine methodologies used to predict engine life and life consumption rates for currently deployed fleets and to understand the limitations of these methods. Within this task, consideration has been given to fracture-critical components of the propulsion system namely, discs, bladed discs, impellers and spacers in both hot and cold ends. Methodologies by which mission profiles are translated into low cycle fatigue (LCF) life consumption have been addressed. The various operational usage data acquisition techniques such as gating, and continuous recording also form part of the study.

Each methodology has been elaborated with respect to its development and validation and to its capabilities in operational usage tracking, in identifying parts life consumption and in application to maintenance strategies. Experience gained using the various methodologies and strategies is documented and major practices identified. Where these conflict or are not congruent, each is discussed. Recommendations for future development are included as appropriate. Primary emphasis of the effort focused on the user viewpoint by addressing the benefits of fleet-wide usage monitoring, the experience from actual field use and the impact of manufacturing quality standards. The study is aimed at fighter aircraft, helicopters and military transport however, Civil experience is also taken into account.

INTRODUCTION

With reductions in defence budgets it is becoming of the utmost importance to achieve maximum use of existing systems. It is important to be able to use all of the system life at acceptable safety and mission-effectiveness levels. Propulsion system life consumption is a major cost driver on the costs of ownership. The WG28 study aims at improving the ability of NATO nations to predict the rate of total life consumption for fracture-critical parts of propulsion systems. It is intended to aid the nations in their identification and adoption of the most cost-effective life usage methodologies, and to provide savings in parts inventory and consumption's and hence life-cycle cost. It should also enable military operators to reduce the rate of failures which would hazard their aircraft, giving improvements in flight safety. This report provides an overview of the work currently being undertaken by PEP WG28.

1. MODES OF LIFE CONSUMPTION IN GAS TURBINE COMPONENTS

To ensure the safe operation of aero gas-turbine engines it is necessary to evaluate the potential failure modes and the effect of such failures should they arise. Where the consequences of an in-service failure are acceptable, the component can be maintained 'on condition' and repaired, or replaced following discovery of the failure. However, when the effect of failure is to hazard the integrity of the engine and the safety of the aircraft, the component is deemed to be 'fracture critical' and a finite safe-life must be determined. Fracture critical components, for which safe operating lives must be defined, include the major rotating parts, the turbomachinery disks and shafts, and structural casings subjected to high loadings. Other components in the rotating assemblies, such as spacers, coverplates and seals, may also be designated as fracture critical. To avoid unacceptable risk of catastrophic failure it is necessary to monitor the life consumption of critical components and retire them from service before their allocated life has been exceeded. A thorough understanding of the failure mechanisms affecting gas turbine components is essential if the failure modes, the safe-life, and the life usage of each component are to be accurately determined and monitored.

In this chapter, of the final report, the main potential failure modes of aeroengine components are described in simple terms and with regard to the steps required to ensure airworthiness requirements are maintained. Engine configuration and running conditions inducing such potential failure modes are discussed together with the design modifications to mitigate such modes. These failure mechanisms include LCF, high cycle fatigue (HCF), thermo-mechanical fatigue, creep, corrosion erosion, fretting and wear. It is shown how the ability of a component to resist the effects of any of these damage mechanisms is a function of the material properties, the component design and the operating environment. External factors which also have an influence on the rate of component life consumption but which can be reduced during engine manufacture, operation and maintenance are also discussed. The factors affecting life usage rate include manufacturing and material defects, build and maintenance errors, foreign object damage (FOD) and limit exceedance.

2. ANALYTICAL METHODS USED IN MODELLING THE MECHANICS OF MATERIALS FAILURE

Having provided an introduction to the various potential failure processes, this chapter develops the theme and introduces analytical descriptions of the basic physics that describes the failure modes. It provides details of how stresses and temperatures are calculated in aeroengine components and how these influence failure. It illustrates how in life assessment methods, the influence of interaction between failure modes must be considered, and how such damage rules are developed. The final report will include comprehensive coverage of all aspects of quantitative damage assessment

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

3. AIRWORTHINESS- CIVIL AND MILITARY PRACTICES

The need for engine reliability and durability has been recognised for some considerable time. The "Safe-life" and the "Damage Tolerance" approaches are the two most widely used design methodologies for producing components to meet airworthiness and life requirements. These procedures which are codified in specifications such as MIL-STD-1783 (ENSIP), MIL-E-5007E, FAR 33, MIL-E-8593A, Def Stan 00971 and JAR-E cover transport, helicopter, and fighter engines. Their scope addresses not only design, but material characterisation, engine testing, gathering of information on usage, maintenance and inspection. In this chapter the evolution of engine controls and methodology for determining the life of components that ensure gas turbine engine durability are discussed. Design practices that emphasise durability, and the advent of electronic controls have improved the reliability of helicopter, transport, and fighter gas turbine engines. Developed as a response to engine structural problems, damage tolerance and safe-life practices are part of wider specifications to ensure engine durability.

3.1 Usage Monitoring System Development

Engine usage monitoring systems depend on, but are independent of, the design data and rules of the systems they monitor. In the earlier hydro-mechanical control systems most of the transient information was held inside the control system as pressures, flow rates and diaphragm displacements. These could not easily be extracted. The advent of electronic control systems made it possible to monitor complex situations without introducing dedicated sensors. Major improvements in design methods coincided with the introduction of electronic engine controls. These have contributed to improved engine reliability and safety, by providing better data to usage monitoring systems. A discussion of Control Systems and their role in usage monitoring will form part of the final working group report.

3.2 Structural Integrity Requirements

Early structural validations were based on 150 hours of engine testing. The test cycle was simple but unfortunately it did not reveal many subsequent service related problems. Engine specifications were deficient in such areas as duty cycles, and mission analysis. Testing was not mission based. Lessons were learned by reviewing the TF30 and other engine designs led in 1973 to the U.S. Navy issuing the MIL-E7E-5007D specification. This established rigorous standards for ensuring engine durability. This document applied the Safe Life concept and recognised that components should be retired at a calculated and validated fatigue life.

Commercial regulations for gas-turbine reliability practise the safe-life approach. Review of these regulations is pertinent to the discussion of military transport, helicopter, and fighter applications as many of these commercial engines have found their way into military applications. In some cases, engines designed to civil specifications are further scrutinised under a separate set of military standards. In 1965, in the United States, the Federal Air Regulations (FAR) were codified. These defined operating limitations for disks and spacers in terms of service life. By 1970, FAR 33.14 placed restrictions on allowable life of components and required test demonstration. By 1980, the FARs approved procedures covered both analytical methods and testing.

3.3 Development of Engine Integrity Procedures

The final working group report will provide details of the essential elements of the various US Mil Spec and European Design and Lifing Policy Documents. The following documents have been selected as indicative of the significant elements of the major standards.

MIL-E-5007E (Military Specification) was issued by the US Navy in 1983. The specification requires that the safe-life approach is practised on fracture critical parts. It also requires the development of a life management master plan to cover the complete engine life-cycle. Requirements are that all engine critical parts must to be designed to twice the service life; conventional structural criteria must ensure durability from such failure modes as strength, vibration, LCF, HCF and creep. Materials are to be characterised to -3σ levels, and fracture toughness of the material must be evaluated. Verification and testing requirements include: turbine and compressor rotor overspeed testing. This includes disc burst testing of all rotating discs designs to a minimum of 122% of the maximum allowable steadystate speed. Engine durability must be substantiated via a Durability Proof Test consisting of 300 hours of accelerated mission-oriented endurance testing and by Accelerated Simulated Mission Endurance Testing (ASMET) equal to at least 1000 hours or one half of the cold parts lives. Conventional LCF validation via spin pit testing is also required.

Field monitoring is designed to validate design mission profiles and to record operating conditions. By monitoring operating experience, and providing feed-back into the life analysis process, new lives can be determined for components in a service environment. To improve life assessments further, inspections of fracture-critical in-service parts are carried out on an opportunistic basis. Recently, the US Navy has introduced damage tolerance as a refinement to the MIL-E-5007E practices.

MIL-STD-1783 (ENSIP). The Engine Structural Integrity Program (ENSIP), was issued in 1984 by the United States Air Force. ENSIP embodies a disciplined approach to structural design, analysis, development, production, and life management. It emphasises the damage tolerance approach. Components are classified as:
Safety Critical where failure could result in the probable loss of the aircraft, or in a hazard to personnel.
Mission Critical where failure could generate a significant operational impact by degrading mission capability to the extent of creating an indirect safety impact on the aircraft.
Durability Non-critical where failures could result in a minor economic impact or minor degradation of mission capability, and could initiate unscheduled maintenance actions.

Additionally, it is required that structural design must take account of realistic mission usage in setting service life requirements. Conventional structural criteria must ensure durability from such failure modes as strength, vibration, LCF, HCF, and creep. Damage tolerance (Crack growth resistance) criteria must maintain reliability in the presence of materials, manufacturing, and processing defects. Parts classified as 'safety' or 'mission' critical must receive damage tolerance analysis. Compliance may be achieved through analysis and/or verification and the internal operating environment must be defined from analysis and engine thermal/performance test data. Durability must be substantiated by engine testing or an Accelerated Mission Test (AMT). Multiple missions should be represented in the AMT evaluation.

United States Federal Air Regulations (FAR) ensure civil engine durability and are similar to military specifications in a number of ways. Requirements include the operation of a safe-life practice which must set operational requirements approved by the Federal Aviation Administration (FAA). Components must be classified as to whether they can be contained, or not, in the event of a fracture. Components judged to be non-contained must operate to life limits and component tracking. Component lives may be set through analytical prediction and/or component testing. Initial certification may be to 33% of the predicted safe-life by component testing. Additional testing is permitted to extend the safe-life (life extensions may be up to 1 cycle extension for every 2 cycles tested). The Certified Life is set at an allowed percentage of the analytically predicted safe-life. The required suitability and durability of materials must be defined and the acceptability of the materials test data base must be demonstrated/proven. The engine design and construction must minimise the development of an unsafe condition of the engine between overhaul periods. Most other international military aircraft projects create their own airworthiness standards and brief details will be provided in the full report.

European Joint Airworthiness Regulations (JAR-E) are a harmonisation of existing national regulations into an international requirement. Most nations with aviation design and operating requirements are involved in their development. They may be considered as a combination of the US FAR, MIL-STD-5007E and the British CAR regulations.

3.4 Application of Fighter Engine control Systems

The final report will address the application of hydro-mechanical and digital control systems and how these integrate with the lifing specifications associated with the particular engine. Space restrictions preclude further current discussion.

3.5 Preventative and On-Condition Maintenance Concepts

Under the ENSIP philosophy a classification is forced on each and every component which places it into one of five categories. Special tracking procedures are applied according to the durability-critical classification level of each component. A fixed period of unrepaired service (safety limit inspection interval) for safety and mission critical components is required.. This process forces appropriate maintenance actions down to component level. Flight safety and mission capability are the paramount concerns, while cost is emphasised to a lesser degree in this approach to maintenance actions.

Under "Safe-life" procedures, components are not as tightly classified. In this case, inspections are normally conducted only when the aircraft, engine, or component is available for disassembly. Fracture critical components are only forced to be replaced once a fatigue limit life is reached. Components can also be classified with different life limits depending on whether they are in the hot or cold section of the engine (i.e. turbine or fan and compressor), which also influences the period at which a component or engine module is brought in for overhaul and inspection. Frequently, the hot section life between overhauls is set at half the capability requirement of the cold section. Flight safety is again the paramount concern Because inspection facilities are not planned into the infrastructure as part of the lifing policy, service parts non-availability may influence maintenance decisions which directly affect safety

The two most popular maintenance concepts are *Preventative Maintenance* and *On-Condition Maintenance*. In a *Preventative Maintenance* practice the engine is brought in at a fixed interval, dismantled and inspected before any incipient problem is detectable or operational use is affected. Components are also retired at the end of their LCF life or can be extended through retirement for cause. Preventative maintenance practice does have life cycle cost benefits that can outweigh the implementation costs. Engines must however, be designed to allow for ease of disassembly. Isolated design/manufacturing faults can be more effectively managed under preventative maintenance practices due to routine scheduled inspection opportunities.

Under an *On-Condition Maintenance* regime, inspections and component replacement are conducted only when a problem has been detected and tracked to a point where a maintenance action is necessary to prevent loss of function. Life usage tracking and component replacement when the component reaches the end of its useful fatigue life falls naturally into this system. Inspections are more opportunistic, conducted when the aircraft, engine, or assemblies are accessible. Engine overhauls are scheduled to occur once a minimum fatigue life is reached for the limiting rotating components in the engine. The opportunistic periods for inspection, while not fixed, do eventually catch the majority of the population. Isolated design and manufacturing defects may produce premature cracking or component failure. In an On-Condition Maintenance regime these failures may not be found until there is a detectable loss of functionality. This places a requirement on designers to ensure that incipient failures are detectable before a total loss of function occurs, and that appropriate monitoring systems are in place.

Improved durability design practices and the introduction of electronic engine controls into gas turbine engines coincide historically. 'Safe-life' and 'Damage Tolerance' design practices are the most commonly followed procedures to ensure reliable engine design. Neither practice is right for every agency. When choosing a design practice, consideration must be given to fleet utilisation, access to inspection facilities and to the results of life cycle cost comparison (i.e. safe-life versus damage tolerance approaches). The safe-life method is more widely practised than the damage tolerance method, however, the competitive nature of the engine market place is driving many new engine designs towards a damage tolerance approach for highly stressed components. Comparison of the safe-life and ENSIP based processes shows that each offers advantages and disadvantages to the Services that practice them. A safe-life practice offers lower maintenance, maximisation of "on wing" time without inspection and lower facilities and equipment costs associated with maintenance. Disadvantages of the safe-life method include fewer fleet management options and no focused inspection process available if cracking problems develop. Although modifications to the safe-life practice offer an opportunity to extend life of components though the application of fracture mechanics, after a minimum safe-life is reached, inherent damage tolerance design practices have the advantages of potential increased use of parts beyond LCF limits, more thorough testing and verification of designs and a more readily available focused inspection capability to address service cracking problems should they arise. Disadvantages of imposing damage tolerance practices include higher implement costs than with the safe-life process, weight may be added to the design, a larger infrastructure is required and parts handling is increased.

4. USAGE TRACKING SURVEY AND MISSION ANALYSIS

The main tasks of a ground station are trend analyses forecasts and display of the results in a variety of tabular and graphical formats. System integrity should be ensured by appropriate input data checking and frequent backups. Missions are characterised by sequences of manoeuvres (e.g. engine start-up, taxi, take-off, etc.). The manoeuvres can be described as blocks of profiles of the different engine parameters. These blocks can be either directly obtained by recording measured data or generated from simple description using performance models of aircraft and engine. With these blocks, synthetic missions can be built up and analysed with respect to lifing aspects, e.g. life consumption of a single mission, 'typical' life consumption, improvement of theoretical design missions and refinement of engine design. Usage survey and mission analysis provide the justification for usage tracking. The accuracy of the results obtained with such systems depends strongly on the capabilities of the algorithms applied. In practice these range from the simple to the complex. The latter procedures take into account the transient metal temperatures development in the components, centrifugal stresses, thermal stresses, assembly stresses, gas pressure stresses and stresses induced by manoeuvre loads and other influences.

4.1 Usage Tracking

Usage tracking' is defined as a process which monitors the usage of each individual critical part in terms of life consumption. It is required because fracture critical parts in aero engines have limited fatigue life, and because of the desire to operate the engines safely and economically. On-line usage tracking computes the consumed life directly from measured engine parameters. Bulk data storage is required to gather flight data where ground processing is in operation. Stored data may be used not only for life usage tracking, but also for algorithm verification and design improvement. For critical life limited parts, the airworthiness regulations require that each individual part be traceable throughout its service life history, and that complete records of usage and supporting activities (inspections, repair) are maintained.. The main tracking methods currently used include time based, event based, mission type based and cycle extraction based systems.

Time Based Usage Tracking relates life consumption directly to time. A cyclic exchange rate must be established to provide the relationship between life consumption and engine ground-running time.

Event Based Usage Tracking is a development of Time -based systems and relates the usage to 'events', which may be defined as single throttle excursions, sequences of throttle movements or complete flights. Generally, a flight (or an engine run) contains one main excursion from zero to max back to zero and a number of sub-excursions (e.g. idle-max-idle, idle-intermediate-idle, max-intermediate-max). Since excursions with different ranges cause significantly different amounts of fatigue damage, it is necessary to weight the excursions according to their contribution to life consumption.

Mission-Type Based Usage Tracking: Since military aero engines fly a variety of different mission profiles (or sortie patterns). The profiles are made up of a number of elements such as take-off, climb, cruise high, cruise low, low-speed dash. Each profile is identified by a mission or sortie code. For each of the different mission profiles typical throttle sequences can be specified. These sequences can be decomposed into individual throttle excursions and an equivalent number of main excursions. The corresponding life consumption per mission can then be determined.

Cycle Extraction Usage Tracking : The approach here is to derive the stress-temperature histories at the monitored areas from measured engine operating parameters and convert them into life usage. These algorithms can now be incorporated into on-board monitoring systems.

The accuracy of the applied algorithms must be analysed and taken into account when individual component life usage tracking methods are assessed. If a 'reduced' or simplified model is used for analysis it must be validated against the full design model, which should include the full component and materials databases. Additionally, if the analysis is done on-board then only the results need to be transferred on a routine basis.

4.2 Cyclic Exchange Rate

The Cyclic Exchange Rate describes the relationship between life consumption during operational usage and the correlated engine running time. For most military applications the engine flight time in hours (EFH) is used as the base. The determination of 'safe' cyclic exchange rates involves collection of recorded flight data from a statistically representative number of engines with a sufficient number of flights per engine; calculation of the life consumption increment of individually monitored critical areas for each engine and each flight. This enables calculation of the feature specific individual cyclic exchange rate, i.e. the average life consumption per unit time for each individual critical area for each engine. A statistical frequency distribution of these cyclic exchange-rates can then be calculated for all engines and the 'safe' cyclic exchange rate established as an acceptable quantile of this distribution.

4.3 Data Selection Criteria

Input parameters necessary for life usage tracking include engine parameters, aircraft parameters, configuration data and supplementing general information. The required signal quality needs to be specified in terms of accuracy, resolution, possible drift and update rate. Plausibility checks and data restoration processes need to be applied to ensure data integrity. Life usage calculations require a variety of input parameters. Engine parameters may include: spool speeds, shaft speeds, intake temperature, compressor exit temperature, stator outlet temperature, turbine blade temperature, exhaust gas temperature, engine intake pressure, compressor exit pressure, jet pipe pressure, oil temperature and fuel flow. Aircraft parameters may include air temperature, indicated airspeed, pressure, altitude, angle of attack, angle of side slip g-load, normal acceleration, yaw angle velocity and weight-on-wheels indicator. Additional configuration data required may include: aircraft type, aircraft variant engine type, engine variant, aircraft and engine serial numbers, monitoring system hardware and software versions. Correlations can sometimes be used to derive other parameters thus making some direct measurements unnecessary.

Signal quality is mainly determined by the characteristics of the sensor and signal acquisition chain. These characteristics include accuracy, resolution, possible drift, and time delays in the response to transient inputs. Signal sampling rate is also important since frequencies need to be high enough to detect all modulations of the signal and to avoid aliasing errors.

Plausibility checks are necessary since monitored data are sensitive to disturbances. To ensure that lifing calculations are reliably executed, typical checks are range, rate, cross, and model checks. A strategy for handling implausible data must be decided and built into any automated life usage monitoring equipment. There are strong pressures to use data compression to reduce the amount of data to be handled and transferred. Monitored data must be uniquely assigned to an individual engine, engine component or engine part.

4.4 Data Transfer

Monitoring systems are distributed systems and communication between the different parts is essential. The necessary data transfer is carried out with magnetic media (diskettes, tapes, solid state modules) or data-bus links. A high level of automation will help to satisfy the requirements for data integrity and robust and user-friendly operation. Reliable and accurate communication between these system parts is essential. Simple systems (which only collect data on-board) require only uni-directional data transfer, i.e. download of recorded data into the ground station for further processing. More sophisticated applications need regular bi-directional communication.

4.5 Records and Accounting

It is necessary to record the life usage data of engine parts, engine components (or modules) and complete engines in a central or distributed register, and to keep these records up to date. Maintenance of any type of configuration data requires logistics information. This will encompass keeping track of which engines are on which aircraft with regular updating of part numbers and serial numbers for all tracked components, modules as during the life of an aero engine, components are removed from service several times for overhaul and repair.

The ground station should contain all the information necessary for life usage tracking of each engine belonging to the squadron or fleet. This information should include a complete description of the engine configuration, serial numbers of engine, modules and life-limited parts as well as the life usage data for all the parts. This should also encompasses: calculation of cyclic exchange rates, substitution and restoration of lifing data, trending, conversion of cyclic data into time based data, reading across from monitored parts, preparation of initialisation data for the on-board system. Cyclic exchange rates can be provided for individual engines mission profiles and for fleetwide averages. Safety factors should be included for deviations and scatter in all cyclic exchange rates. Substitute calculations should be performed if lifing data are corrupted or lost.

Trending of lifing data means both keeping track of historical data and projecting trend lines into the future to anticipate the need for action. Trends are normally provided for individual engines, but they can also be evaluated for certain mission profiles, air bases or fleetwide. Evaluation of trends produces information about changes in usage and isolated events which can be correlated with reported problems and diagnosis

4.6 Mission characterisation

Mission analysis is undertaken to enable the prediction of engine life consumption for mission profiles which are composed of the above mentioned manoeuvres. One approach is to build up a complete mission profile by combining a number of blocks which contain the profiles of the engine parameters corresponding to certain manoeuvres. In a first step, a representative number of mission profiles are analysed to identify both the manoeuvres which occur and the frequency at which they occur in each mission profile. Recordings for all identified manoeuvres are obtained and separated into blocks. The life usage related to particular manoeuvres or blocks, is determined respectively. Once the life usage of a special type of manoeuvre is found from this analysis, an average profile of such a manoeuvre can be created.

In Transient Mission Analysis the engine parameters are recorded transient data (i.e. to describe the transient response of the engine to modulations of the pilot's lever). This means that the engine performance model must be capable of providing transient outputs. Accurate prediction of transient engine behaviour is much more difficult than the prediction of a steady-state response. Nevertheless, fine tuning of the performance model should be carried out using recorded engine parameters. With the tools and mission blocks described above, 'synthetic' mission profiles can be generated. These mission profiles are composed of arbitrary sequences of all of the available blocks.

5. STATISTICAL ASSESSMENT OF USAGE DATA FROM MONITORING SYSTEMS

Once all of the research and development described elsewhere is finished, an aircraft, engine, and engine monitoring system will be introduced into service. This chapter addresses the statistical basis for deciding how many stress features, engines or aircraft require to be monitored. It provides analysis of real data collected from in-service aircraft engines and the interpretation of these data.

5.1 The Statistical Aspects of Fleet Component Monitoring

Proportion Monitored: On production lines that mass produce car components for example, less than one in a hundred components may be inspected. This is enough to ensure that very exacting standards are met. With a fleet of several hundred engines it may therefore be considered adequate to monitor only a small proportion. Unfortunately there is a vast difference between the two situations. Assuming that the measured parameters of each of the above populations (aeroengine component usage and motor car component production,) fit standard distribution curves. Although the number of samples for each may be similar, one is likely to have a very small variance or spread whilst the other will be large. On the production line, most of the factors which cause variations have been brought under tight control, whereas for military aircraft, and more particularly the engine, the usage spectrum has not. The final report will illustrate the use of the Student's t statistical function in the analysis of such data. This simple analysis shows that a) the number of components to be sampled depends primarily on the variance (or degree of scatter) of the sampled data and the sample size, rather than on the fleet size; b) Once the desired sample size for a particular fleet or aircraft type has been determined, a doubling of fleet size, with no change in the way in which the aircraft are used, requires about a 40% increase in the required number of sampled aircraft; c) The method can be modified and applied to any lifed component with a usage distribution that can be mapped onto a normal distribution. (an example is the log normal fatigue distribution); c) In deciding how many engines in a fleet should be monitored it is necessary to have enough sample data so that the normalised sample variance can be determined in a statistically valid way (for guidance, typical values observed on the UK Harrier and Tornado aircraft lie between about 0.4 and 1.6 depending on the component selected whilst most components have values between 0.4 and 0.7). d) At least 30 samples should exist for each statistical degree of freedom, for each permutation of variables such,, operating base, sortie pattern, time of year, etc.

Evaluation of Recorded Data: When the Harrier GR Mk 5 entered service in 1989, Pegasus Mk 105 engine component usage data were recorded on a sample of about the first 1700 flights of the first twelve aircraft. These data were collected at the rate of eight samples per second with real time on-board computation. Typically, HPC life usage results revealed approximately Poisson distributions for individual engines and Normal distributions for the whole fleet. From a general statistical viewpoint the variation in observed life usage counts was found to be high. This is because the standard deviation is at least 50% of the mean throughout. To put this into context this ratio is usually in the order of 20% to 30%. The analysis of usage data for an HPT disc showed that at least 70% of the fleet should be monitored if an error between the sample and population mean of less than 10% is to be achieved

The GAF fleet of Tornado aircraft with RB199 engines is equipped with the On-board Life Usage Monitoring System (OLMOS), which has provided engine life usage data from many thousands of flights. Detailed assessment of the data has shown that significant differences in usage exist between the right and left engines, due to engine handling procedures (namely, which engine is first switched on and off, and the resulting warming up and cooling down periods) and that there is a sharp cut-off for low life usage values, that is independent of flight time (due to the minimum power requirement for take-off, and to the start-up and shut-down stresses). Differences in engine response to the control system and differences in settings (resulting possibly in overshoots at the end of accelerations) have also been shown to contribute.

Statistical Modelling: The final report will provide illustrations of the result of fitting a 3 parameter Weibull distributions to the life usage data. The distribution function describes the difference between parts, components, engines, but not different flights. Scatter in life consumption between different flights has already been smoothed due to the accumulating nature of life consumption. It has been noted that the small number of high life-usage points were not very well approximated by this distribution, and that this is one of the reasons for individual engine monitoring. When these RB199 observations are compared with the distributions discussed for the Harrier, there is sufficient similarity to suggest that these findings may be generally true. One factor that may change this is an improvement in the underlying algorithms which brings new factors into consideration. Because airworthiness is subject to formal procedures and therefore well proven, it seems likely that future changes will be refinements of the existing procedures. Examples of such improvements are that within the GAF a thrust-rating system is used for take-off. This has resulted in an observed increase in life- usage rates against time in service. In some engines the transient thermal stresses are now included in the dynamic models and result in much accurate modelling than in the past.

Fleetwide Life Usage Monitoring: The fleetwide normalised probability distribution for life usage across all RB199 fracture critical components shows a Normal/Poisson distribution, with some skew to the higher values. There is a considerable smoothing effect due to the varying life usage characteristics, but a lower cut-off point of about 0.5 can be clearly established. Of more practical use to fleet management is a diagram showing the distribution of life usage in the fleet for specific engine components. For the first IP compressor disk, such a diagram (presented in the full report) reveals from the shape of the low end tail that spare components have recently been built into engines. Between the 30% and 80% life usage band relates to the majority of parts which were built into the engines as new. The high end components are of greatest significance for safety and planning purposes. They represent fast/hot running engines, and the repeated assignment of certain aircraft to particularly demanding missions. Similar charts can be derived for all components. This is often a task for the fleet management systems.

Individual Engine Usage Monitoring: There must be a point beyond which the cost of gaining another few cycles from a part is greater than the benefit. It has been shown that the replacement cost may exceed the purchase price of a part by a factor of 10 and that individual monitoring of component life usage can more than double the available life for a given part. For older engines, lifing methods may use the concept of Tactical Air Cycles (TACs). This is based on dividing the speed and temperature range that the engine runs through into bands, and then assigning damage counts according to a predefined set of rules. Typically the bands are associated with the qualified power ratings. In practice there are several drawbacks to these methods. The resolving power of the methods used mean that some significant cycles are not counted, and particularly when small cycles are accruing at high speeds this is often not directly allowed for. Additionally, it is not possible to review retrospectively and reassess component life usage with the accuracy that is possible with later methods which are designed to catch turning points.

When usage data collected by such simpler methods are plotted on a distribution chart, the appearance can be misleadingly similar to more advanced methods. When TACs are employed, the exploited life may be slightly greater than the fleetwide value, but the uncertainty value would also be expected to be large.

6. TRANSLATION OF SERVICE USAGE INTO COMPONENT LIFE CONSUMPTION

This chapter discusses the design techniques used to provide the structural component life required by the customer with the safety levels imposed by the regulatory authorities. It discusses the development and validation of the materials algorithms developed both to determine the declared component life for use in the usage recorders to evaluate the rate at which this life is consumed under service usage. Obviously differences exist, both between the qualification requirements set by the various regulatory authorities, and between the various manufacturers' design methods used to satisfy the authorities. This topic will form an essential part of the final report. Since it will be broadly covered under a separate paper it will not be further discussed herein.

7. MAINTENANCE POLICIES AND PROCEDURES

There are three criteria for the success of a maintenance policy for fracture critical rotating engine components. Primarily, any policy must ensure the airworthiness of the aero engine by ensuring that the parts are removed from service in time to prevent catastrophic failures, it must also be affordable, and the engines must provide the combat readiness needed to support the national security strategy. Amongst other things, maintenance policy is influenced by the era in which the engine was developed, the sophistication of the aircraft in which the engine is installed, the information technology base of the operator, design improvements introduced by the engine manufacturer and the inherent life of the engine components.

7.1 Engine Classification

Today's fighter engines now have unrestricted throttle movement with stall-free operation at all points of the flight envelope as standard features. They also have slam-acceleration times of a few seconds, rather than the 30 to 60 seconds of earlier engines. This results in the potential accumulation of many types of cycles resulting from throttle changes. Some modern engines also limit the effect of rotor transients by modulating thrust using the variable vanes, exhaust nozzle position and fuel flow to maintain a constant rotor speed. Older engines have operating limitations which may limit throttle movements and they may not experience the variety of cycle types experienced by a more modern engine.

Turbine engine powered transport aircraft may have either a turboprop or turbofan/turbojet engines. In general, the age of the system determines the engine maintenance policy. Transport aircraft tend to fly with far fewer throttle movements than fighter aircraft. Exceptions to this are that a large number of throttle excursions may be used when automatic landing systems are in use, during tanking (inflight re-fuelling) manoeuvres and low level freight drops .

Helicopters have been fitted with turbo-shaft engines for many years. Once again, the age of the system and design requirements determine the engine maintenance policy. Turbo-shaft engines are typically designed with cyclic-life limited components and are maintained to a hard-life hourly limit. Numerous rotating components are life limited due to the rotor-speed related stresses that they endure. Other components in the turbo-shaft engines are effectively life limited by thermal cycling because they operate at a constant rotor speed for power settings above idle. Special considerations for helicopter engines are: torque-matching of engine power outputs to protect helicopter gearboxes; multiple engine control system adjustment to ensure that one engine does not use excessive life, compared to the others.

7.2 Operations & Support Costs

The general relationship between the maintenance strategy and the operating and support costs is fairly simple. The acquisition cost of the system accounts for approximately half the life cycle cost with maintenance and repair costs accounting for the remainder. There appears to be an optimal relationship between the material replaced at each shop visit and operating and support costs. For illustration, commercial experience with CF6-80A engines on the A310 aircraft has conclusively shown that operating and maintenance costs are best optimised by minimising the cost per flying hour, rather than shop visit and material replacement costs. The F-16 C/D with F100-PW-220 operates to an on-condition maintenance program which focuses on restoring functionality at the right time. When this engine entered into service the Mean Time Between Repairs (MTBR) approached 800 engine flight hours, and the operation and supports cost were quite low. As the engine aged and the component failure rate increased, maintenance was performed at the module level without regard for the overall engine performance and coherence. Consequently, the operator found that reliability decreased with a commensurate increase in maintenance costs. Engine life management is a high payoff, relatively low cost process that significantly improves both readiness and supportability.

With regard to cost drivers, a comparison of the basic differences in the MIL-STD-5007E, a predominately safe life approach, with the damage tolerance MIL-STD-1783 approach shows that material characterisation, parts classification, and sub-system verification testing create extra effort in developing the damage tolerant design. This work can add as much as 5 years to an engine development program activity and approximately 10 - 20 million US dollars in cost and another 20 - 30 million US dollars to implement. These additional costs include:- more refined finite element analysis to define stress states for crack growth analysis; crack growth life computations; crack growth characterisation of materials; verification testing of expensive prototype parts; increased fabrication costs; increased cost of damage tolerant process materials; increased inspection costs (focused fluorescent penetrant inspection (FPI) and eddy current) on each engine; inspection facility acquisition. Benefits that go along with these costs include improved component robustness at entry into service, maximised time between overhaul and greater utilisation of available safe life.

Reliability Centered Maintenance (RCM) concept was first employed in the early 1980's. The philosophy is based on an examination of the operational consequences of failure for each component. If the consequences of failure are unacceptable, a maintenance task is established for the component. The result is a program that achieves safety and operational goals while minimising costs, maintenance man-hours and material usage. RCM uses engineering decision logic to evaluate each failure mode and mathematically determines the risk over the life of a system or engine.

7.3 Physical Basis for Inspection and Re-use

There are two competing strategies for life management, the safe-life approach and the damage tolerance approach. In using the safe life approach, the hard life defined in hours is based on a calculated or demonstrated life for a nominal component. Conservative factors are applied to the material properties or the demonstrated test life to allow a safety margin for the minimum material property component and extremes of operation. Within the damage tolerance approach there are two methods: standard ENSIP, and extended ENSIP that includes "retirement for cause".

The safe life approach has been criticised as being overly conservative and costly on the grounds that discs are usually discarded with significant amount of useful residual life. Economical considerations would suggest that procedures which allow the use of components to their individual life rather than the life of the shortest lived component will reduce the need for component replacement. Likewise, total dependence on rejection by inspection for components with long manufacturing lead times and demand conditioned purely on part rejection will lead to readiness issues at fleet lives approaching the nominal. The secret to cost efficient maintenance is to achieve a balance between *part life usage and readiness*. It is quite clear that the maximum life cannot be safely extracted from the whole fleet of components unless each component is considered, and has its life-usage tracked on an individual basis

The damage tolerance approach is dependent on the component to being designed to possess a relatively long crack growth (residual life) period, between a detectable flaw and rupture. This time determines the basis for the interval at which inspections must be performed, to assure the component is still serviceable, and to retire those components that have flaws. The standard ENSIP method sets inspection intervals, but the retirement life is still a "hard time" life as detailed above. The "retirement for cause" method also includes damage tolerance analysis to set inspection intervals, however, retirement life is based on repetitive inspections until a flaw is found, or the part is retired for economic reasons.

At some point in the life of a component, economics will dictate replacement of a component. The lifing procedure employed to establish life limits varies with component type, and the form of damage. In modern gas-turbines, discs and spacers are normally designed to withstand LCF, burst by overspeed, and creep. For discs, the most common lifing procedure for establishing safe life limits follows a 'time or cycles to crack-initiation' criterion. Turbine blades and vanes are designed to withstand creep as well as thermo-mechanical fatigue and HCF. Turbine blades and vanes in aero engines are seldom lifed because of the difficulties associated with predicting the service behaviour of metallurgically complex material systems under conditions that can vary widely with user practice. A 'life on-condition' approach is sometimes employed where, for instance, creep growth or untwist of airfoils is measured and distortion limits are used as retirement criteria.

The successful application of damage tolerance based procedures depends on the supporting technologies. These technologies include NDI, mechanical testing of test coupons and components, structural analysis, mission profile analysis and condition monitoring of components. Extensive materials testing must be performed to obtain crack growth-rate data, for the application of deterministic and probabilistic fracture-mechanics based life-prediction concepts. The Engine Structural Integrity Program (ENSIP) for the USAF takes account of these factors. It recognises the engineering nature of the materials and requires inherent defect distribution characterisation. It also provides for their inclusion into the design process and establishes the level of inspection at manufacture to assure the control of the processes that generate these defects and, based on this, establishes the residual life for each component.

In the operation of a life management policy based on NDI, no single process is universally applicable. Eddy current, ultrasonic and fluorescent penetrant methods have found wide acceptance. X-ray, magnetic particle and others have practical but limited application. Once the initial flaw size has been determined for the *as-manufactured* condition, an interval of failure free operation can be determined for either an actual initial material defect distribution (probabilistic) or assumed flaw distribution (deterministic). Based on the crack growth characteristics of the material, environment, and loading, a crack growth forecast is then calculated. An interval is established to allow the NDI operation to just miss a flaw and provide a safe growth interval (one-half the crack growth life). Subsequent inspections determine whether such a flaw existed. Any inspection process inherently has a finite probability of finding a crack and a therefore a finite chance of missing a crack. It is essential to be able to quantify the probability of detection (POD) to establish an effective inspection process for specific component critical areas, such as bolt holes, broached features, webs and other areas. The emphasis of this approach is on the largest crack which could be missed rather than the smallest crack which can be found Finally, it is important to combine the concepts of "hard time" with "retirement for cause" to minimise readiness concerns, and maximise safety. This means that all components, regardless of their crack-life history should have an overriding 'hard life' at which they will be retired. In most cases it is preferable to retire a fraction of the population before their actual lives are consumed to preclude any inservice critical component failures.

Inspection Requirements for Damage Tolerance (MIL-STD-1783) require NDE of critical components during manufacturing, overhaul and field inspection. It requires consideration of inspection at the design stage and limits the period of uninspected service of all safety critical parts to the demonstrated inspection capability. In NDE reliability demonstration programs to identify confidence limits, based on testing against known standards, it has been found that FPI is capable of finding a 0.035 inch deep by 0.070 inch long flaw with good reliability for whole field inspections and a 0.020 inch deep by 0.040 inch long crack under focused inspection procedures. The verification and use of enhanced eddy current inspection procedures is a major difference between using the damage tolerance and the safe-life management procedures.

7.4 Usage Monitoring in Life Determination

There are several reasons for operating life-usage monitoring. At the most basic level it may simply be to prevent the failure of components in service, and to maximise use for a given cost. It can be used to refine understanding of mission profiles, mission mixes and operating conditions. Monitoring of the fleet usage enables operational changes to be identified in time to adjust life analysis and to make subsequent maintenance and logistics provisions.

At a basic level, Parts Life Tracking maintains a record of the hours of operation of the engine in which the component is installed. The remaining life of the component is recorded as the engine is assembled and then debited at periodic intervals as the engine is operated. When the declared life is reached, the engine is disassembled and the component replaced and retired. More sophisticated methods accumulates cycles by a conversion of operating time and missions to cycles via a fixed algorithm. The conversion factors are sometimes known as exchange rate (beta) factors. A yet more complex method, is for an on-board processor to measure cycles and time in specific temperature bands. This can be downloaded to a ground based computer system that maintains the configuration records and debits the life remaining on the tracked parts. An alternative is for the data to be recorded in bulk and then downloaded for analysis on a ground based system. This allows the full manufacturers design model to be used for analysis if required. It provides the technically most accurate approach. The final enhancement is for the records to be updated in near real time by the on-board processing system as well as maintaining the configuration records. This is periodically downloaded to ground systems for planning use and avoids any out of base deployment data loss. Usually a reduced version of the manufacturers design model will be used to extract the major and minor cycles and sum them for each monitored stress feature

7.5 Retirement Strategies

Retirement strategies generally fall into two categories; predetermined intervals and retirement for cause. Predetermined interval strategies are generally based on LCF life. The LCF limit is established based on a lower bound distribution of crack initiation times. The use of the lower bound minimises the occurrence of cracking and the need for hardware repair. Implementation of this conventional strategy requires 100% hardware replacement at the lower bound LCF limit. Damage tolerance control philosophies may also be operated with this strategy.

By using the retirement for cause (RFC) methodologies, each component is used until a crack is found. Substantial development of fracture mechanics concepts over the last several years has permitted the degree of fidelity and predictability for the crack propagation rates necessary to implement this type of approach. In such an approach, all components are inspected first at the end of a safety limit period divided by an appropriate safety margin (SL/SM) which would maintain the equivalent risk used in the conventional design life. It is important to note that in order to maintain an equivalent risk, there will be an increase in the number of areas to be inspected over the design life safety limit inspections. Only those components containing defects equal to or greater than the specified crack size, a_i are retired. All other components are returned to service with the assumption that, if they have a flaw, it is less than a_i and thus good for another inspection interval. This ensures the crack propagation life is continually reset to a safe value and ensures that components are rejected only for cause (cracks/defects). Each component is allowed to operate to its own specific crack initiation life. Clearly not all fatigue-limited hardware can be handled in this fashion. Suitable components must be designed to have a slow crack propagation rate, the inspection interval must be long enough so as not to put undue constraints on weapon system readiness and the cost of the overhaul, and inspection of the hardware must be economical and not negate the advantage of the life extension.

Developing inspection processes for RFC components is a major undertaking. The inspection techniques must be able to find cracks emanating from any orientation in a finished part (not just a sonic envelope). To achieve this, new inspection standards, from which to measure the adequacy of new inspection techniques, must be designed and fabricated with buried flaws. The USAF has an inspection system and a set of standards which were developed for this purpose at San Antonio Air Logistics Center. An alternative is to use RFC for surface flaws only and to retire hardware at the buried flaw inspection life limit. This approach can still yield significant extension in life over the historic minimum LCF life criteria.

7.6 Damage Tolerance Applied in Risk Management

Damage tolerance analysis can be used under safe life methods to allow life extension beyond LCF limits, and. to enable a life shortfall to be managed. In the case of a life shortfall, crack growth lives are calculated from the minimum size associated with the inspection method (Eddy current, ultrasonic, or FPI). A management plan is developed to keep risk to a minimum level based on the current fleet life distribution and the risks identified with further service usage.

7.7 Overview of Current Maintenance Strategies

Extensive details have been provided from each nation for its individual aircraft types.

For the Royal Air Force, authorised lives are decided and controlled by the appropriate Engineering Authority (EA) with advice from the manufacturers and the MOD(PE). A variety of strategies has been adopted to produce authorised lives, the most common being a safe life determined from fatigue-based criteria, although the damage tolerant approach has been applied in a few cases. In practice, the Pegasus engine has 36 identified fracture critical components which have individual lives based on the original manufacturer's algorithms and safe life limits.

The US Air Force has been employing ENSIP maintenance policies on the F100-PW-100/200 engines. This is a classical damage tolerance approach to life management. Critical rotating components, static structure, and pressure vessels are retired at the minimum predicted LCF life, with the exception of some components which make full use of the damage tolerance philosophy by employing "retirement for cause". All critical hardware are inspected with eddy current, ultrasonic, or fluorescent penetrant inspection at procurement and then re-inspected at pre-determined intervals set by damage tolerance analysis. All older engines which entered service prior to the establishment of ENSIP policies are managed similarly to the Navy. When problems occur with these engines such as LCF cracking, damage tolerance assessments are often employed to establish a risk based life management approach which ensures supportability and fleet readiness while maintaining an acceptable level of risk.

The US Navy maintenance strategy for critical rotating and pressure vessel components is to retire these parts at the minimum predicted safe-life (LCF) limit. Parts are inspected for manufacturing defects, with FPI, when they are purchased. Life limited parts are inspected during operational service when they are fully exposed at the depot for other maintenance purposes. This is known as an opportunistic inspection and is conducted using FPI methods. Whenever an LCF crack is found, a complete review of the lifing methodology is performed to verify or validate the existing life limit All engine LCF life limited parts are managed to a hard life. limit. The majority of mature navy engine LCF limited parts are tracked by engine flight hours. Only the most recently designed engines are fitted with high fidelity engine monitoring systems. Since 1993, the Navy has initiated programs to update existing life limits. The updates include, enhancing thermal and stress models, incorporating fully characterised materials data, substantiating model inputs using instrumented engine test results and lifing to the flight profiles recorded during missions.

8. VALIDATION REQUIREMENTS

The life control process defines the appropriate safe and economic lives for components by using comprehensive engine models and material property databases. The service life consumption of safety critical components must then be monitored using a set of simplified models (reduced order algorithms). These simplified models must be calibrated against more comprehensive models and appropriate factors (based on the quality of the method) incorporated to maintain satisfactory levels of safety (where failure can be safety critical) or accuracy (where failure has only economic consequences). Validation of the total control process covers the engine manufacturer's design process, including models for establishing component stresses and temperatures (full finite element models), material property databases and. verification of algorithms derived to represent service operating environments and how these affect life consumption. Airborne system hardware including sensors, equipment and procedures for data transfer, ground station hardware also require validation..

8.1 Integrity requirements for usage monitoring systems

The consumption of service life is a progressive process. If, therefore, a usage monitoring system counts incorrectly for a small proportion of the time it may not cause a safety hazard. The incorrect counting may arise from a variety of, temporary, causes such as corruption of input data, flight conditions out of range relative to the design envelope, or a failure within the monitoring system. Pessimistic, default values should be used when a detectable fault occurs. A full validation process should cover all foreseeable circumstances and confirm that systematic errors will not occur. Therefore any incorrect counting will be due to 'temporary' causes and should not significantly compromise the accumulated damage values.

These considerations mean that there should be no reason, from the monitoring viewpoint, for any part of the monitoring system to be safety critical (e.g. D0178B class A or B software). However there are other considerations which apply to the elements of the airborne system, such as interaction with safety critical control systems, which will generally result in the airborne software being controlled to D0178B class C.

8.2 System Architectures

Processing of the data for the health monitoring functions may be best carried out on board because the results are required at the aircraft to help the maintenance crew clear the vehicle for the next mission. On the other hand the calculation processes for usage monitoring can be carried out either in the equipment installed on board the aircraft or in a ground station using data which has been recorded by the on board equipment during flight. The choice of system used on any particular project depends on the specific requirements of that project and the degree of interaction between the 'usage' monitoring and 'health' monitoring functions

With modern commercial capabilities for data storage and transfer (smart memory cards and ethernet data transfer standards) the preferred system for usage monitoring is to record data on board and carry out the analysis in ground. This approach minimises the airborne element of the system and thereby reduces the validation and configuration control work required. It also allows the recorded data to be re-analysed at a later date using revised techniques and material information.

8.3 System Elements to be Validated

The objective of the validation process is to demonstrate satisfactory operation of the system in all foreseeable circumstances. Having produced evidence that each element operates in accordance with the full specification requirements for that unit, further testing is required to demonstrate that the total system operates correctly in all foreseeable circumstances. The process needs to consider both hardware and software aspects. It should be noted that the Reduced Order Algorithms may be implemented in either the airborne and/or ground units depending in the architecture chosen. If these algorithms are implemented in the ground station then the airborne unit software is only required to control the data acquisition process. If these algorithms are implemented in the airborne unit then the basic software in the ground station is only required to handle the accumulated usage counts. However it is recommended that the ground station should have the ability to process raw data recorded from service to confirm the satisfactory operation of the system and to allow detailed analysis of any changes to the operating pattern.

8.3.1 Reduced Order Algorithms

The whole process depends on having a set of algorithms which can operate in a small computer and which give acceptable representation of the full life usage algorithms used on a mainframe computer. The first step in the validation process is to verify any set of Reduced Algorithms proposed for any specific application. Each individual simplified model must be calibrated against full modelling; e.g. full vs. simplified thermodynamic modelling (respectively thermal, mechanical, life modelling). Entire life modelling (full vs. simplified) must then be correlated for some flight missions. Ground and on-board engine life monitoring software must be extensively compared using synthetic and real flight profiles. The models introduced in the ELMS employ simplifying assumptions. These simplified models need to be adjusted to best represent the full model.

8.3.2 Initial calibration of the individual functions

Specific functions must be individually calibrated against full FE modelling. The coefficients are then adjusted so that the simplified modelling represents to a certain prescribed precision the behaviour of full modelling. This is performed for different flight profiles to guarantee that simplifications are nevertheless able to describe the typical situations encountered in service. Several flight profiles, synthetic and real, are used as input to life calculations using life monitoring algorithms and FE modelling to confirm that no extrapolation of data is needed for life calculations and to correlate the effects of authorised deviation incurred via the simplifications of ELMS.

8.3.3 Airborne Hardware and Software

The airborne hardware is basically a group of accessory unit data (speed, pressure and temperature sensors, wiring, avionics unit etc.). Their functioning in the environment for the application is validated using the same specifications as the rest of the units on the engine.

Standard approval process to DO 178B at the appropriate level depending on the overall architecture of the application. This will normally be at level C/D but there may be a need to clear it to a higher level if it is closely integrated with other software (e.g. engine control) which itself requires a higher level.

8.3.4 Data Transfer

This unit may be part of the airborne system and part of the ground station (if a smart card and reader) or it may be purely a piece of Ground Support Equipment. The validation process will depend on the system architecture and the methods should be appropriate to status of each item.

This includes, the number of aircraft which can be downloaded to the transfer unit without return to the ground station, storage capacity of airborne unit and transfer unit and the required frequency of downloads. It should also include methods for identifying the failure of a system component or a listed failure case.

8.3.5 Ground Hardware and Software

The 'validation' activity should be carried out in the selection process used to decide which unit to procure. Any specific testing should be based on the normal practice for ground support equipment used to handle data. The project specific software which implements the Reduced Order Algorithms should be subject to a Beta test to check that the algorithms have been implemented correctly. This Beta test should be carried out on the proposed operating system to ensure that the combination functions satisfactorily.

8.3.6 Systems

Successful operation of the usage monitoring system requires that the airborne and ground hardware units, airborne and ground software, data transfer equipment and procedures all operate effectively together. Therefore it is essential that a programme of tests is carried out on the total system once all the individual units have been qualified. A typical approach is to carry out a sequence of testing designed to cover demonstration that the system operates correctly over the full range of input data values covering all possible flight conditions and that all the 'loops' and 'witches' in the software function as intended. All that is then required is to demonstrate that the system operates correctly in real time with typical dynamic inputs and that it operates correctly in the real environment (EMI Weather, Service operation etc.)

8.3.7 'Laboratory' Tests

The purpose of these tests is to demonstrate that the total system functions over the full range of input parameters and that all the 'loops' in the software operate correctly. The testing requires a representative set-up of the total monitoring system. Ideally this set-up should include any other units from the total aircraft system which interact with the monitoring function (FADEC, Cockpit instruments etc.) This can be carried out on a powered mock-up of the system or as a ground test on the complete vehicle.

8.4 Examples of Application

The Agusta S129 helicopter engine has been selected to illustrate how the ELMS has been designed to conform to a specific usage specification. It focuses on validation process and the ease and cost of operating an engine life monitoring system. This helicopter has an Integrated Multiplex computing system which carries out several functions including aircraft and engine monitoring. The main requirement which led to this configuration was the need to operate for extended periods away from base and to output the calculated life consumed on the cockpit display screen at the request of the crew. The engine element of this system calculates the LCF life consumed for all the critical rotating parts and the creep life consumed for the gas generator turbine blades. It also monitors the usage of the rating structure to ensure that the time limits at each rating are not exceeded. Since the airborne processing was carried out in a 'universal' computing system which already carried out flight critical functions there was no need for any hardware testing specifically for the engine monitoring functions. There were 5 elements in the validation programme for the usage monitoring functions. These included

Validation of the simplified algorithm against full stress analysis results.

Validation of the implementation into the onboard units by testing in the systems laboratory using test cases which exercise all the different routes within the calculation process.

Flight testing in an aircraft with a data recording capability, comparing the results calculated by the monitoring system with those calculated using a ground system fed with data from the flight test recorder. where the ground system was programmed with the same algorithms as the airborne system. This indicated agreement within 0.5% between the two sets of results.

Re-analysis of a small sample of the flight test data in an independent ground system with algorithms implemented by a different software author.

Evaluation of a period of service operation on several aircraft to confirm that the correlation between the usage measured by monitoring system and the known usage of the aircraft was expected.

All these elements were applied to the LCF counting function. However, turbine blade failure on a multi-engine aircraft does not represent a flight safety hazard and therefore a reduced validation assessment was applied to the turbine creep monitoring function. The evaluation of the creep monitoring function revealed that for 'continuous accumulation' a monitoring system has to calculate the damage rate over small time intervals (less than 0.5 second).

Final Comments

In the drive to achieve maximum use of existing propulsion systems, it is essential that neither safety nor mission effectiveness levels are compromised. The report has shown that when choosing a design practice, life cycle cost analysis should consider fleet size and utilisation and access to inspection facilities.

Safe life practices offer lower maintenance, lower facilities costs associated with maintenance and do not require in-service inspection. This may be particularly attractive for smaller fleet sizes, however, in the event of life related problems arising in-service, fewer management options are available.

Airworthiness issues and the opportunity to use more of the available individual component lives may sway the selected design lifing procedure towards damage tolerance.

Usage monitoring systems have increased in complexity from the earliest time based systems. Where the aim is to achieve maximum use of the available component life, it is imperative that the latest sophisticated thermal transient lifing algorithms are utilised in usage monitors and that fleetwide monitoring is implemented.

Acknowledgements

This paper is based on the current status of the PEP WG28 activity 'Current Practice for Monitoring Gas Turbine Engine Life Consumption'. The author gratefully acknowledges the contributions of all his colleagues in Working Group 28, in particular the Chairman, Glen Lazier, Sverdrup Tech. Inc. and lead authors responsible for the specific chapters: 'Gas Turbine Component Life Consumption' Chris Eady, UK; 'Analysis Methods in Mechanics of Material Fracture' Wiaslaw Beres, NRC; 'Civil and Military Practices' Carl Meece P&W USA; 'Usage Survey and Mission Analysis' Juergen Broede MTU GE; Usage Results from Monitoring Systems' Mike Sapsard Implement UK; 'Component Usage Related to life Consumption' George Harrison DERA UK; 'Maintenance Policies for Fracture Critical Parts' Otha Davenport WPAFB USA; 'Validation Requirements' Pierre Mosser SNECMA FR. Within the restrictions of this report, it has been necessary to omit significant contributions of the working group, for this the author apologises.

Methods of Modern Lifing Concepts Implemented in On-Board Life Usage Monitoring Systems

Jürgen Broede and **Manfred Köhl**
MTU Motoren- und Turbinen-Union München GmbH
Dept. TPMM
P.O. Box 50 06 40
80 976 Munich
Germany

1. ABSTRACT

Modern lifing concepts for fracture critical parts include safe crack initiation life and safe crack propagation life. In military aero engine applications, a portion of the safe crack propagation life is more and more utilised to extend the usage period beyond the limits set by the concept of safe crack initiation life. To really use the benefits of such life extension without reduction of flight safety, it is essential that all engines involved are monitored with respect to individual life consumption. Thus, on-board life usage monitoring systems need to address the crack propagation phase in the same manner as the crack initiation phase.

It is shown how the calculations of fracture mechanics parameters and the resulting crack propagation process are integrated into the algorithms of on-board life usage monitoring software. Applicability of the methods is underlined by results obtained with OLMOS - the on-board life usage monitoring system of the German Tornado fleet.

2. INTRODUCTION

Aero engine fracture critical parts undergo cyclic loading during their operational usage. Due to the nature of this loading, the material experiences fatigue at some highly stressed areas, what leads to initiation and propagation of fatigue cracks.

Fatigue is the life limiting damage mechanism for most of the fracture critical parts, since growth of cracks beyond a certain stage of their development (in other words: beyond a certain crack depth) increases the probability of part failure. In order to maintain the structural integrity of the engine in accordance with the required safety standards, it is necessary to retire a part before an accepted risk level is exceeded. This means that a fracture critical part has got a limited service life.

Fatigue life of a part is defined as a number of cycles (with a given stress range at given temperature conditions) that the life limiting critical area of that part is able to endure until a crack with specified properties has developed.

3. MODERN LIFING CONCEPTS

The classical method to treat fatigue life is the concept of safe crack initiation life (frequently just called: the safe life concept). 'Safe' means in this context that the scatter of material strength is considered and the life of the weakest part of the whole population declared as the life of each member of this population. The consequence of this concept is that only the fewest parts will have generated a small fatigue crack when they all are taken from service. The majority of the parts will be crackfree. And even parts which have developed an initial crack will still have a remarkable portion of remaining life, since in most of the cases the crack can be allowed to grow to some certain crack depth before the engine integrity will be jeopardised.

This behaviour led to the idea to safely utilise some portion of the crack propagation phase. The underlying lifing concept is called the concept of safe crack propagation life. This concept is equivalent to that of the safe crack initiation life, but allows instead of roughly 0.4 mm depth a fatigue crack to grow until some other safety criterion is reached. This criterion is called dysfunction. It includes a number of different cases which all could reduce the structural integrity. Of course - the concept does not accept that the dysfunction condition will be really reached, but provides a well defined safety margin. Due to this real safety margin, the concept of safe crack propagation may be judged as 'safer' than the classical safe (crack initiation) life concept.

In military aero engine applications both these concepts are combined, where the safe crack propagation life is used to extend the initially released crack initiation life.

Another approach - also applicable to military aero engines - is to use the safe crack propagation life alone, thereby ignoring the crack initiation phase. But in this case crack propagation is assumed to start at significantly smaller crack depths (around 0.1 mm). This approach is typically applied for parts made of high strength powder material or for parts of conventional metal if they are loaded beyond the traditional stress levels.

4. SAFE CRACK INITIATION LIFE

The basic idea for the concept of safe crack initiation is that

- a new part is free of defects
- a defect (in this case a fatigue crack) is generated in service
- the part's life is expired, when the defect has been created

The criterion for 'existence' of a crack is that the crack has been initiated and grown to a certain depth. A commonly used value for this crack depth is 0.4 mm. This criterion is a little arbitrary, although sensibly based on long experience.

The safe crack initiation life is established as the number of cycles to reach an accepted statistical probability for the existence of a crack with that depth. The statistical probability takes into account that material strength exhibits some scatter. And for the weakest individuum of the parts' population, the structural integrity must be ensured. Generally accepted

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

statistical probabilities for that weakest part are in the range of 1 out of 750 to 1 out of 1000.

Parts lifed under this concept are not inspected whether the crack is really present when they are retired, and only the fewest of them would contain one. Nevertheless, re-use of the parts beyond the safe life - as defined above - is not considered.

The criterion of this concept - namely the existence of a crack - does not mean that the part will fail immediately when the accepted crack depth is exceeded. Some safety margin to final part failure will remain. However, the concept cannot provide a measure for the real safety of the critical part, since it is unable to predict a value for the failure margin. In most of the applications there will be sufficient margin for a crack to grow to part dysfunction, but it is also possible that the dysfunction life is very close to the crack initiation life. In such very rare cases the crack initiation life cannot be considered as really safe.

5. SAFE CRACK PROPAGATION LIFE

The concept of safe crack propagation life is based on the idea that

- the part contains an initial defect at the beginning of the crack propagation phase (where the defect behaves like a crack of a certain depth)
- the crack propagates under service loading
- the part's life is expired when the crack enters the phase of part dysfunction

Part dysfunction may include a number of different criteria, for example

- unstable crack growth under basic operational loading
- onset of continuous crack propagation due to superimposed vibratory stresses (i.e. if high frequency stress levels exceed the crack growth threshold)
- loss of overspeed capability (i.e. a crack depth where overspeed conditions could cause spontaneous failure)
- unacceptable out-of-balance conditions

In contrast to the crack initiation criterion, the dysfunction criterion really determines the end of the part's life. This enables us to define a measurable safety margin. Thus, the part will be taken from service when a certain portion (e.g. two third) of the number of cycles to dysfunction have been accumulated.

If the concept of safe crack propagation life is used for life extension, the number of cycles to dysfunction encompasses both the crack initiation and the crack propagation phase. The safety factor of two third will then be applied to the total number of cycles.

The safe crack propagation life is established as 2/3 of the number of cycles to reach an accepted statistical probability for the presence of the applicable dysfunction condition. The statistical probability takes into account that material strength and crack growth properties exhibit some scatter. For the weakest individuum of the part's population, the structural integrity must be ensured. Generally accepted statistical probabilities for that weakest part are in the range of 1 out of 750 to 1 out of 1000.

Parts lifed under the safe crack propagation lifing concept are not inspected for cracks when they are retired. Re-use of the parts beyond the safe crack propagation life is not considered, although most of the parts will not contain a crack grown to the depth which is correlated with the dysfunction criterion.

6. PRINCIPLES OF LIFE USAGE MONITORING

Fracture critical parts in aero engines are released only for limited life. They must be retired from service when their life limits are reached. Life usage monitoring activities serve to identify the proper time. How long a critical part can be kept in service, depends on both the released life at the critical areas and their life consumption due to operational usage. Different methods for life usage monitoring have been established.

The traditional method is to count the engine flight time and to multiply it with a cyclic exchange rate. The cyclic exchange rate (also called β-factor) provides a relationship between the flight time and the life consumed at a critical area. But the correlation between flight time and cyclic life consumption is very weak. This means that conservatism needs to be incorporated into the β-factor, what in turn leads to overestimation of life consumption for most of the parts.

In reality, the life consumed during an engine run or flight is based on stresses and temperatures at the critical areas of the components. These parameters depend obviously on the actual mission profiles, engine intake conditions, individual pilot reactions and many other influences. Thus, one can conclude that better exploitation of the released life is achieved with individual monitoring, where life consumption of each part is individually calculated using actually measured engine parameters.

The procedures for individual monitoring consist of effective algorithms for use in real time, able to calculate the consumed life directly from measured engine signals. The algorithms allow for fast transition of the input signals as they appear under real aircraft and engine manoeuvring. Results are available immediately after the end of a flight. Life usage is measured in damage related physical or technical units.

Details of the method have been published at several occasions [1-5]. Here only a summary is given. The method determines the thermal and mechanical boundary conditions for the engine components on the basis of measured time histories of engine operating parameters (such as spool speeds, intake conditions and gas path temperatures and pressures). Based on these boundary conditions, the transient temperature development within the components is calculated. Stresses or strains at critical areas are computed, which are then used together with the corresponding temperature histories to predict the related damage. Critical area damage is accumulated over all engine runs, so to build up complete life consumption records for all monitored parts of an engine.

In order to ensure undisturbed operation of the monitoring algorithms in on-board life usage monitoring systems, the input data are checked for plausibility. Range and rate checks are applied to all input signals. Additional cross checks are performed based on relationships between signals which exhibit sufficient correlation. If data are found faulty, corrective actions are taken trying to restore them. Interpolation of the signal is used over short drop-out periods. If the signal fails for longer periods, substitutes derived from other valid input signals are taken. If no model for signal substitution is available or too many signals fail simultaneously, the life usage monitoring process is stopped and the need for further corrective actions is flagged.

The monitoring results are checked for plausibility at the end of each engine run. If the results are implausible, particularly if faulty input signals have not allowed to complete the monitoring process, an estimate of life consumption is made for the current engine run based on the flight time or - for ground runs - converted engine run time.

The procedures for individual life usage monitoring outlined here (which are basically also applied in the process of determining β-factors for traditional life usage tracking) are closely related to the life prediction process which is part of the entire engine development process. To show how the life usage monitoring algorithms - particularly those parts related to life consumption in the crack propagation regime - are derived from the life prediction process, a short overview over this process is given.

7. LIFE PREDICTION PROCESS

The life prediction process starts from the design mission which is usually defined in the engine specification. The design mission provides the required thrust and power as a function of time. After the engine hardware has been designed, the part geometry and the materials are defined.

In a first step, there are engine performance parameters derived from thrust and power requirements. The performance parameters consist of the temperatures and pressures in the main gas stream, the spool speeds and shaft loads and torque for each point of the design mission. Additionally, the cooling air flows, temperatures and pressures in the secondary air streams are determined.

In a second step, these performance and cooling system parameters are used as boundary conditions for the calculation of transient temperature distributions in the engine components.

The third step is concerned with the mechanical analysis. Total stresses are calculated as sum of centrifugal stresses (due to part rotation), thermal stresses (induced by temperature gradients), stresses from pressures, shaft forces and torque and of assembly stresses. Based on the results of the stress analysis, the critical areas of the engine parts can be identified. Critical areas are those areas which are exposed to the highest stresses and largest stress ranges, and which can be expected to determine the fatigue life of the part.

Finite element programs are employed for temperature and stress analysis. The engine rotor systems, which contain the majority of the fracture critical parts, are of particular interest. Since the rotating parts are mainly axi-symmetric, a 2D analysis is usually sufficient. Disturbances of the axi-symmetry (caused by holes and scallops in flanges, arms and cones) are treated with stress concentration factors. Stress concentration factors are also applied for other areas where the FE mesh might not be fine enough. All load cases of the design mission are investigated.

The stress concentration factors are either taken from text books or determined by a 3D detail analysis. The 3D detail analyses are only performed for a limited number of load cases.

The second and third step together provide stress-temperature histories at the critical areas over the entire design mission. The stress-temperature histories are analysed with respect to their cyclic content. The most damaging cycle is identified and declared as the reference cycle for the considered critical area.

For most of the critical areas, the concept of safe crack initiation life is employed. Under this concept, the number of reference cycles needs to be predicted, which the critical area can undergo until a fatigue crack will have been generated and grown to the predicted depth of 0.4 mm. Material SN-curves are used which describe the relationship between applied stress range and the corresponding number of cycles to crack initiation. Mean stress and temperature effects are covered as well.

For critical areas where the safe crack propagation life concept alone is applied or where the original safe crack initiation life is extended into the safe crack growth regime, additionally the safe crack propagation life needs to be predicted. For this prediction, typically the methods of linear elastic fracture mechanics are used. Two things are required, namely the stress intensity factor range of the most damaging cycle accompanied by R-ratio (which is the ratio of the cycle minimum stress intensity factor to the cycle maximum stress intensity factor) and temperature, and the crack propagation law relevant for the used material. The crack propagation law describes the crack propagation rate as function of the stress intensity factor range, considering also R-ratio and temperature. The accumulating crack growth process is simulated by integration of the crack propagation rate from the initial crack depth to the onset of instability under reference cycle loading. From the number of cycles necessary to propagate the crack up to the dysfunction criterion, one can derive the safe crack propagation life.

8. GEOMETRY FUNCTION

For prediction of the safe crack propagation life and also for life usage monitoring in the crack propagation regime, it is necessary to determine the stress intensity factor. The stress intensity factor depends on the geometry of the component, on the stress field around the critical area, and on the current shape of the crack. As these quantities are very complex, text book solutions for the stress intensity factors are generally not available. Thus, they are calculated by finite element analyses (or on the basis of experimental results, see below).

The crack shape is given by the crack surface (which in many cases may be considered as a plane) and the crack front. It is essential to consider the crack as a whole. In particular, it should be noticed that the stress intensity factor varies along the crack front, so that the crack growth velocity is different at different points of the crack front.

A 3D finite element model is used, where the crack surface and the crack front are introduced. As an example, in Fig 1a,b a modelled crack front at a critical area in the rim slot fillet of a turbine disc is shown. An automated procedure has been developed. Details of this technique were already published in [6-10]. With this procedure, the development of the crack is simulated using finite element calculations in combination with an appropriate crack growth law.

The procedure starts with an initial crack. The initial crack shape is either taken from test experience with real parts or simply assumed as a half or quarter elliptical front at a plane perpendicular to the direction of the maximum principal stress. The exact shape of the initial crack is not so important as the crack will develop into a balanced shape anyway.

For the part containing this initial crack, the stress intensity factor along the crack front is calculated. Typically, the load case belonging to the maximum stress of the reference cycle is used. All relevant loads (as centrifugal and thermal loads) contributing to the stress field around the critical area are included. With the stress intensity factor range and the crack propagation law, the crack growth increment at each point of the crack front is calculated and a new crack shape predicted. Since the crack develops slowly, the stress intensity factor will not change significantly during a small number of cycles. Thus the crack growth for a number of cycles can be computed with the same stress intensity factor distribution. But when the grown crack is distinctly different from the original one, a new stress intensity factor (SIF) calculation becomes necessary. This process (calculation of the SIF, determination of the according crack growth rate and prediction of the new crack shape) is repeated several times building up a complete crack growth history over the number of applied cycles. This crack growth history ranges from an initial crack (which is usually smaller than or equal to that for the crack initiation criterion) to the onset of unstable crack growth.

For this procedure it is assumed that the stress fields around the critical area are proportional for all load cases of the mission. In fact, this is not the case. But the error is considered negligible as long as the stress intensity factors for the higher stress levels are modelled correctly. Deviations for sub-cycle stress intensity factor ranges are acceptable as their contribution to the overall damage is small. However, if it turns out that the inaccuracy becomes intolerable, an additional influencing parameter (e.g. the stress gradient) needs to be incorporated into the geometry function.

Currently only a correction for residual stresses is made. The basic idea for this correction is that during the first load cycles some local plastification occurs, what causes some redistribution of the stress fields, as illustrated in Fig 3. After initial plastification, the component is assumed to behave linearly, so that the application of linear elastic fracture mechanics methods appear adequate. This redistribution is

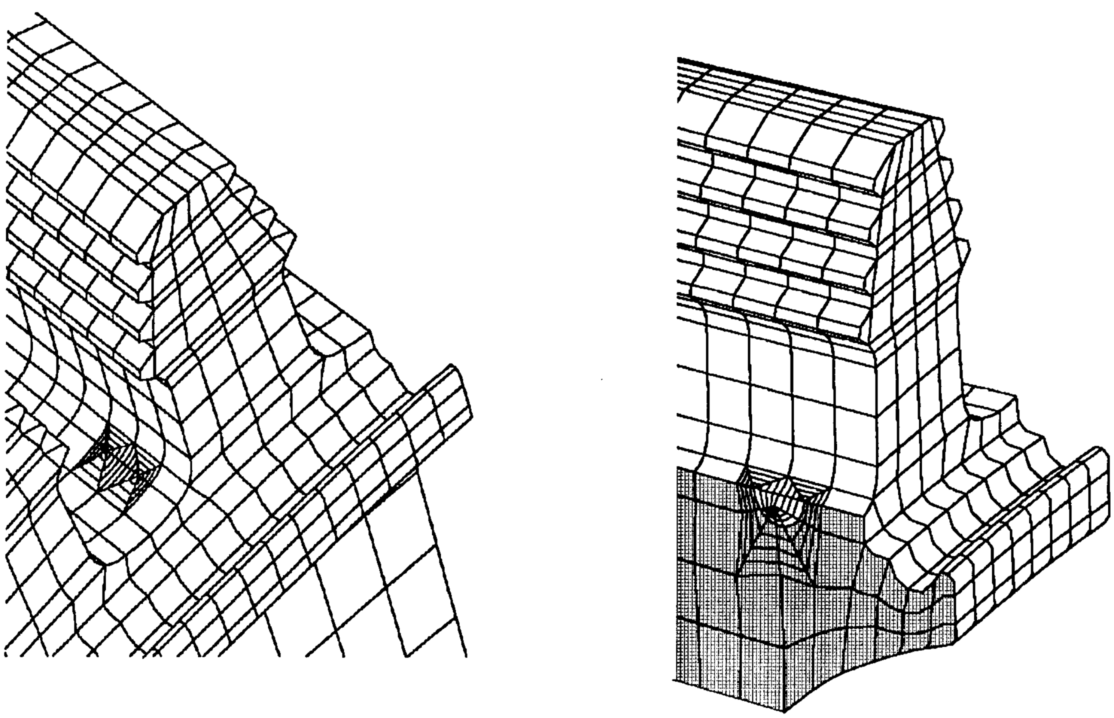


Figure 1: Remeshed FE model of a turbine disc in the vicinity of an introduced crack.

a) View at disc surface

b) View at crack plane

accounted for by an additional additive term in the formula for the stress intensity factor calculation.

Now, one can choose a path at the crack surface from the point where the crack has originated into the depth of the part, as shown in Fig 2. Along this path the crack depth is measured. For this path, a relationship between the crack depth and the corresponding stress intensity factor can be established. Dividing the stress intensity factor by the stress present at the uncracked critical area itself, defines a so called geometry function. This geometry function provides the relationship between the stress at the critical area and the stress intensity factor at the crack front for each value of the crack depth and enables us to calculate the time history of the stress intensity factor from the time history of the stress at the critical area including the effect of increasing crack depth.

The just verbally described procedure to determine the geometry function $g(a)$ and the additional additive term $K_{add}(a)$ as a function of the crack depth a is now summarized :

Firstly, a 3D finite element analysis of the uncracked structure under reference load conditions (assumed as extreme load case also in operational usage) is performed, linear-elastically as well as elastic-plastically, in order to determine the linear-elastic stress $\sigma_{elastic,ref}(a)$ and the elastic-plastic stress $\sigma_{plastic,ref}(a)$ along the crack path a. The difference defines the residual stress

$$\sigma_{residual,ref}(a) \;:=\; \sigma_{plastic,ref}(a) \;-\; \sigma_{elastic,ref}(a). \qquad (1)$$

The effective stress $\sigma(a)$ in the uncracked structure can now be formulated as

$$\sigma(a) \;=\; \sigma_{elastic}(a) \;+\; \sigma_{residual,ref}(a)\,, \qquad (2)$$

where $\sigma_{elastic}(a)$ is approximated by the monitored elastic stress $\sigma_{elastic}(0)$ at the critical area (a=0), scaled by the respective stresses produced under reference load conditions :

$$\sigma_{elastic}(a) \;:=\; \sigma_{elastic}(0) \;\cdot\; \sigma_{elastic,ref}(a)\,/\,\sigma_{elastic,ref}(0)\,. \qquad (3)$$

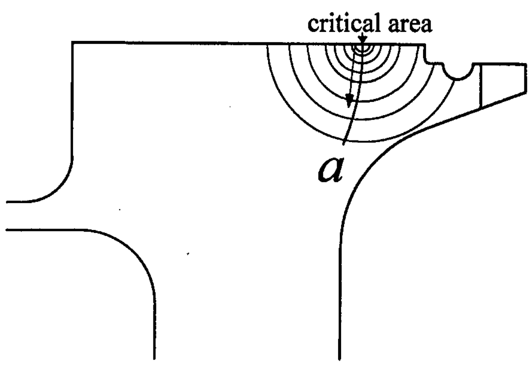


Figure 2: Analysed crack fronts and crack path a

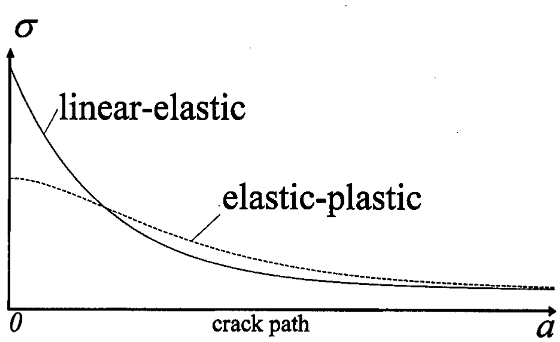


Figure 3: Stress redistribution due to plastification

Secondly, a couple of linear-elastic 3D finite element analyses of the structure containing a crack of different depths a according to Fig 1 and Fig 2 is performed, again under reference load conditions, yielding the stress intensity factor $K^*_{ref}(a)$. The stress intensity factor can be represented as the product of the local stress $\sigma_{elastic,ref}(a)$ and a geometry factor $g^*(a)$. Thus, we obtain

$$g^*(a) \;:=\; K^*_{ref}(a)\,/\,\sigma_{elastic,ref}(a)\,, \qquad (4)$$

and the stress intensity factor $K(a)$ due to an arbitrary local stress $\sigma(a)$ can be written as

$$K(a) \;=\; \sigma(a) \;\cdot\; g^*(a)\,. \qquad (5)$$

The aim, finally, is to find a representation of $K(a)$ as a function of the known (since monitored) elastic stress $\sigma_{elastic}(0)$ at the critical area (a=0) :

$$K(a) \;=\; g(a) \;\cdot\; \sigma_{elastic}(0) \;+\; K_{add}(a)\,. \qquad (6)$$

This formulation defines the desired geometry function $g(a)$ as well as the additional additive term $K_{add}(a)$ due to residual stresses. After substituting equations (2), (3) and (4) into equation (5), comparison with equation (6) yields

$$g(a) \;:=\; K^*_{ref}(a)\,/\,\sigma_{elastic,ref}(0)\;, \qquad (7)$$

$$K_{add}(a) \;:=\; \sigma_{residual,ref}(a) \;\cdot\; K^*_{ref}(a)\,/\,\sigma_{elastic,ref}(a)\,. \qquad (8)$$

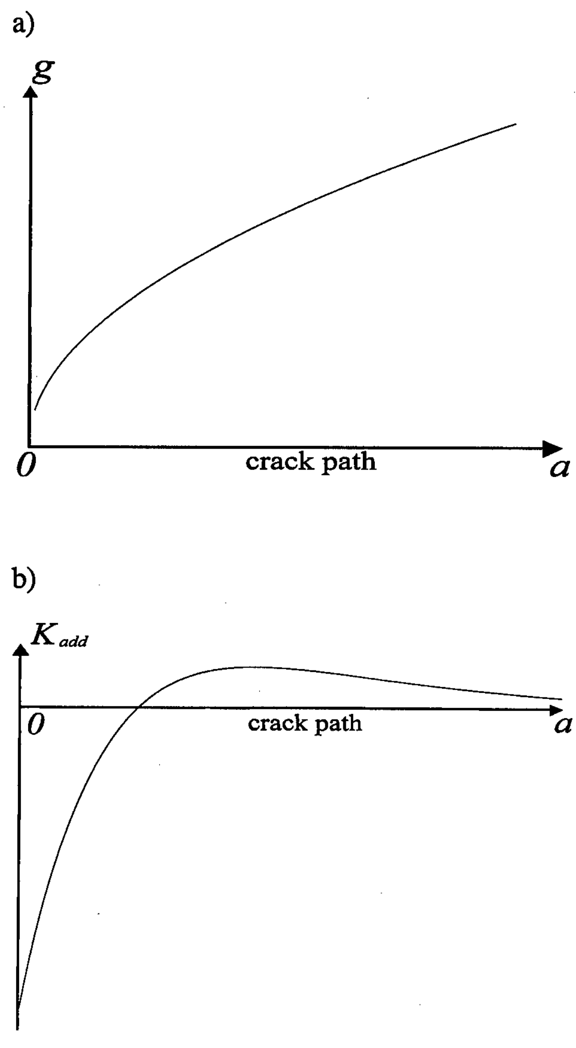


Figure 4: Geometry function g and additive term K_{add}

Fig 4 shows a sketch of these quantities as functions of a. The geometry function $g(a)$ (Fig 4a) increases usually monotonically with the depth a. The additive correction $K_{add}(a)$ (Fig 4b) starts with a negative value at the critical area ($a=0$), increases with increasing depth a , and becomes slightly positive. This behaviour of $K_{add}(a)$ is caused by a stress redistribution after plastification as already illustrated in Fig 3. In the chosen example, the highest plastification occurs at the critical area ($a=0$), reducing the stress intensity factor, while - as a static balance - in deeper, not plastified regions the level is greater compared to purely elastic results. Up to what depth values a the level of the stress intensity factor is increased depends on the size of the plastified region and on the amount of plastification.

If the geometry function $g(a)$ and the additive term $K_{add}(a)$ shall be used for the prediction of experimental test results rather than for engine monitoring, the quantities occurring in equations (1) to (8) have to be derived under test load conditions. The equations still have the same form, only the index "*ref*" needs to be replaced by "*test*".

There is another way to determine the quantity $K^*_{test}(a)$ which is needed for the evaluation of $g(a)$ and $K_{add}(a)$ in (7) and (8). This way avoids the finite element calculation of the cracked structure and is based on experimental results. It is presumed that for the considered critical area the crack depth a is observed and recorded as a function of accumulated test cycles N. The material's crack growth law and the 3D finite element results $\sigma_{elastic,test}(a)$ and $\sigma_{plastic,test}(a)$ of the uncracked structure must also be known.

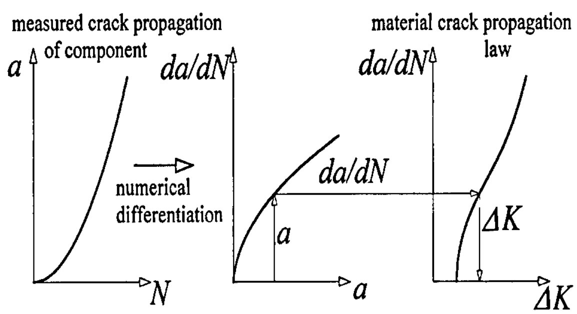


Figure 5: Procedure to determine the geometry function by test results

Fig 5 illustrates the procedure. The crack depth a versus the number of cycles N (left diagram) is differentiated to obtain the crack growth rate da/dN (middle diagram). The crack growth rate can be considered as a function of the crack depth a or as a function of the number of cycles N. Based on the crack growth law (right diagram), one can determine the stress intensity factor range ΔK corresponding to the current crack growth rate da/dN. Thereby, it is assumed that the R-ratio for the stress intensity factor is the same as for the stress cycle at the critical area itself (i.e. at $a=0$). The test evaluation yields the quantity $K(a)=K_{test}(a)$ subject to test load conditions. Using equation (6), written for the test load $\sigma_{elastic}(0)=\sigma_{elastic,test}(0)$, and equations (1),(7) and (8),

$$K^*_{test}(a) \;=\; K_{test}(a) \cdot \sigma_{elastic,test}(a) \,/\, \sigma_{plastic,test}(a) \qquad (9)$$

can be derived, which is needed for the evaluation of $g(a)$ and $K_{add}(a)$ in (7) and (8) where the index "*ref*" is replaced by "*test*".

In case of a dominating elastic behaviour of the structure (only small plastification), the test based evaluation of $K_{test}(a)$ is also applicable for the engine monitoring. In this case $K_{add}(a)$ is negligible, and the geometry factor can be directly obtained from

$$g(a) \;:=\; K_{test}(a) \,/\, \sigma_{elastic,test}(a) \,. \qquad (10)$$

One should be aware that this is an approximation which assumes a proportionality of the stress fields around the critical area between reference and test load conditions which not in all cases can be fully achieved.

The geometry function g - irrespective of the way it has been established - and K_{add} can be interpreted either as a function of the crack depth (as above) or as a function of the number of reference cycles applied. For given reference conditions and a given initial crack depth, there exists a direct correlation between the accumulated number of applied cycles N and the crack depth a (to obtain by integration of the crack propagation law, see lower diagram in Fig 6). This means that both depictions are equivalent. If the geometry function is derived from test results and shall be drawn as a function of cycles, one should be aware that tests are very often performed under overload or with temperatures different from those of the engine reference conditions. Corresponding corrections have to be made.

Nevertheless, for the purpose of life usage monitoring it is preferred to formulate the geometry function as a function of the number of accumulated reference cycles.

9. CRACK PROPAGATION MODEL FOR LIFE USAGE MONITORING

It is the intention that the crack propagation model can be directly integrated into the structure of existing life usage monitoring systems. This can be easily achieved if the life usage monitoring systems are build in a modular architecture.

With such a constellation, only the damage module is concerned, and in this module only the conversion from stress-temperature cycles into the corresponding damage increments. Admittedly, additional input information is required.

Under the concept of crack initiation life, the damage accumulation process is assumed to be a linear process. The damage increments are independent from the current state of accumulated damage and the Miner's Rule is used.

The damage accumulation process in the crack propagation regime - in contrast - is a non-linear one. The damage increment depends additionally on the currently accumulated stage of damage. The physical representation of the accumulated damage is the crack depth, but in the terminology of life usage the number of consumed reference cycles is preferred. The current stage of damage determines on one hand the value of the geometry function and on the other hand the crack growth increment of the reference cycle which is internally used as reference.

If the concept of safe crack propagation life is used to extend the life beyond the safe crack initiation life, then it is necessary that the algorithm switches from one procedure to the other controlled by the current stage of cumulated damage. To distinguish between both procedures, it is checked if the already consumed number of cycles is below or above the released number of cycles to crack initiation. If it is below, then the crack initiation damage process applies, otherwise the crack propagation process.

We know that the life usage monitoring process for aero engines is strongly related to the history of an engine run [1]. In particular, each engine run is treated separately and at its end all cycles are closed and the damage accounts are updated. Since an engine run can be considered as short compared to the life of fracture critical parts and the cracks at critical areas grow slowly, we can assume that the current state of damage is nearly constant during one engine run. This allows us to determine the values of the damage dependent parameters only once per engine run, namely at its beginning.

The functionality can be specified as follows:

- Both the geometry function and the inverse of the reference cycle crack propagation increment (which is the quantity really needed) are given as functions of the accumulated number of cycles. Usually a representation in form of tables is used where the current values are obtained by linear interpolation.
- As part of system and algorithm initiation at the beginning of an engine run, it is checked for the respective critical areas whether the crack propagation regime has been entered. If yes, then the values of the geometry function and the inverse of the reference cycle crack propagation increment are determined. They are kept constant for the whole engine run.
- During the main calculation steps and also within the final calculation, all extracted stress cycles are converted into stress intensity factor cycles using the geometry function. The corresponding crack propagation increments are calculated by evaluation of the crack propagation law according to stress intensity factor range, R-ratio and temperature. Multiplying the respective crack propagation increments with the inverse of the reference cycle crack propagation yields the damage of the considered cycle in terms of the relevant life consumption units (i.e. multiples of reference cycles). Damage increments are accumulated over the whole engine run.
- In the final calculation phase - i.e. after the engine has been switched off - the damage accumulated over this engine run is added to the damage accounts, provided the result checks have been passed.

With this procedure it is ensured that life usage monitoring in the crack propagation phase is completely equivalent to that of the crack initiation regime, and that both lifing concepts can be commonly applied.

10. DETERMINATION OF β-FACTORS

The β-factor (or cyclic exchange rate)

- provides the relation between cyclic life consumption and flight time
- serves to monitor life consumption if there is no on-board monitoring system installed
- serves to fill gaps in life consumption history where the on-board monitoring system is unable to provide correct data

Before defining the β-factor, the relative cyclic damage

$$D_{cycle} := \textit{(damage of cycle) / (damage of reference cycle)}$$

is introduced. This ratio is evaluated by different expressions for the crack initiation and the crack propagation phase. Denote the number of cycles to crack initiation with respect to a given stress cycle by N_{cycle} and with respect to the reference cycle (e.g. the main cycle of the design mission) by N_{ref}. For the crack initiation phase, the damage of a cycle is given by $1/N_{cycle}$ and the damage of the reference cycle by $1/N_{ref}$, thus the relative cyclic damage is $D_{cycle} = N_{ref} / N_{cycle}$. For the crack propagation phase, the damage of a cycle is the crack propagation rate $(da/dN)_{cycle}$ and the damage of the reference cycle is $(da/dN)_{ref}$, yielding the relative damage $D_{cycle} = (da/dN)_{cycle} / (da/dN)_{ref}$.

The β-factor is defined as the sum of the cyclic damage D_{cycle} for each stress cycle of a representative number of flights, divided by the accumulated flight time taken in hours :

$$\beta := \Sigma D_{cycle} / \Sigma \textit{(hours of flight time)}$$

The β-factor indicates the average damage per flight hour expressed in terms of the number of reference cycles.

The cyclic damage accumulated over all cycles and subcycles of an individual flight can be expected to be higher in the crack propagation regime than in the crack initiation phase. The following effects are expected:

- the main cycle damage is approximately equal to 1 (assuming main cycle similar to reference cycle)
- the subcycle damage in the crack propagation regime is higher due to different slopes of SN curve and crack propagation law
- more damaging subcycles exist in the crack propagation regime since the crack propagation threshold is relatively lower than the endurance limit in the crack initiation phase
- The ratio of cyclic damage of crack initiation to crack propagation strongly depends on the flight mission profile

Considering these effects, one may expect higher scatter in flight to flight damage for the crack propagation phase. In the following, the damage evaluation of the critical area according to Fig 1, performed for a representative number of real flight missions, will show whether the expected effects occur. Before, a description of the procedure is given.

Since the damage for the crack propagation phase is a function of the crack depth a (or, equivalent, of the applied reference cycles N), the cumulated damage is calculated according to the method outlined in section 9 by assuming constant damage over a period of one flight. The necessary auxiliary parameters ($g(a)$, $K_{add}(a)$ and $1/(da/dN)_{ref}$) are updated at the beginning of every flight.

A fast procedure to simulate the damage accumulation of the total component life is illustrated in Fig 6. The upper diagram is the result of a selected number of flight damage calculations for different crack depths (D versus a). The lower diagram shows the relation between crack depth and accumulated damage due to repeated reference cycle loading (a versus N). This relation results from integrating the material crack propagation law. In order to obtain the cumulated damage, these two relations are combined. Starting from an initial crack depth a_0, the upper diagram gives the damage increment per flight D_1 which corresponds to a number of applied reference cycles $(N_1\text{-}N_0)$, yielding the new crack depth a_1 by virtue of the lower diagram. Inserting a_1, the upper diagram gives the corresponding flight damage increment D_2, and the iteration process continues.

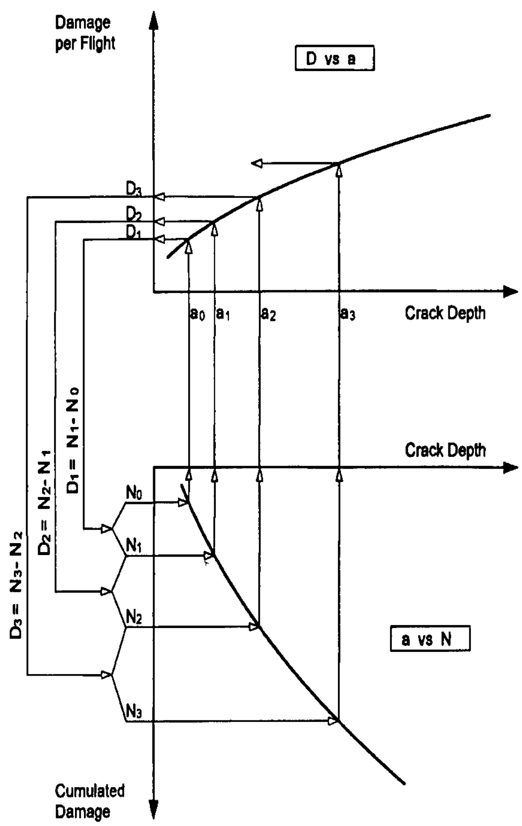


Figure 6: Procedure to simulate the damage accumulation

This procedure was applied to the critical area in the rim slot fillet according to Fig 1 for a representative number of real flight missions. As a result, the crack propagation β-factor β_{prop} turned out to be greater by a factor between 1.5 and 2.0 compared with the crack initiation β-factor β_{init}. This effect was expected (see above). The effect of an increased scatter of the damage between different flights was observed, but turned out to be smaller than expected.

Finally, the question on the benefit gained by the introduction of the safe crack propagation life concept can be treated by comparing the crack initiation life consumption with the crack propagation life consumption over a representative number of flight missions.

To quantify the benefit due to the extension of the safe crack initiation life concept to the safe crack propagation life, we refer to the introduction of the 2/3 dysfunction life (cf. section 5) which is equal to 2/3 of the number of reference cycles up to the dysfunction criterion (e.g. unstable crack growth), including crack initiation and propagation phase. Let N_{init} and N_{prop} be the number of cycles in the respective phase. If we divide these numbers by the respective β-factor β_{init} or β_{prop}, the respective life times in flight hours are obtained. Thus, we are able to compare the flight hours in the crack propagation phase with those in the crack initiation phase. The ratio yields a measure for the benefit gained by the inclusion of the crack propagation regime.

In the example shown in Fig 1 (rim slot fillet of a turbine disc) the service period in terms of engine flying hours can be increased by about 40% (corresponding to $\beta_{prop}/\beta_{init}=2$), if in addition to the crack initiation regime a safe percentage of the crack propagation regime is utilised.

11. CONCLUSION

Algorithms to monitor life consumption in the crack propagation regime have been developed. They are formulated in such a way that they are compatible with already implemented formulas for crack initiation monitoring, in particular they are also formulated on the basis of reference cycles rather than on crack dimensions. The example of a critical area in the rim slot fillet of a turbine disc shows that the service period in terms of engine flying hours can be increased by about 40% if in addition to the crack initiation regime a safe percentage of the crack propagation regime is utilised.

12. REFERENCES

1. Broede, J., "Engine Life Consumption Monitoring Program for RB199 Integrated in the On-Board Life Monitoring System", AGARD CP 448, Quebec 1988, Paper 9.
2. Broede, J. and Pfoertner, H., "Advanced Algorithm Design and Implementation in On-Board Microprocessor Systems for Engine Life Usage Monitoring", 15th Symposium AIMS, Aachen 1989, DLR-Mitt. 90-04,1990.
3. Dhondt, G. and Möhres, W., "Modelling the Engine Temperature Distribution between Shut Down and Restart for Life Usage Monitoring", 16th Symposium AIMS, Munich 1991, DLR-Mitt. 92-01, 1992.
4. Pfoertner, H. and Roß, C., "Preparing Life Usage Monitoring for the Next Decade", 18th International Symposium AIMS, Stuttgart 1995.
5. Broede, J. and Pfoertner, H., "OLMOS in GAF MRCA Tornado - 10 Years of Experience with On-Board Life Usage Monitoring", 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Seattle, WA 1997.
6. Dhondt, G., "Zyklische Mode-I Rissfortschrittsberechnungen mit ABAQUS", Proceedings of the German ABAQUS User's Conference, Innsbruck, Sept. 1997, pp 1-17.
7. Dhondt, G., "Zyklische Mode-I Rissfortschrittsberechnungen in Komplexen Bauteilen mit der Finite-Elemente-Methode", Proceedings of the DGLR Jahrestagung, Munich, 14-17 October 1997, pp 381-390.
8. Dhondt, G., "Automatic 3-D Mode I Crack Propagation Calculations with Finite Elements", Int. J. Num. Meth. Engng., *41*, 1998, pp 739-757.
9. Dhondt, G., "Cutting of a 3-D Finite Element Mesh for Automatic Mode I Crack Propagation Calculations", *to appear in* Int. J. Num. Meth. Engng., 1998.

10. Dhondt., G., "Automatic Three-Dimensional Cyclic Crack Propagation Predictions with Finite Elements at the Design Stage of an Aircraft Engine", *to appear in* the Proceedings of the "Applied Vehicle Technology Panel Symposium on Design Principles and Methods For Aircraft Gas Turbine Engines" of the RTO, Toulouse, France, 11-15 May 1998, Paper 33.

THE CONTRIBUTION OF HEALTH AND USAGE MONITORING SYSTEMS TO CALCULATIONS OF DAMAGE STATE AND FUTURE LIFE OF HELICOPTER COMPONENTS UNDER SAFE LIFE AND DAMAGE TOLERANT DESIGNS

P. E. Irving
R. A. Hudson
Cranfield University
Cranfield
Beds MK43 OAL, UK

SUMMARY

Techniques for calculation of fatigue damage on rotor components of helicopters are assessed and investigated using service flight load data. Fatigue damage for the dogbone blade linkage on the GKN-Westland Lynx has been calculated from loads measured during flight for a range of manoeuvres. It is found that variability of non -zero damage for individual manoeuvres is between 100 and 1,000 times. The difference in damage per flight hour between worst case damage and an upper limit for the most probable damage is a factor of 7.5. The contribution to damage of the loads scatter factor of 1.2 is quantitatively assessed. Finally the proportions of damage which occurs during a flight within manoeuvres and that which is between manoeuvres has been investigated. The results are discussed for both flight load synthesis and flight condition recognition approaches to indirect FUMS techniques

1 INTRODUCTION

Health and usage monitoring systems make separate and distinct contributions to calculations of damage state of aircraft components. Health monitoring in its simplest form provides data on whether or not damage exists. In more sophisticated implementations health monitoring systems may identify the location and measure the extent of damage (1). It may thus be viewed as a form of continuous in-service non destructive testing. Health monitoring provides no data on damage growth beneath the threshold of detectability, and prognosis on damage growth is limited to extrapolation of past trends.

In contrast usage monitoring provides no direct data on damage state. It provides data on usage - a general term for the drivers which create and progress structural degradation processes such as fatigue, wear, corrosion and creep. These processes determine the structural life of the component. The life limit of the component is reached when structural strength is reduced below design limit load, or the component or mechanism cannot operate. In components designed on the safe life approach, a different definition of service life is used - the life at which probability of the development of cracks becomes unacceptable.

Usage monitoring allows prediction of future component life and calculation of damage levels less than the threshold of detectability by a health monitoring system. There are more benefits of usage monitoring on safe life approaches, than on damage tolerant approaches, as usage monitoring can estimate life consumed even though there is no detectable damage. Damage tolerant designs will always require detectability of damage,

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

and hence usage monitoring is then a more accurate way of predicting reduction in residual strength.

In theory, safe life design lives include all forms of degradation, corrosion, wear and random mechanical damage. In practice, the only degradation process which is quantitatively assessed for a safe life is fatigue. Aircraft operating conditions can be related to the size and number of fatigue cycles, and quantitative models exist which can relate the load cycles to life. Other forms of degradation are still operating and may curtail component life even if a Fatigue Usage Monitoring System or FUMS indicates that the permissible limit of fatigue damage has not yet been reached.

This paper will investigate the techniques used for calculation of fatigue damage in the Flight Load Synthesis and Flight Condition Recognition approaches to FUMS.

1.1 Procedures for calculation of fatigue safe lives in helicopter components

Helicopter manufacturers agree on the general procedure for calculation of fatigue safe lives, but differ in the detail. A notable investigation of variability of safe lives calculated by in house procedures of different manufacturers (2) concluded that safe lives for a pitch link with identical material properties and identical service loading histories varied from 58 flight hours to over 27,000. A further investigation of the sources of this variability (3) demonstrated that 70% of the variation could be attributed to differences in the ways different manufacturers treated the service load data - precisely the area of procedure involved in FUMS.

Standard procedures for prediction of life time to crack initiation (4,5) are shown schematically in Figure 1. Material fatigue properties gathered using smooth specimen fatigue data tested under strain control, are combined with service load cycles data. A local strain analysis of the notches where fatigue cracks will initiate is performed. A Miners summation is used to calculate the damage using the calculated values of notch strains. The service load cycles spectrum is derived using rainflow cycle counting of the

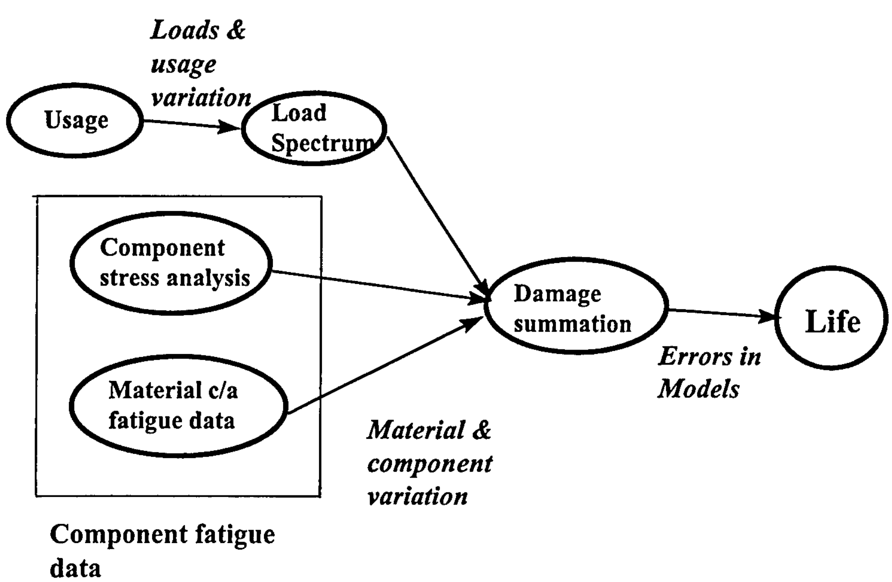


Figure 1 Schematic diagram of fatigue crack prediction process

entire service load history. Both range and mean values of individual cycles are derived by the count procedure. Accuracy of the lives calculated under these conditions is estimated using validation trials as a factor of 3 on life at best. The material fatigue data can be factored to produce an acceptably low probability of failure.

Details of the procedures used in calculation of helicopter safe lives differ from this procedure in a number of ways.

(a) High frequency load cycles are assumed to have a constant mean and are separated from low frequency manoeuvre loads for damage calculations. Damage from the two routes is subsequently summed.

(b) A factor of 1.2 is often incorporated in the load history in situations where little data is available, to allow for scatter.

(c) Rainflow cycle counting has not been universally performed, particularly for older helicopters.

(d) A component fatigue curve is often used rather than smooth specimen coupon data together with a component stress analysis.

2 TECHNIQUES FOR FUMS IN HELICOPTERS

2.1 Direct loads monitoring

For fixed non- rotating components, the task is relatively easy. Selected sites may be strain gauged and the strain history throughout the flights continuously recorded. The strain history is converted into a cycles histogram using standard algorithms for rainflow cycle counting (6). The process is performed as the cycles are completed. Any uncompleted cycles are held open until the cycle which closes them is encountered, and they may then be added to the histogram. The conversion to completed cycles is required before conversion to fatigue damage can be accomplished, and has the additional advantage that histogram storage is extremely economical of data storage space.

A rainflow cycle count performed throughout the flight will automatically record both high frequency rotor cycles as well as low frequency cycles originating in manoeuvres such as the ground-air-ground cycle. Combination of the cycles histogram with the component fatigue data by application of Miners rule then yields the damage for this period of flight, which could be updated on a continuous basis. The only errors in the process are associated with errors in the damage summation (at best a factor of 3 on life) and errors in the strain measurement process (at most 5% of the strain ranges).

Direct continuous monitoring of service strains for fatigue design calculations is performed frequently. However its use for FUMS monitoring of life on a routine basis as opposed to demonstrator use is still surprisingly uncommon. Wind Energy, Nuclear Industry and North Sea Structures are areas where it has some application (7).

2.1 Flight load synthesis

For rotating components, direct loads monitoring is not a viable option, although there are developments in miniaturised recording and processing data acquisition modules, which could in principle be mounted on the rotor head. The long-term durability of the instrumentation may be a problem in the severe rotor head environment, and the long term performance has yet to be proved.

Instead, indirect techniques are proposed. These require calibration against damage calculated using direct measurement of service loads. The flight load synthesis approach calculates flight loads in the rotating components from inputs of flight parameters such as airspeed, control lever positions, rotor torque or other parameters. The calculations are complex and a number of different approaches have been suggested. These include development of statistical correlations between measured service loads and flight parameters, using multiple regression (8), pattern recognition (9), and neural networks (10, 11).

The work of Haas and co workers (8,10,11) has shown that it is possible to obtain correlation coefficients of better than 0.95 between predicted loads and measured loads in push rods and other rotor related components. Similar values have been obtained using multiple regression approaches. Cronkite et al (12), again using multiple regression, obtained correlation coefficients of 0.768 between measured and predicted loads on a yoke beam. This lower value was in spite of using a weight function to ensure that the larger damaging loads were better fitted to the regression equation than were the more numerous non damaging loads.

Azzam et al (13) in a variant of the flight load synthesis approach, developed flight parameter correlations with fatigue damage rates rather than with the loads themselves. A neural network approach was used. This is a much simpler approach than calculation of loads in that the fatigue damage rate is a single parameter which rises and falls depending on the flight parameter conditions. It has been argued that it is less susceptible to error than calculation of damage using predicted values of loads followed by a Miners summation calculation.

Because of scatter in service load prediction, Cronkite et al (12) reduced their correlation line between predicted loads and observed ones by 3 standard deviations so as to produce a conservative prediction. In many instances this resulted in damage prediction rates which were often faster than the linear accumulation of damage predicted by the accumulation of flight hours, and considerably faster than the actual fatigue usage.

2.3 Flight condition recognition

This is similar in concept to the procedure for development of design safe lives. Damage is calculated for a number of manoeuvres or flight conditions. In flight manoeuvres are identified via monitoring of flight parameters and controls, and the appropriate damage level is assigned once the manoeuvre has been completed. As the damage associated with each manoeuvre is a constant value, this form of usage monitoring is really a monitoring of the incidence of particular manoeuvres or flight conditions. Moon et al (14,15) have reported variation in the incidence of specific manoeuvres in the operation of US Navy helicopters. A sophisticated system with over 360 different flight conditions was used. Factors of up to 8 difference between the designated percent time occupied by flight conditions and the actual time occupied, were reported.

The work in the present paper explores the variability of loads and fatigue damage for particular manoeuvres, and uses this to estimate the likely benefits of a FUMS. The advantages and disadvantages of flight load synthesis and flight condition recognition are explored. The sources of fatigue damage during helicopter manoeuvres are explored and used to assess the relation of true fatigue damage to the various types of damage estimation performed in usage monitoring.

3. ANALYSIS TECHNIQUES & DATA

The loads data used consisted of 10 hours of data recorded on a Westland Lynx Mark IX helicopter. The data consisted of 55 channels of flight parameters of which 7 parameters were strain, load or bending moment measurements made on components. Complete details of the channels and their data recording frequencies have been documented previously (16,17).

The component studied most intensively was the dogbone rotor linkage, with supporting work on the spider rotor component. In this paper the dogbone results will be presented. The dogbone is subjected to cyclic bending at the 5.5 Hz rotor frequency, with a superimposed component of the blade passing frequency of 22 Hz. The fatigue damage calculation techniques used were broadly in accordance with those used by the helicopter industry. Constant amplitude component fatigue data were supplied by GKN WHL, and were used with factors on both strength and cycles to reduce the 50% probability of failure curve to approximately minus 3 standard deviations, or a failure probability of 1 in 1,000.

A factor of 1.2 was applied to the bending moment- time histories. To investigate the difference to fatigue damage content which application of this scatter factor causes, manoeuvres were evaluated both with and without it.

Fatigue damage was evaluated using a Miners summation with failure being defined at a sum equal to 1. A Goodman mean stress correction was applied. The damage calculated for individual manoeuvres was invariably very small. Hence the calculated damage was multiplied by 10^6. This quantity was called microdamage. Microdamage may be defined as $10^6/N_r$ where N_r is the number of repeats of the load sequence to failure. The damage content and damage rate of identified manoeuvres was calculated by identifying the start and finish of a manoeuvre, cycle counting and summing the damage. This was performed for all manoeuvres in the Lynx design spectrum. The damage content of repeated manoeuvres was calculated to provide an indication of the scatter in the data. Damage rates were derived by dividing total damage for the manoeuvre by the elapsed time.

The proportion of damage which occurs between the manoeuvres was identified by calculating the damage for an entire load history and subtracting from this figure the damage calculated for the individual manoeuvres.

4 RESULTS

Figure 2 shows a typical Bending Moment (BM) history on the dogbone throughout a cruise turn manoeuvre. Also shown are the rainflow cycle count spectrum from the history, and the corresponding damage spectrum where the damage produced by each cycles bin is shown. The BM history shows the increase in high frequency cycle range and also the increase in mean level within the manoeuvre. Both high frequency and low frequency cycles are recorded on the rainflow spectrum along with their damage contributions.

The total damage associated with a manoeuvre results from a group of cycles of widely differing range and mean. Representations of this group by use of simpler parameters such as the maximum range, or a single value of the mean would be highly inaccurate. Even representations of the load spectrum by two parameter (range and mean) analytical distributions do not accurately reflect its form. Analysis of variability in the load spectra would be more complicated still.

For this reason it was decided to work with the fatigue damage content of the load spectra, rather than the load spectra themselves.

A selection of manoeuvres in the design

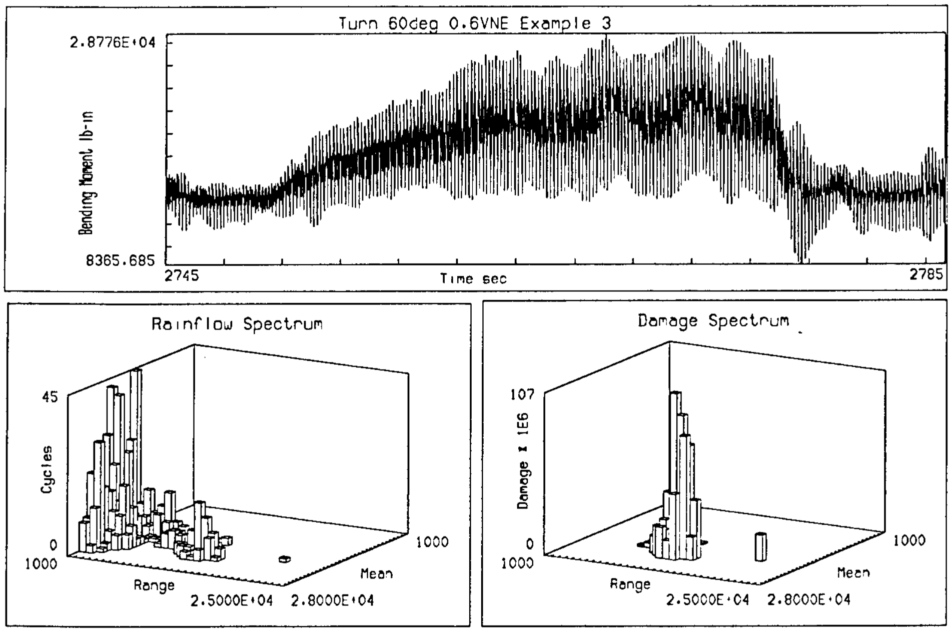


Figure 2 Bending moment-time history, rainflow spectrum and damage spectrum for cruise turn manoeuvre

Table I
Calculated fatigue damage for Lynx dogbone-selected manoeuvres

Manoeuvre	Microdamage/second worst case	Microdamage/second 50% cumulative probability
Level flight- (all speeds)	0.152	0
Cruise turns 60^0	21.05	5
Cruise turns 30^0	0.024	0.008
Control reversals	25.75	6.0
Hover	0.86	0.025
Sideways flight	4.80	0.250
Climb	0.53	0.008
Descent	0.60	0.020

spectrum for the Lynx helicopter is shown in Table I, together with worst case damage and damage at 50% cumulative probability of occurrence. For many of the manoeuvres, particularly forward flight, the most common damage level was zero. The most damaging manoeuvres for the dogbone component were high angle cruise turns, control reversals, hover control reversals, sideways and rearwards flight. Damage at 50% cumulative probability of occurrence was invariably considerably smaller than the worst case

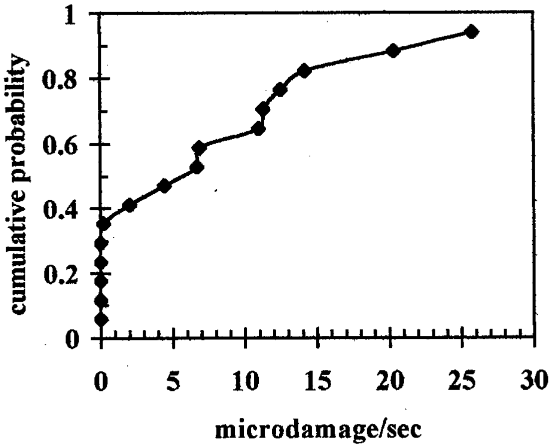


Figure 3 Control reversals - damage rate variability

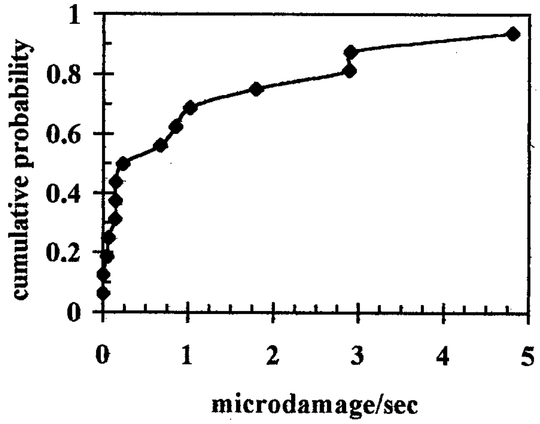


Figure 4 Sideways flight - damage rate variability

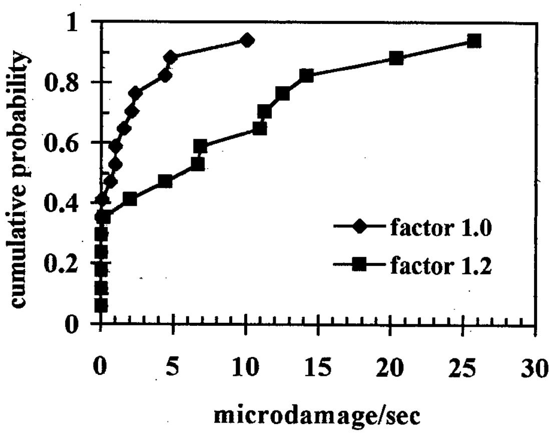


Figure 5 Control reversals - comparison of damage rate with loads factors of 1.2 and 1.0

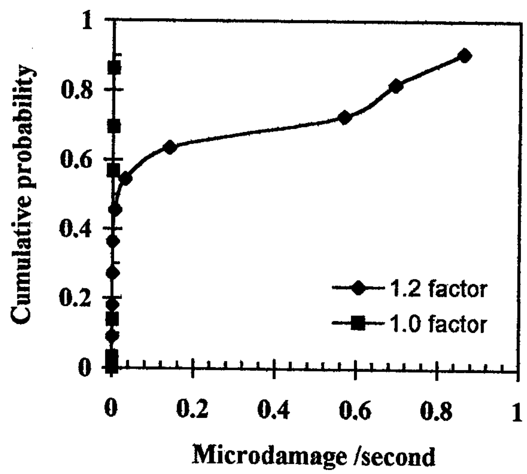


Figure 6 Hover - comparison of damage rate with loads factors of 1.2 and 1.0

damage. In many manoeuvre types, the damage and damage rate at 50% probability of occurrence was less than 10% of the worst case damage. There was no qualitative differences between the damage distributions and the damage rate distributions.

Examples of the cumulative damage rate distributions for two of the more damaging manoeuvres are shown in Figures 3 and 4 for control reversals and sideways flight. The most common values of the damage and damage rate was between zero and the value for 50% cumulative probability of occurrence. The figure for 50% cumulative probability of occurrence may therefore be taken as an approximate upper limit for most probable damage rate.

All the values shown in Table I have the 1.2 loads factor applied to them. The effect on damage rate distribution of removing the factor is shown in Figure 5 for the control reversal manoeuvres. The largest values of damage are reduced by a factor of about 4, from 80 to 20 microdamage. Smaller values at 50% cumulative probability are reduced by a factor of about 10, and values less than 10 microdamage/second become non damaging. The effect of removing the factor on the

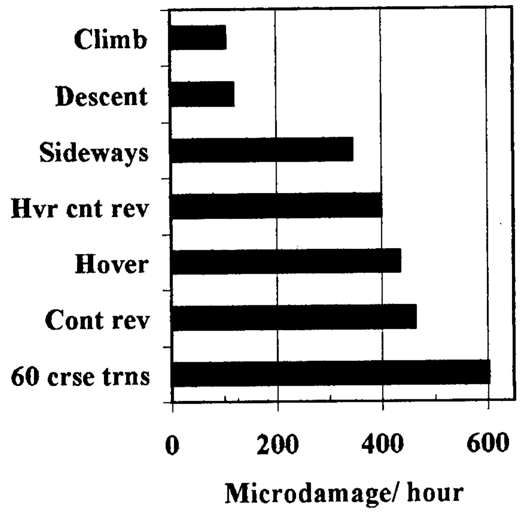


Figure 7 Microdamage/hour for worst case damage

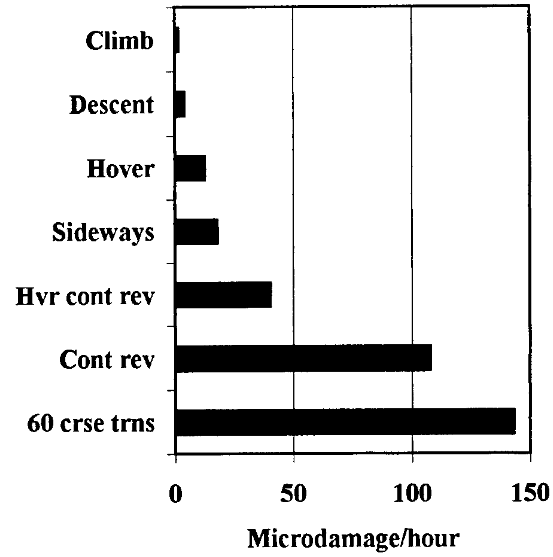


Figure 8 Microdamage/hour - 50% cumulative probability

hover damage rate distribution is shown in Figure 6. With the factor, the greatest damage rates are 0.85 microdamage/second, without the factor, all except the largest value become zero.

The non linearity of the reduction is a consequence of the non linear component fatigue curve, which is of course extremely flat in the high cycle low damage region. Small damage values will result from the spectrum creating damage in the high cycle low gradient part of the curve. Reductions in the spectrum loads caused by removal of the factor will have a much more dramatic effect in this region than in the lower cycle high gradient part of the fatigue curve.

Figure 7 shows the damage contribution per hour for the most damaging manoeuvres in the design spectrum. In this figure the damage rates per second have been modified to account for the different proportions of time occupied by the different manoeuvres in the designated flight spectrum, It was found that 5 manoeuvres dominate the total damage per hour, these being high angle cruise turns, control reversals, hover control reversals, hover and sideways flight. These total over 60% of the damage per hour. Figures 3 & 4 show that the most probable damage levels are in the region less than 50% cumulative probability (in the relatively small interval between zero and the 50% cumulative probability point). A similar plot to Figure 7 can be drawn for 50% cumulative probability and is shown in Figure 8. Once again the same manoeuvres dominate the usage for this component, with the addition of rearwards flight. In this case the most damaging 3 manoeuvres contribute over 80% of the total damage. The ratio between the worst case total damage and the total damage at 50% cumulative probability, which represents an upper limit of the most probable damage level, is 7.5.

Performing similar calculations with the loads multiplication factor of 1.2 removed, yields similar plots to Figures 7 and 8 but with the total damage reduced by a further factor of 7 for worst case damage, and a factor of 10 for the 50% cumulative probability. In the latter case, there is only three manoeuvres contributing damage; all others are at or near zero. The ratio between worst case damage and most probable damage increases slightly for the unfactored

Table II
Total microdamage/hour - all manoeuvres

Worst case - with 1.2 scatter factor	50% cumulative probability of occurrence - with scatter factor of 1.2	Worst case - 1.0 scatter factor	50% cumulative probability of occurrence - 1.0 scatter factor
2768	363	495	36

damage summation, and is about 8-9. A summary of these total damage calculations is shown in Table II.

The damage contained within strictly defined manoeuvres is one of the major sources of damage. There are two other major sources of damage during helicopter flight. Firstly, there is the damage which occurs during transitions from manoeuvre to manoeuvre. The shifts, particularly in mean level associated with manoeuvre transitions will cause large cycles recognised by the rainflow count and which will invariably be damaging. Additional damage may also occur within manoeuvres, when the stipulated restrictions on speed and height are violated. For instance, Hill et al, (16) found damage rates of up to 4 microdamage/sec for forward flight in situations where speed and altitude were not being strictly controlled. This is a factor of 40 more damaging than the forward flight worst case recorded in the current work.

Further variability in total damage levels will arise from changes from helicopter to helicopter in the proportion of time occupied by particular manoeuvres, and the number of times a manoeuvre is executed. Published US Navy data (13,14) on the AH-1 W helicopter in a sample of 35 aircraft show large shifts from the designated design spectrum figures. Cruise turn usage for example varied from 2 to 16%, with the designated usage being 11.3%. No equivalent data are available for UK RAF usage.

To investigate whether there is damage occurring during transitions between manoeuvres which would not be detected by calculating manoeuvre damage and summing the parts over the whole flight, the following experiment was conducted. Individual flights were split into separate manoeuvres without leaving gaps between them. The damage content of each manoeuvre was calculated according to the procedures described earlier, and the total damage summed over all manoeuvres. The total damage calculated this way was compared with that obtained by rainflow cycle counting the whole flight, prior to damage calculation. The two damage counts were almost identical with an error of less than 0.1%.

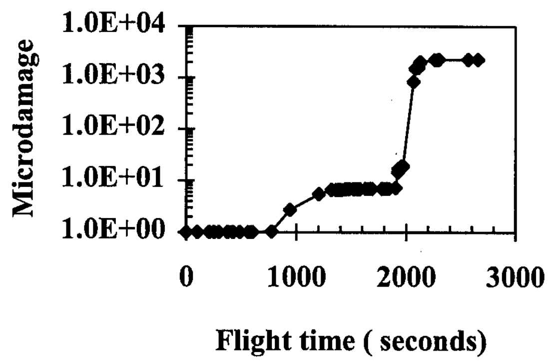


Figure 9 Cumulative damage throughout selected flight

Figure 9 shows the cumulative damage plot for a selected flight. Each point represents the end of a manoeuvre. The damage increment caused by the manoeuvre is the difference between the manoeuvre point and the previous one. The total damage in the flight was 2240 microdamage, the total from summing all the manoeuvres was 2250.

This result suggests that the rainflow procedure identifies **all** damaging cycles, both high frequency rotor based and low frequency manoeuvre based. The damage data which would not be included in a manoeuvre based analysis with rainflow would be damage due to unclassifiable manoeuvres, and manoeuvres which have not been performed within the permitted limits on parameter variation to define the flight condition.

This result confirms previous published work, (17) in which the service load histories were split into sections of lengths from 1 second to 30 seconds and the damage for individual lengths was calculated. The damage sum accruing from all the intervals was compared with the value obtained by rainflow cycle counting the entire history. Providing that the largest incomplete load cycle in each interval was closed by the programme, the results of the two techniques were indistinguishable unless the lengths of service history were less than 1-2 seconds.

5 RELATIONSHIP OF USAGE CALCULATIONS TO DESIGN SAFE LIFE

These damage calculations cannot be related directly to design safe lives. The damage values calculated will be conservative to the extent that a reduced component fatigue curve has been used together with a factor of 1.2 applied to the service bending moments. The cycles counting procedure used identifies all cycles in a section of flight, whether high or low frequency; there is no need for separate counting of rotor induced cycles and low frequency cycles of the ground-air-ground type. There is damage associated with events which are not classified as manoeuvres within the design spectrum. There is a further possible non conservative error associated with the accuracy with which the constant amplitude fatigue data, together with the Goodman mean stress correction and Miners summation procedure predicts failure. This is an unknown quantity. No information is available on the outcome of validation trials which might establish this point.

The safe life calculation procedure, as noted earlier, ensures conservatism but the accuracy and relation to "true" values of damage is unknown. It may be possible to build a usage monitoring damage calculation system which is based on individual manufacturers in house calculation procedures. This would have the disadvantage that the more conservative the system the less maintenance credits would be available. It is suggested that usage monitoring damage calculation techniques have a fixed, conservative relationship to validated fatigue damage calculation results, rather than to a changing procedure which varies with flight state and with Design Authority.

5.1 Implications for flight load synthesis and flight condition recognition techniques

It is clear from the present results on variability of damage for particular manoeuvres together with the manoeuvre usage data presented in (14,15) that there is considerably more variability in damage for individual manoeuvres than there is in the variability in usage of the manoeuvres themselves. A usage monitoring system based on flight condition recognition must always assume worst case damage for the flight conditions, and hence the damage assumed will be much more conservative than has actually occurred.

Some of the variability in damage for manoeuvres found in the present work is probably associated with differences in flight regime which a more sensitive discriminator of flight condition would assign to different conditions. Nevertheless, the variability in damage associated with each manoeuvre remains high. A system based on flight condition recognition would also be required to assign maximum damage to periods of flight which were in an unrecognised condition. However, there is no evidence that there is unrecorded damage occurring in the transitions between manoeuvres, provided that the damage for the individual manoeuvres has been calculated using rainflow cycles counted load data.

An accurate flight load synthesis approach will most comprehensively record the damage accruing on the various helicopter components, as it will fully recognise the level of damage occurring independently of the flight condition. However, the published evidence is that the scatter on calculated flight loads compared with the actual ones is such that conservative predictions indicate greater damage rate than flight hours based damage monitoring. More development is clearly needed before the FLS technique can be applied.

6 CONCLUSIONS

(1) Variability for non zero damage levels for nominally identical manoeuvres is between 100 and 1,000. This factor is considerably greater than published data on the variability of manoeuvre occurrence which is a factor of little more than 3-4.
(2) The differences in damage level produced by application of the loads scatter factor of 1.2 is a factor of 2-3 on worst case damage, and a factor of about 10 for loads which are marginally damaging .
(3) A single rainflow cycle count fully identifies all fatigue cycles in a selection of manoeuvres, including both high frequency and low frequency manoeuvre loads.
(4) A fatigue damage calculation methodology for application in a FUMS system should have a known, fixed and conservative relationship to a calculation of a "true" fatigue damage.

ACKNOWLEDGEMENTS

The continued financial support of the UK Civil Aviation Authority and the Ministry of Defence is gratefully acknowledged. The helicopter load data were provided by MOD. Thanks are also due to many colleagues at Cranfield, MOD, RAF and CAA for numerous helpful discussions in the pursuit of viable FUMS.

REFERENCES

(1) Thiagarajan, C., Irving, P.E., "In Service Damage Monitoring Techniques for composites", *in* "Proceedings Aerotech 1994", IMechE, 1994.
(2) Borgman, D., "Helicopter Fatigue Methodology", AGARD CP 297, 1981.
(3) Gunsullus, C.T., Nagy, E., Stennett, P.G., Connelly, W.G.F., "Investigation of Variation in Fatigue Life Calculated Using Damage Fraction", J. Am. Heli. Soc., *35*, 1990, p 35.
(4) Dowling, N.E., "A Discussion of Methods for Estimating Fatigue Life" *in* Proceedings of 1982 Fatigue Conference, paper No 820691, SAE 1982.
(5) Dowling, N.E., "A Review of Fatigue Life Prediction Methods", *in* Proceedings of 1987 Conference, paper No 871966, SAE 1987.
(6) Downing, S.D., Socie, D.F., "Simple Rainflow Counting Algorithms", Int. J. Fatigue, January 1982, pp 31-40.
(7) Karadeniz, H., "Fatigue Analysis of Offshore Structures under Narrow Banded Stress Processes", *in* Proceedings of the first European Offshore Mechanics Symposium, Trondheim, Norway; Int. Soc. Offshore & Polar Engineers, 1990, pp 221-228.
(8) Haas, D.J., Imber, R., "Identification of Helicopter Component Loads using Multiple Regression", *in* AIAA Dynamics Specialist conference, AIAA 1992, Paper

92-2110, pp 302-314.

(9) Tang, S.S., Moffat, J., "Development of an On-Board Fatigue Processor for Helicopters", Int. J. Fat., *18*, 1996, pp 179-189.

(10) Haas, D.J., Milano, J., Flitter, L., "Prediction of Helicopter Component Loads Using Neural Networks", *in* Proceedings 34th AIAA Structures, Structural Dynamics & Materials Conference, AIAA, 1993, paper 93-1301, pp 12-25.

(11) Haas, D.J., Flitter, L., Milano, J., "Helicopter Flight Feature Extraction for Component Load Monitoring", *in* Proceedings 35th AIAA Structures, Structural Dynamics & Materials Conference, AIAA, 1994, Paper 94-1308, pp 1-16.

(12) Dickson, W., Cronkhite, J.D., Summers, H., "Usage and Structural Life Monitoring with HUMS", *in* 52nd Meeting of the American Helicopter Society, AHS, 1996.

(13) Azzam H., "The Use of Helicopter Flight Parameters for Damage Prediction", Unpublished Report, 1993.

(14) Barndt, G.L., Moon, S., "Development of a Fatigue Tracking Program for Navy Rotary Wing Aircraft", *in* 50th Meeting of American Helicopter Society, American Helicopter Society, 1994, pp 1359-1389.

(15) Moon, S., Menon, D., Barndt, G., "Fatigue life Reliability Based on Measured Usage, Flight Loads and Fatigue Strength Variations", *in* 52nd Meeting American Helicopter Society, American Helicopter Society 1996, pp 366-389.

(16) Hill, K., Hudson, R.A., Irving, P.E., "Loading Spectra, Usage monitoring, and Prediction of Fatigue Damage in Helicopters", *in* "Estimation, Enhancement and Control of Aircraft Fatigue Performance", Proceedings ICAF 95, Eds., Grandage, J.M., Jost, G.S., EMAS, 1995, pp 531-550.

(17) Irving, P.E., Hudson, R.A. "Damage Estimation Techniques for Usage Monitoring in Helicopters", *in* "Life Assessment, Life Extension of Engineering Plant, Structures and Components", Eds. Edwards, J.H., Flewitt, P.E.J., Gasper, B.C., McLarty, K.A., Stanley, P., Tomkins, B., EMAS, 1996, pp 253-264.

SH-60 Helicopter Integrated Diagnostic System (HIDS) Program Experience and Results of Seeded Fault Testing

Andrew J. Hess
NAWC AD
Propulsion and Power Dept.
Building 106, Unit 4, 2215 Elmer Rd
Patuxent River, MD 20670-1534, USA

Bill Hardman
NAWC AD
Propulsion and Power Dept.
Building 106, Unit 4, 2215 Elmer Rd
Patuxent River, MD 20670-1534, USA

Chris Neubert
NAWC AD
Propulsion and Power Dept.
Building 106, Unit 4, 2215 Elmer Rd
Patuxent River, MD 20670-1534, USA

The evolution of automated diagnostic systems for helicopter mechanical systems has been aided by a Navy program of systematic testing of drive train components having known anomalies (seeded faults) while simultaneously executing a suite of diagnostic techniques to identify and classify the mechanical anomalies. This program, called the Helicopter Integrated Diagnostic System (HIDS) has been carried out using an iron bird test stand (SH-60) at NAWC - Trenton, and SH-60B/F flight vehicles at NAWC - Patuxent River. The SH-60 HIDS program has been the Navy's cornerstone effort to develop, demonstrate, and justify integrated mechanical diagnostic system capabilities for its helicopter fleets. The objectives of the program were to:

1. Acquire raw data for multiple cases of "good" and seeded fault mechanical components on a fully instrumented drive train to support the evaluation of diagnostic algorithms and fault isolation matrices. Data is being acquired from 32 vibration channels simultaneously at 100 kHz per channel while a continuous usage monitoring system records parametric steady state data from the power plant and airframe.
2. Analyze vibration and other diagnostic indicators to evaluate sensitivity and performance of all available diagnostic methods when analyzing well-documented parts. Evaluate relative effectiveness of these various diagnostic methods, indicators, and their associated algorithms to identify and optimize sensor location combinations.
3. Demonstrate the ability to integrate and automate the data acquisition, diagnostic, fault evaluation and communication processes in a flightworthy system.
4. Integrate and evaluate comprehensive engine monitoring, gearbox and drivetrain vibration diagnostics, advanced oil debris monitoring, inflight rotor track and balance, parts life usage tracking, automated flight regime recognition, power assurance checks and trending, and automated maintenance forecasting in a well-coordinated on-board and ground-based system.
5. Provide an extensive library of high quality vibration data on baseline and seeded fault components. This data can be made available to anyone wanting to prove their diagnostic techniques or develop new capability.
6. Provide a "showcase", state-of-the-art, fully functional Integrated Mechanical Diagnostic system to act as a catalyst demonstration which might lead to interest in a fleet wide production application.

This paper will describe the overall program, the goals and objectives, the facilities used, the system evaluated, the accomplishments and the results and conclusions obtained to date. The results of extensive gearbox and powertrain "seeded fault" testing will be presented. Lessons learned which can be applied to future Helicopter Usage Monitoring Systems (HUMS) and/or Integrated Mechanical Diagnostic (IMD) systems will also be discussed.

Introduction

Background

The U. S. Navy and U. S. Marine Corps have long had a requirement to improve several aspects of their rotary wing operations in order to improve readiness through more effective maintenance, eliminate losses of aircraft and personnel, and dramatically reduce maintenance related costs. The requirements to extend the service life of aircraft and the limitations on manpower have increased the urgency of affecting these types of improvements. A majority of the Class A mishaps (loss of aircraft and/or personnel) in Navy helicopters are caused by engine and drive train failures (Ref. 1). The need to accurately identify and diagnose developing faults in mechanical systems is

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

central to the ability to reduce mechanically induced failures and excessive maintenance. The Navy has successfully developed and deployed fixed wing engine monitoring systems, notably on the A-7E and subsequent fighter/attack aircraft. These fixed wing Engine Monitoring Systems (EMS) have impacted flight safety, aircraft availability, and maintenance effectiveness. The Navy also successfully demonstrated a promising automatic mechanical fault diagnostic capability on its gearbox overhaul test stands in Pensacola, Florida.

The U.S. Navy would clearly benefit from a reliable state-of-the-art diagnostic capability on-board rotary wing aircraft. Based upon the Mission Need Statement (Ref. 1), such a system is expected to enhance operational safety and significantly reduce life cycle cost through it's ability to predict impending failure of both structural and dynamic drive system components and consequently direct on-condition maintenance actions and/or alert the pilot to conditions affecting flight safety.

There is currently considerable activity underway to develop integrated health and usage monitoring systems particularly for helicopter subsystems (transmissions, rotor head, engines, tail drive systems, etc.). A major challenge is acquiring and managing large quantities of data to assess the health and usage of the aircraft system.

A significant disadvantage of first generation commercial systems in 1992 was the lack of raw data acquired to validate and optimize the full, Integrated Mechanical Diagnostic (IMD) functionality. Such is a necessary component of any development effort in order to lend confidence to the users, that the system will reliably indicate mechanical and rotor system faults, avoid false alarms, and develop structural and mechanical system usage routines. These are the keys to preparing an IMD system for deployment.

Present Work

The Naval Air Warfare Center Aircraft Division is currently leading a comprehensive program as authorized (Ref. 2) to evaluate diagnostic technologies. The SH-60 was selected as the test vehicle because it offered the best availability of test assets and the highest potential for support because of the large fleet of aircraft among the Navy, Army and Coast Guard. The program designated Helicopter Integrated Diagnostic System (HIDS) uses state-of-the-art data acquisition, raw data storage, and algorithmic analysis provided under contract by Technology Integration Inc. [TII - now part of BFGoodrich Aerospace (BFG)] to evaluate the propulsion and power, rotor, and structural systems. Cockpit instruments and control positions are recorded during the entire flight for usage monitoring and flight analysis. Rotor track and balance is performed via the trackerless ROTABS system. Analyzing vibration signals acquired from a comprehensive suite of accelerometers assesses dynamic component health.

The program reported herein is structured to evaluate two functionally equivalent TII/BFG systems at the following test sites:

1. Flight Testing at NAVAIRWARCENACDIVPAX (Naval Air Warfare Center, Patuxent River, Maryland): Demonstrate the integration of a comprehensive integrated diagnostic system which performs rotor track and balance, mechanical and rotor system diagnostics, and dynamic and structural component usage monitoring. Evaluate the operability of the demonstration system and provide a foundation for the user interface requirements functional specification for fleet procurement. In addition, evaluate a real time engine performance estimation algorithm provided by General Electric Aircraft Engines in cooperation with Dr. Peter Frith of the Australian Mechanical Research Laboratory (AMRL) via implementation onboard the HIDS flight test aircraft.

2. Ground Testing at NAVAIRWARCENACDIV-TRENTON (Naval Air Warfare Center, Trenton, NJ): Conduct fault detection validation testing in a unique universal full scale Helicopter Transmission Test Facility (HTTF) which currently consists of the entire SH-60 power drive system (engines, transmission and tail drive system). Evaluate and validate the TII/BFG system and associated algorithms to detect seeded faults while building a base of raw data for evaluating other fault detection methods. In addition, the program is evaluating other advanced technologies in parallel with the TII system. The information generated from this testing will form a body of knowledge from which specifications can be written to procure effective production versions of the integrated diagnostic system.

The purpose of this paper is to describe the overall program, the diagnostic system, the NAVAIR-WARCENACDIVTRENTON test cell, the seeded fault testing, flight testing and major accomplishments to date.

Description

This section will describe the systems and facilities that are being used in support of HIDS. The test articles are the diagnostic technologies. The SH-60 test facilities are being utilized to exercise these diagnostic technologies.

HIDS Diagnostic System

In 1993, the NAVAIRWARCENACDIV awarded a competitive contract on the Broad Agency Announcement to TII for two functionally equivalent integrated diagnostic systems. (TII elected to make a substantial investment in the program through providing Commercial Off the Shelf (COTS) hardware and software.) One system was configured for rack mounting in the Trenton, NJ test cell and the other is flyable ruggedized commercial grade hardware. The TII system uses an industry-standard open architecture to facilitate modularity and insertion of new hardware and software. TII has divided the system into two main avionics units, the commercial off-the-shelf KT-1 aircraft parameter-usage monitor and the KT-3 vibration acquisition, analysis and rotor track and balance system. System architecture and data flow is shown in Figure 1. Though not a production type unit, the KT-3 is essential to acquire the raw data necessary to substantiate the diagnostics technology and obtain enough knowledge to write the minimum acceptable production specification.

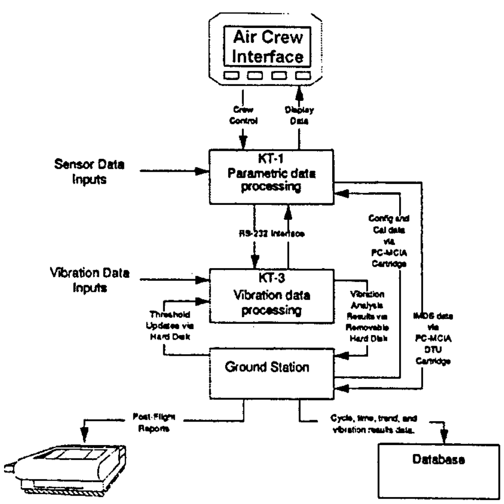


Fig. 1. Diagnostic System Architecture.

Structural Usage Monitor (KT-1): The TII/BFG system performs aircraft usage monitoring, engine condition monitoring, drive shaft condition monitoring, gearbox condition monitoring, chip detector monitoring and rotor track and balance. The first generation system (Aircraft 326) acquired aircraft and engine parameters during flight at a rate of three hertz, and the second generation system (Aircraft 804) at 10 hz. Averaged data is stored to a PCMCIA card at one hertz. If parameters (temperatures, pressures, speeds, pilot stick reversals, load rates, etc.) go into exceedance, all data acquired at the high data rate with a 15 second preview and 15 second post-view of the event is stored. This data provides the usage spectrum of the aircraft, engine performance information, and the flight regimes for trending gearbox vibration information and an actual record of the mission, and is available for post-processing for re-calculation of regime recognition and structural usage routines. The past 150 exceedance events are stored in non-volatile ram in the case of data card damage or loss. The list of parameters recorded includes those sanctioned by Navy structures competency for use to execute structural usage monitoring.

Engine Performance: The HIDS Cockpit Display Unit (CDU) depicted in Figure 2 interfaces with the pilot to execute and display results of automated NATOPS T700 engine health checks and Engine Power Performance Index (PPI) which are accomplished by the KT-1. A fourth order quadratic, the PPI is a best fit curve representing an engine degraded 7.5% from the specification line. The PPI output is a value in degrees C calculated from the 7.5% degraded line. This provides significant improvement by automating the acquisition and collecting hundreds of points per flight versus one or two. It can provide a warning to the pilot when an engine has degraded due to salt ingestion, sand erosion or other foreign object damage (FOD).

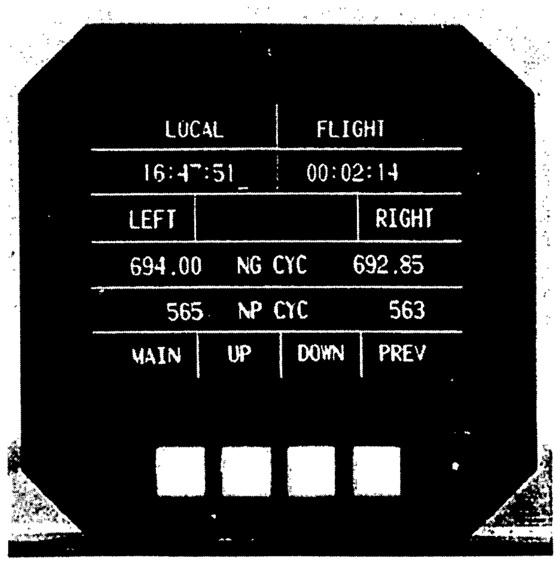


Fig. 2. Central Display Unit.

Vibration Based Mechanical Diagnostics: The focal point of this program was to explore a wide variety of diagnostic methods based upon vibration inputs, in a manner that would lead to a rational selection of reliable "production" techniques with a high confidence in accurate detections with low false alarm rates. Vibration data recorded at both Trenton, NJ and Patuxent River, MD uses the same acquisition system, sensors, mounting and accelerometer locations. The data sets are digital time series records, recorded simultaneously for up to 32 channels (accelerometers and tachometers), at 100,000 samples per second, 0-50Khz bandwidth, for 30 seconds. This proof-of-concept system records five sets of raw data per flight for post flight data analysis in the ground station. Drive system accelerometer locations are shown in Figure 3 for the input and main modules and Figure 4 for the tail section. The mechanical diagnostic system algorithms provided by TII/BFG under investigation are "classical," model based diagnostics. That is, the model is composed of the Sikorsky proprietary gear and bearing tables for the SH-60B drive system. No fault or anomaly detection training is required. The system provides three significant contributions to the development and verification of diagnostics for helicopters:

1. First, the system acquires data from all channels simultaneously. This makes it possible to use multiple channels to analyze a single component; an essential element of false alarm reduction. Today, the HIDS system is the only flying data acquisition system that has demonstrated the ability to record the raw and processed data set for an entire aircraft propulsion and power drive system. The HIDS system saves raw time series data, for all channels including tachometers for post flight evaluation and future algorithm development. This minimizes the possibility that a malfunction in the preprocessing could contaminate the data base.

2. Second, the system has the capability to automatically adjust to provide good signal to noise ratios for all channels. The system starts each acquisition with a one second acquisition, and internally sets the gains based upon the measured signal amplitude to maximize dynamic range. The gain for each channel is recorded with the raw data for future analysis.

3. Third, is the capability for on-board processing. All gears, bearings and shafts are analyzed and the diagnostic result will be written to the KT-1 trend data according to flight regime. The raw data files can be held in RAM until the analysis is complete, then discarded if no anomalies are identified by the limit check. If a parameter is deemed to be in "maintenance" or "alarm" status by the KT-1 limit checks, the component of concern would have all of the accelerometers that are used for its analysis plus the aircraft tachometer saved as raw digital time series data for post flight investigation. When data is taken by a pilot-activated switch, raw data is written to disk with all of the analysis results. The HIDS program is in the process of determining alarm limits and algorithm sensitivities to achieve this goal and level of integration.

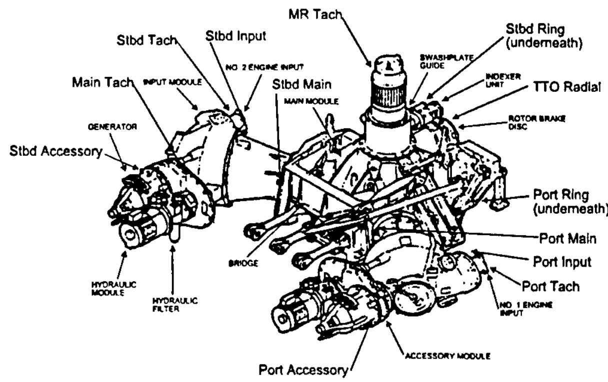


Fig. 3. Accelerometer Locations on Input and Main Modules.

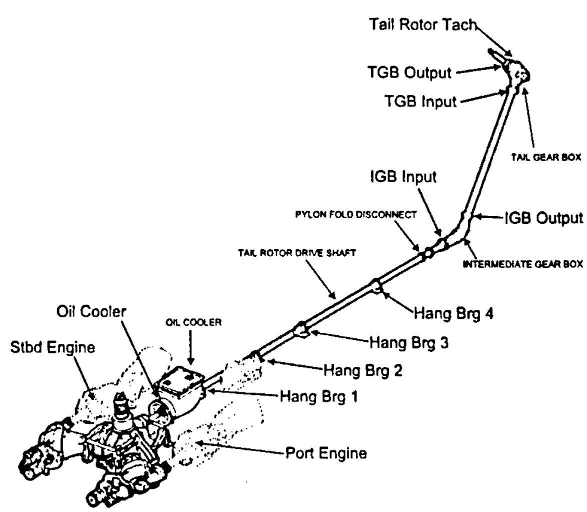


Fig. 4. Accelerometer Locations on the Tail Drive System.

Rotor Track and Balance: The ROTABS system promises to negate the need for on-board trackers and utilize higher order mathematics and a significantly larger data set to resolve the adjustments required to keep the rotor system in track and balance. ROTABS does not collect or use track data to compute rotor adjustments. ROTABS adjustments are computed from vibration data collected by the HIDS (Helicopter Integrated Diagnostic System) installed on the aircraft, using six accelerometers at three locations within the fuselage. These transducers are arranged as follows: A single-axis device sensing vertical vibration, and a two-axis device sensing vertical and lateral acceleration, both attached to the bulkhead immediately behind the pilot and copilot. One is located near the copilot's left shoulder, while the other is near the pilot's right shoulder. A three-axis device sensing vertical, lateral, and fore/aft vibration is located on the cabin ceiling just aft of the vibration absorber, roughly on the centerline of the fuselage. The system simultaneously acquires six accelerometer channels and then processes them simultaneously and resolves the corrections using transfer function from a training data set including pitch rod sensitivity, hub weight sensitivity, and tab bend sensitivity. On other aircraft types the system has demonstrated the ability to maintain track limits while simultaneously optimizing vibration in 6-degrees-of-freedom at 1/rev and selected harmonies thereof. Main and tail rotor track and balance accelerometers on the aircraft are recorded as part of the vibration data set. They will be processed at the pilots command and automatically in predetermined flight regimes for trending. Adjustments will be recommended by the groundstation upon completion of a flight test. Raw vibration data is stored for algorithm training and validation.

Groundstation: The HIDS groundstation houses maintenance, pilot, and engineering windows to support complete health and usage functionality. Tools are provided for parts and maintenance tracking, rotor track and balance, mechanical diagnostics, flight parametric data and flight regime replay, pilot flight logs, and projected component retirement times. During a flight data download, the groundstation calculates flight regimes from downloaded parametric data, and updates life usage on pre-selected serialized components in a data base upon aircraft data download. Functions to trigger usage based maintenance and component replacement are designed into the system. Historical data replay provides regime, event and exceedance information along with all aircraft parameters for the entire flight. Pilot control inputs are displayed along with all aircraft parameters for the entire flight. Pilot inputs are recorded along with other parameters which is essential for understanding events during a flight. The ground station has been shown to reduce the paperwork associated with daily operations and to direct maintenance personnel to the faulty component identified by diagnostics.

Description of the Test Cell

The NAVAIRWARCENACDIVTRENTON Helicopter Drive System Test Facility has been described in detail (Ref. 3). The test cell uses aircraft engines to provide power to all of the aircraft drive systems except the rotors. Power is absorbed through both the main rotor mast and tail rotor shaft by water brake dynamometers. The main rotor shaft is loaded in bending, tension and torque to simulate flight conditions. There is a speed increasing gearbox between the main rotor mast and the water brake which raises the main rotor speed by a factor of 32. This allows water brakes to extract up to 8000 shaft horsepower (SHP). The complete aircraft lubrication system is used with the oil cooler, oil cooler blower and blower drive shaft part of the system assembly. The tail drive system is installed and power is extracted from the tail at operating speed. The tail water brake can extract up to 700 SHP.

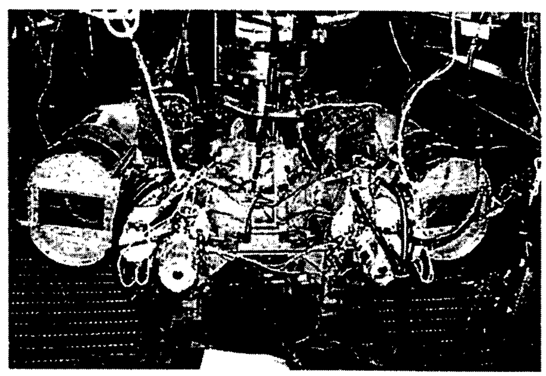


Fig. 5. Main Transmission Assembly including Accessories.

The tail drive system installation allows balance and alignment surveys on the blower, tail drive shafts and disconnect coupling. Aircraft viscous damper bearing assemblies support the installation. The length of the test cell limits the number of tail drive shafts, so two of the aircraft shafts are not installed. The test cell also supports the aircraft accessories. Generators and hydraulic pumps are mounted on the accessory gearboxes and loaded to simulate aircraft operation (see Figure 5). This is a significant capability, especially when diagnostics using vibration acquisition is the test objective. Vibration signatures collected from NAVAIRWARCENACDIV-TRENTON test cell include frequency content from all dynamic components of the loaded power drive system. The complex signal is representative of the aircraft environment.

Since this cell has the ability to operate all the aircraft mechanical systems together, the diagnostic system can record all the component "signatures" to a data base. This data base can then be interrogated to determine system health, and system performance rather than a diagnostic evaluation of a single component or fault. This is a significant improvement over single component regenerative rigs that tend to have two gearboxes that generate the same frequencies (and cross-talk) bolted to a single stand and none of the adjacent mechanical systems.

Aircraft Installation

The HIDS installation is the first health and usage monitor with advanced gearbox diagnostics to be placed on-board a US military helicopter. The system has a menu driven cockpit display (see Figure 2) for pilot information/interface. The KT-1 usage monitor is built on an open architecture, STD-32 bus housed in a 1/2 ATR short box, which has unused slots for future integration of selected KT-3 functions. Download from the KT-1 is accomplished via the data transfer unit (DTU) Type II PCMCIA card. The KT-3 vibration analysis system is housed in a large vibration isolated chassis with removable hard drives and a full VME chassis. This system is necessary for the development program to acquire all of the raw data that generates an airborne warning or alarm for either confirmation of the fault, or development of additional algorithms that identify a data problem that resulted in a false alarm. A significant benefit of this system is the comprehensive database, which is a powerful resource for diagnostic development. This system, although conspicuous in appearance, is being reduced to a set of cards in the 1/2 ATR KT-1 after successful demonstration of the required diagnostics, i.e., optimization of the number of simultaneous channels, gain control and processing.

Evaluation

The SH-60 was selected for this program since it offered the best availability of test assets and highest potential for support due to the large fleet of aircraft between the Navy, Army and Coast Guard. The NAVAIRWARCENACDIVTRENTON drive system includes two General Electric T700 engines, the main transmission, oil cooler and the tail drive system .

Test Objectives

To insure a comprehensive test effort, the planning for this test program included support from individuals and organizations involved with the design of the H-60 aircraft and diagnostic systems. The team developed and documented the program plan (Ref. 3). All seeded fault test planning is discussed with Sikorsky drive system engineering prior to execution. Team discussions led to the objectives and test sequence summarized below.

1. Demonstrate operation of an integrated diagnostic system for tracking usage of the helicopter power drive train.

2. Evaluate the ability of the diagnostic system to identify localized faults in an entire drive system.

3. Quantify the level of signal for a known defect size to develop operational limits and trending for the SH-60 drive system.

4. Evaluate the diagnostic algorithms for cracked gear fault identification and sensitivity.

5. Evaluate the diagnostic systems ability to identify a degraded performance engine and damaged engines removed for cause.

6. Evaluate the diagnostic systems sensitivity to defects and faults in tail drive shafts and bearings.

7. Evaluate the diagnostic systems sensitivity to bearing defects in gearboxes.

8. Evaluate the diagnostic systems ability to identify oil cooler blower faults.

9. Evaluate variability of data across flight regimes (including torque and weight variations).

10. Evaluate sensor placement sensitivity for the various defects. The objective is to minimize the total number of sensors required to identify faults large enough to require maintenance action and to increase robustness via use of secondary sensors.

11. Determine ambient temperature affects on the diagnostics.

12. Support The Technical Cooperation Program (TTCP) in evaluating new and emerging technologies in diagnostics.

13. Evaluate the potential for detecting misalignment, bad pattern and improper shimming during assembly that may be the cause of premature damage in mechanical systems.

14. Develop seeded fault data library that can be used to evaluate systems in the future without repeating the test program.

15. Verify ROTABS rotor track and balance.

16. Demonstrate automated engine health monitoring by automating the Health Indication Test (HIT) check and implementing a real time engine performance algorithm.

17. Evaluate as many currently available propulsion and power drive system diagnostic technologies as possible in test cell 8W and assess their relative effectiveness.

18. Evaluate the data collected on-board the aircraft with the test cell data to validate the pertinence of test cell proven algorithms for use on-board an aircraft.

19. Categorize diagnostic results with respect to aircraft flight regime to define optimized system acquisition and processing requirements.

20. Demonstrate automatic acquisition of mechanical diagnostics an ROTABS via flight regime recognition.

21. Demonstrate real time, on-board analysis and health assessment of drive system gears, shafts, and bearings.

22. Demonstrate flight data replay and structural usage functions in groundstation.

23. Demonstrate ability of the diagnostics to reduce component "false removals" and trial and error maintenance practices.

24. Demonstrate methods that improve the accuracy of component condition assessment and reduce false alarms.

Test Plans

Testing of the diagnostic system has been divided between two Navy activities that can exercise as much of the entire diagnostic system as possible. The NAVAIRWARCENACDIVTRENTON Helicopter Transmission Test Facility (HTTF) and NAVAIRWARCENACDIVPAX aircraft both operate the entire propulsion and power drive system during testing. Test plans maximize the return on investment when the system is evaluated in a single test vehicle.

1. Usage Monitoring KT-1: Usage monitoring requires continuous measurement and recording of a number of parameters that directly or intentionally relate to the fatigue life determination of critical mechanical and structural components. Evaluation of the usage data products and ground station is primarily accomplished through the demonstration and use of the system. Accuracy of the signals compared with aircraft cockpit parameters, and proper operation of exceedance and event functions have been documented. Production specification requirements for minimum acceptable functionality will be the primary product.

2. Vibration Monitoring and Diagnostics: Reliable fault identification from vibration signatures is a well documented, but difficult task. In many test cases, the researcher has been able to show that a given process can successfully identify a fault in a small scale test. Production use in complicated systems that have varied operational parameters with time has proven to be much more difficult to implement without false alarms and missed detections. In order to maximize the potential benefit of the HIDS program, early program decisions drove the diagnostic system to be a state-of-the-art data collection and processing system, with the intention of acquiring the raw data, and using it as a foundation to allow rational selection and evaluation of diagnostic parameters such as data rate, sample length, degree of redundancy required, etc., and also to identify the anomalies that result in inconsistent system performance. The KT-3 and NAVAIRWARCENACDIVTRENTON test cell have been combined to create a unique mechanical diagnostics laboratory.

NAVAIRWARCENACDIVTRENTON began acquiring seeded fault assets at the program inception. These parts had been removed from the overhaul process for discrepancies and were set aside for test rather than scrapped. This provided a tremendous cost savings by avoiding purchase of good parts for artificially seeded fault specimens, while supplying naturally created faults for test. Sikorsky Aircraft parts from prior bench qualification tests are also available for test. These parts are "bench test only" assets since they experienced over-torque conditions during test. The program has over two full sets of Not For Flight Asset (NFFA) gearboxes. The spares can be implanted with faults while another gearbox is tested. The testing initially concentrated on the tail drive system to verify the TII/BFG diagnostic system operation and performance. Subsequent testing has been performed on all drive system components, including artificially implanted and naturally occurring faults. The test conditions have consisted of sequentially varying power settings throughout the normal range of operation. It is essential to understand the sensitivity of the diagnostic algorithms as a function of changing aircraft power. Ambient temperature variation effects are included in the analysis. The first data set from each run is taken before the oil warms up at low torque to obtain a data base that can be compared to flat pitch maintenance ground turns for troubleshooting.

Test runs to evaluate component assembly (i.e. build-up variation) requires gearbox disassembly, assembly and test sequences without changing any parts. All four of the input and main gearbox assemblies in the data base were tested for sensitivity to bolting being loosened, housings jacked apart, and then reassembled with the same components.

System Installation

The HIDS system is capable of accommodating multiple configurations. NAVAIRWARCENACDIVTRENTON test cell and aircraft 162326 installations are the same for a majority of the inputs. The aircraft has many additional parameters that are not present in the test cell, such as flight parameters including altitude, airspeed, pitch, roll and heading. Also, the aircraft system measures fuel quantity while the test cell system measures fuel flow. Aircraft 162326 was made available for instrumentation in the spring of 1994 and the entire system was installed by 1 August 1994. The initial installation was completed with a majority of parameters in good operation and a system that functioned and passed installation acceptance tests. Several modifications have been incorporated since the commissioning. Performing checkout of system functionality at NAVAIRWARCENACDIVTRENTON tested the aircraft system changes, and many of the aircraft discrepancies were found to be in areas where the aircraft was different from NAVAIRWARCENACDIVTRENTON. The interface documentation was updated and validated accordingly. In March of 1997, the next generation HIDS system (with improved KT-1) was installed on PAX aircraft 804 for continued analysis and development.

Vibration Data Analysis

The HIDS program is correlating the seeded fault test data acquired in the NAVAIRWARCENACDIVTRENTON test cell to the NAVAIRWARCENACDIVPAX flight data. The diagnostic system user interface and its ability to detect faulty components in a full drive system are being evaluated using NAVAIRWARCENACDIVTRENTON data. The operational characteristics, rotor track and balance and user interface are being evaluated at NAVAIRWARCENACDIVPAX.

Data is recorded at both sites using the same acquisition system, sensors, mounting, and accelerometer locations. The data sets are digital records, recorded simultaneously on all channels at 100,000 samples per second for 30 seconds. This system is believed to exceed the requirements for a total on-board health and usage monitoring system. However, by exceeding the requirements for data acquisition under known conditions, HIDS will provide the rationale to specify the minimum system requirements needed to achieve the low false alarms and complete functionality goals. This system can store and analyze large amounts of meaningful raw data and has significant value when new aircraft types or newly overhauled aircraft require a new baseline.

The TII/BFG diagnostic system has a comprehensive scientific development environment that aids the user in evaluating and tuning diagnostic system performance. Trending of indicators and adjustment of limits is a useful part of the system, and the flexibility to add and develop new algorithms is also noteworthy. This ability makes it possible to review and modify the processing in the ground station to optimize on-board system performance.

The HIDS program, by taking advantage of these tools for diagnostic system development and verification has an excellent opportunity to properly bound the operational issues that have limited the successful implementation of currently available health and usage monitoring system. Extensive analysis and algorithm development of the baseline and fault raw data continuously improves the performance of the system through scientific understanding of the mechanics of the helicopter, and through detailed study of the events that have resulted in false alarms. By utilizing the database, HIDS has been able to develop and validate quality assurance routines that identify maintenance required to the diagnostic system rather than an on-board alarm.

Two means of collecting vibration data are being implemented at HTTF. The TII/BFG diagnostic system saves raw digital data, while Metrum VHS digital tape recorders are used for making parallel raw data tapes. The test cell does not provide the airframe inputs or the rotor pass vibration inputs, but these frequencies are relatively low compared to the engine and gearmesh frequencies. The impact of this limitation on component- specific algorithms is restricted to the lowest speed components.

Rotor System Track and Balance/Diagnostics

Rotor Track and Balance using ROTABS has shown the ability to prescribe all necessary adjustments for rotor track and balance without the need for multiple flights or the use of optical trackers. The potential benefit obtained by eliminating the need for a tracker solves the operational and reliability issues associated with a full time on-board tracker. When extensive rotor changes are made, current RT&B systems use the tracker during ground turns to adjust the track to acceptable limits, then fly to balance. HIDS will investigate if ROTABS can similarly reduce track to acceptable values after major rotor system maintenance. Concurrently, flight testing will be performed to determine the capability of ROTABS and a tracker to detect rotor system faults. This program will provide valuable demonstration to help resolve the tracker issue for day-to-day rotor smoothing to improvement in aircraft comfort (pilot fatigue), airframe aging and avionics life.

Other Support

The Team approach has been utilized to develop, plan and support the HIDS effort. The propulsion and power drive system community as well as the diagnostics community have been heavily involved in determining what to demonstrate and how to put it to the test. SH-60 design engineers from the Naval Air Systems Command (NAVAIR), NAVAIR-WARCENACDIVTRENTON, Naval Aviation Depot (NADEP), Cherry Point, NC, Army Aviation and Troop Command (ATCOM), St. Louis, MO, Aviation Applied Technology Directorate, Ft. Eustis, VA and Sikorsky Aircraft, Stratford, CT have participated in planning the NAVAIRWARCEN-ACDIVTRENTON test cell efforts that are used to baseline and then challenge the diagnostic system. Diagnostic engineers from the same organizations have participated in program planning and system design reviews. Not for Flight Assets (NFFA) have been collected from Sikorsky, NADEP Pensacola, FL, NADEP North Island, CA, NADEP Mayport, FL, Corpus Christi Army Depot, Corpus Christi, TX and the Coast Guard for test.

Accomplishments

Accomplishments Summary

The KT-1 COTS hardware and software has been successfully installed and operated in both the NAVAIRWARCENACDIVTRENTON test cell and NAVAIWARCENACDIVPAX aircraft Bureau Number (BUNO) 162326 and 164176. The HIDS aircraft is flying with engine algorithms and recording cockpit instrumentation, control positions and alarm panel indications. The cockpit display can notify the pilot when there is an exceedance and the ground station reiterates those exceedances during data download into the ground station. The system has functioned as a flight data recorder providing a complete history of the flight. The ground station tracks serialized part numbers and times, correlates maintenance performed and part change data, and has a variety of report and plotting options. The system has continued to improve towards, and provide valuable data for, defining the specification of a production Navy system.

The KT-1 usage monitor and maintenance tracking system is also being used in the test cell to track what faulted components were run on any given day. The system has a list of all gear and bearing serial numbers which we can correlate to the faults. All component changes are tracked chronologically and the files are maintained by the test cell mechanics.

The KT-3 32 channel simultaneous sampling vibration acquisition have proved to be reliable and robust for both test stand and flight activities. The system recorded data in aircraft BUNO 162326 as a not-to-interfere secondary test. The hardware installation required that the system be stood on end to fit into the aircraft during the initial installation, and later was moved from the front of the aircraft to the rear. Data has been acquired from three airframes for the main gearbox and one aircraft for the entire system. A total of 85 hours of flight and 254 data sets have been recorded on aircraft 326. The helicopter drive system test facility at NAVAIRWARCEN-ACDIVTRENTON has operated for 396 hours of diagnostic system evaluation. Seven main gearboxes, seven input modules, two accessory modules, three intermediate gearboxes, four tail gearboxes and six engines have been tested and 31 faults have been run in the test cell. Extensive investigation into signal quality, and gain control has provided good confidence of the data acquisition quality. The analysis has provided a significant diagnostic capability for the detection of degraded components.

The data library consists of over 2000 sets of 32 channel simultaneous acquisitions of raw time series data with tachometers and accelerometers recorded together. This allows for time domain and frequency domain analysis to be performed post flight.

Compliance with Objectives

1. Demonstrate operation of an integrated diagnostic system for tracking usage of the helicopter power drive train. The capability to track the usage

of drive system components is illustrated in the Figure 6 histogram display generated by the groundstation. A composite distribution of main transmission and engine torque during three flights is shown.

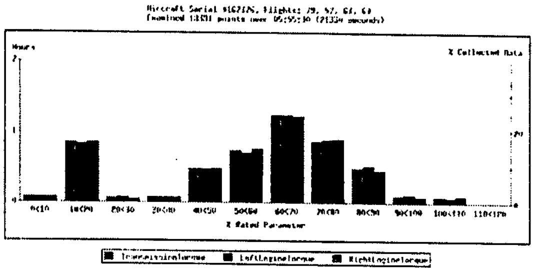


Fig. 6. Life Usage Tracking Window.

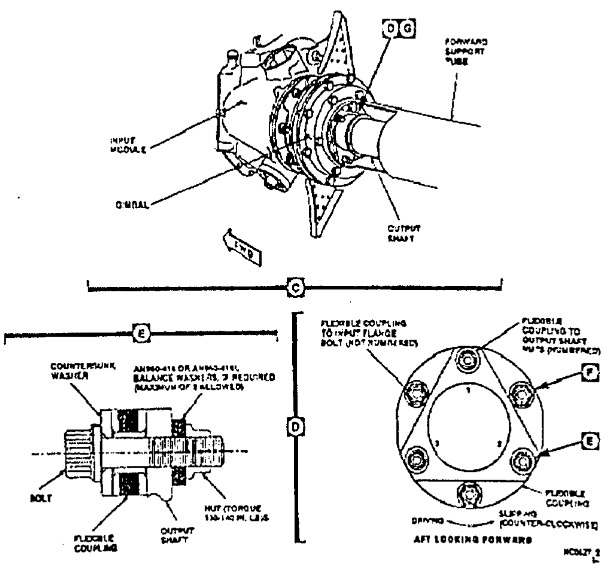


Fig. 7. High Speed Shaft Interface.

2. Evaluate the ability of the diagnostic system to identify localized faults in an entire drive system. The HIDS system has demonstrated the ability to identify localized faults on a number of H-60 drive system components. The engine high speed shaft/input module interface (see Figure 7) has been a problem area, where the difficult to inspect Thomas Coupling disc pack has suffered several failures. The Figure 8 engine high speed shaft (with cracked Thomas Couplings) was removed from the fleet and tested at Trenton. Figure 9 illustrates baseline test data with good driveshafts, and the degraded component installed at the starboard engine location for one acquisition at run number 31. The HIDS system detects the fault and isolates it to the starboard side. This provides a rationale for providing a cockpit alert for critical, rapidly degrading components. The HIDS system also detected a fleet removed input module suspected of being an every-other-tooth gearmesh candidate. These gearboxes were emanating a strong tone at one-half the normal gearmesh frequency, and it was believed this tone was contributing to premature removals of the mating T700 engines due to torque reference shaft wear. Figure 10 exhibits a gear health indicator (algorithm) of such a component tested at Trenton which shows baseline and fault (run numbers 149 through 170) data.

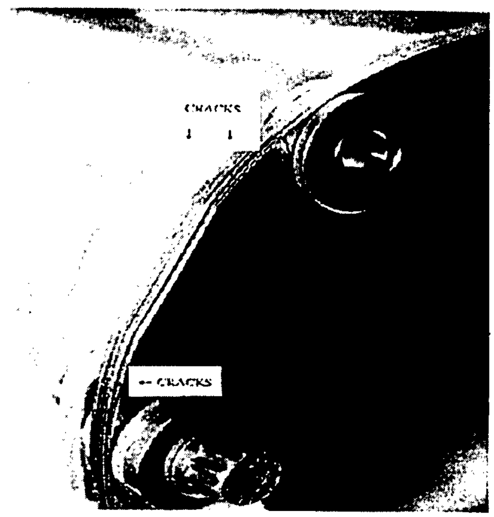


Fig. 8. Cracked Thomas Coupling.

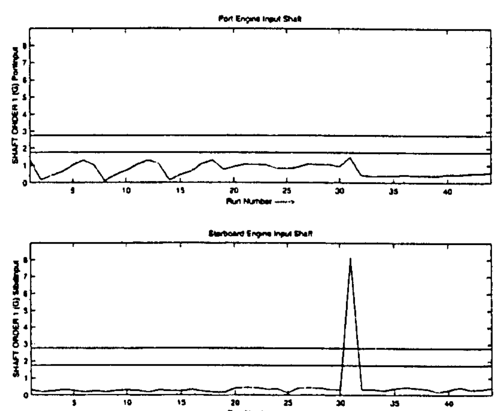


Fig. 9. Degraded Shaft at Position 31.

3. Evaluate the diagnostic algorithms for cracked gear fault identification and sensitivity. A critical part of the HIDS program is to demonstrate the detection of catastrophic gear faults. The most serious of which are root bending fatigue failures.

Depending upon gear design, this type of crack can either propagate through the gear tooth causing tooth loss, or through the web causing catastrophic gear failure and possible loss of aircraft. A means used in the helicopter community to promulgate this type of investigation is to weaken the tooth by implanting an Electronic Discharge Machine (EDM) notch in the gear tooth root. This action creates a localized stress concentration at the tooth root in an effort to initiate a crack. The HIDS team had previously attempted this test on other gear teeth, but with no success. Discussions with the transmission design departments at Agusta Helicopters and Boeing Helicopters assisted us in determining optimum notch placement. Figure 11 is a cutaway of the SH-60 intermediate gearbox. Two EDM notches (.25" Length x 006" Width x .040" Depth) were implanted along the length of the intermediate gearbox (IGB) gear tooth root by PH Tool of New Britain, PA. The location of the notches is critical as they were implanted where the gear tooth root bending stress is greatest.

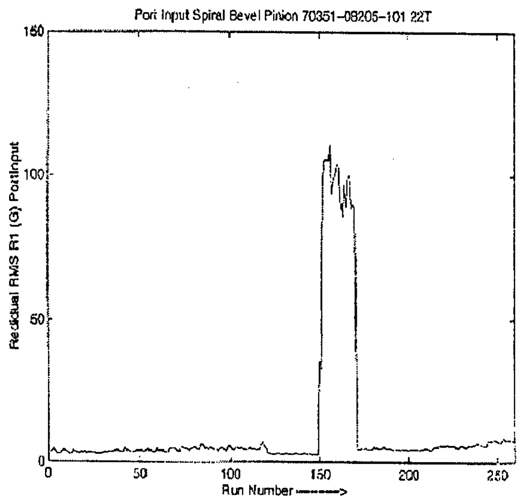


Fig. 10. Response for Half Gearmesh Anomaly.

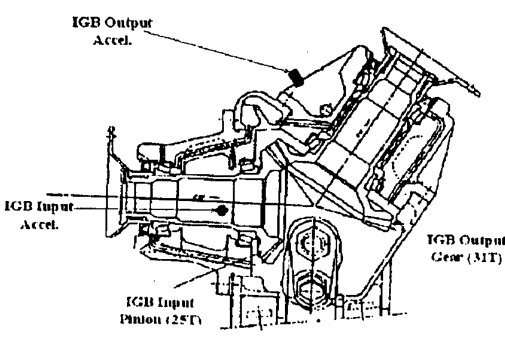


Fig. 11. SH-60 Intermediate Gearbox Cutaway.

Fig. 12. Cracked Intermediate Gerabox Pinion.

The test was run at 100% tail power for a total of 2 million cycles, when testing was terminated prior to gearbox failure when a gross change in the raw FFT spectra was observed on the HP36650 Spectrum Analyzer. Subsequent to test termination the gearbox was disassembled and inspected. The input pinion's faulted tooth exhibited a crack initiating from the tooth root and extending through the gear web and stopping at a bearing support diameter. Figure 12 exhibits the subject pinion at the end of the test. There is a void at the toe end of the notched tooth where a large section of the tooth broke off, and a through web crack extending to the bearing support diameter. No indication from the gearbox chip indicator was observed.

A review of the diagnostic results shows the TII/BFG model based algorithms successfully detect the presence of the gear tooth fault. Figures 13, 14 and 15 respectively exhibit "Component Condition" and the early and late responding health indicators from which it was derived. After indicating a healthy gear for roughly 267 minutes (most acquisitions were acquired 15 minutes apart), the indicator levels raised steadily for the next 139 minutes, thereafter exhibiting sharp changes in level until test termination at 548 minutes (Ref. 4 discusses indicator results of another pinion tooth fault). Test results illustrated an EDM notched tooth behaves much like adjacent teeth until the part is fatigued and a crack develops. The crack effectively weakens the tooth in bending,

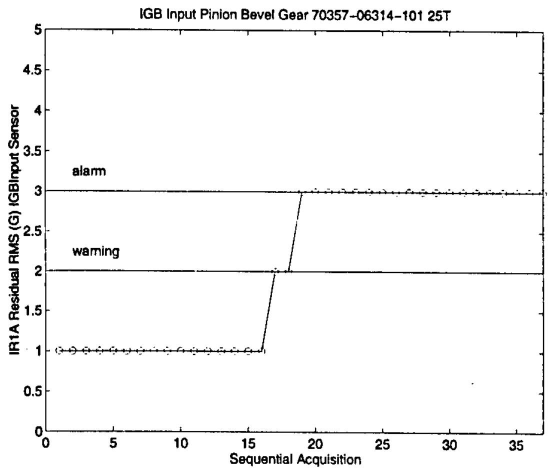


Fig. 13. IGB Pinion Component Condition.

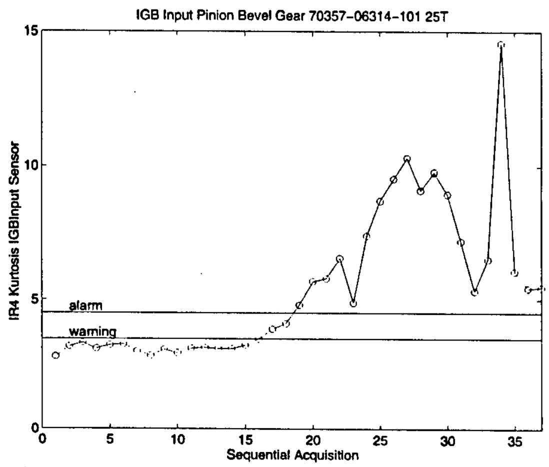


Fig. 14. Early Responding Health Indicator.

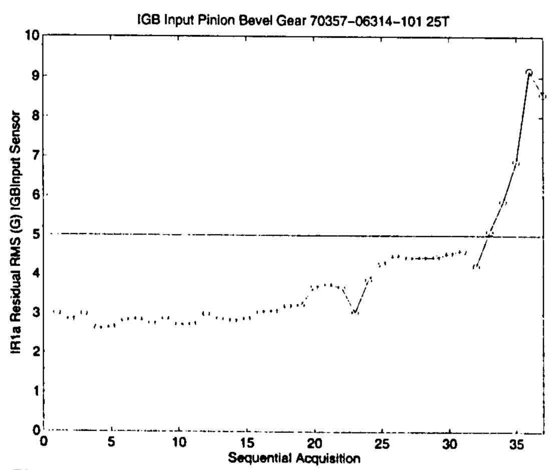


Fig. 15. Late Responding Health Indicator.

causing the faulted tooth to share load unequally with adjacent teeth. Depending upon the crack path, other dynamic anomalies are manifested. Also, synchronous averaging techniques employed in model based diagnostics can "filter out" non-synchronous vibration providing a health determination of a specific component.

A root bending fatigue propagation test was repeated on a main transmission input pinion. This test promised to be a more challenging effort for several reasons. First, the main transmission module is a larger and more complex system than the intermediate gearbox. The background noise is greater and the fault is located deep inside a larger housing. The gear form was also different. The intermediate gearbox pinion has a large web, where the main module pinion teeth are closer to the shaft centerline and therefore has a great deal of support at the tooth root. These observances made, the HIDS team determined to investigate the crack propagation properties of the more robust gear form.

Two EDM notches were implanted in the root of one geartooth and run for 12 million cycles at 110% power, removed and inspected, and then tested for another 10 million cycles. After 12 million cycles, small cracks less than 2mm in length emanating from the notch corners were present. Figure 16 exhibits the pinion after another 10 million cycles. A large part of the faulted tooth has broken off, and a crack propagated the length of the part forward (toe end), and aft (heel end) to the bearing support. No indication from the gearbox chip indicator was observed.

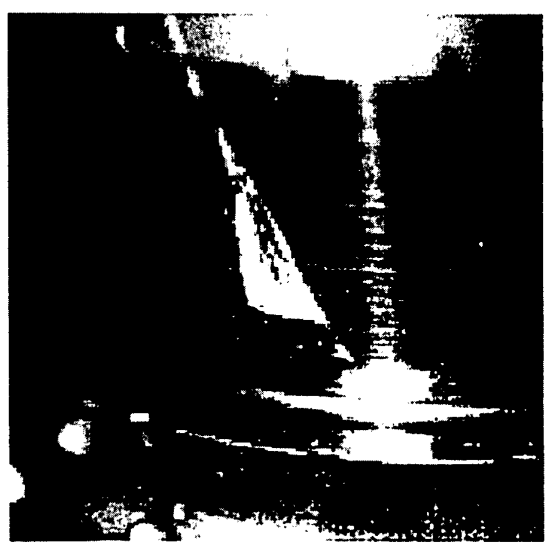


Fig. 16. Main Transmission Input Pinion Crack.

Figure 17 shows an indicator response for the test. Run numbers 1-206 are data from the first gearbox build, and run numbers thereafter from the second. It is interesting that key fault response indicators reached only half the level as for the IGB fault. Speculatively speaking, this may be due to the fault being deeper inside the gearbox, but is most probably due to the other main module pinion emanating "healthy" synchronous gearmesh tones and masking indicator response.

It is presumed the steep increase can be attributed to either the gear tooth breaking off, or the crack propagating through the web. It is indeed impressive that these components held together considering their condition and the loads transmitted.

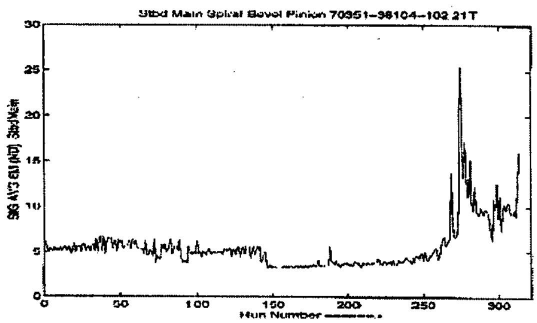


Fig. 17. Response to Main Module Pinion Fault.

These tests demonstrated (1) the HIDS diagnostic algorithms successful early detection of root bending fatigue failures, (2) chip detectors are unreliable for the detection of classic gear failures caused by root bending fatigue, (3) H-60 drive system components are particularly robust, and (4) root bending fatigue cracks on gear tooth forms such as the main module pinion can propagate through the web (vice only the tooth) to a catastrophic condition.

4. Quantify the level of signal for a known defect size to develop operational limits and trending for the SH-60 drive system. As discussed above, the IGB root bending fatigue failure provided excellent results in component fault detection and condition assessment. Figures 13, 14 and 15 exhibit the gear "Component Condition" indicator, and two gear health indicators which determine the component condition. The IR4 Kurtosis indicator provides early warning of a local gear tooth anomaly, and the IR1a indicator is excited as the gear tooth crack has propagated to a severe condition. These indicators could therefore be integrated into the diagnostics package as early warning and impending failure indicators respectively.

5. Evaluate the diagnostic systems ability to identify a degraded performance engine and damaged engines removed for cause. Two USCG T700-GE-401C engines were removed from the fleet and provided to Trenton for engine algorithm investigation. Engine serial number 366497 was removed from the fleet at approximately 25 degrees C off specification, and serial number 366622 approximately 45 degrees off of spec. Results from these tests showed the algorithm provided a constant, reliable value at powers between 60-90% (see Figure 18). Considerable data scatter was present, and a smoothing algorithm was recommended. The air data correction (.95 exponent) also appeared to cause divergence at low ambient temperatures, and an exponent of .65 provided improved results (see Figure 19). The algorithm values however estimate the engine performance for both engines to be 20-30 degrees C below actual, suggesting a bias correction is required.

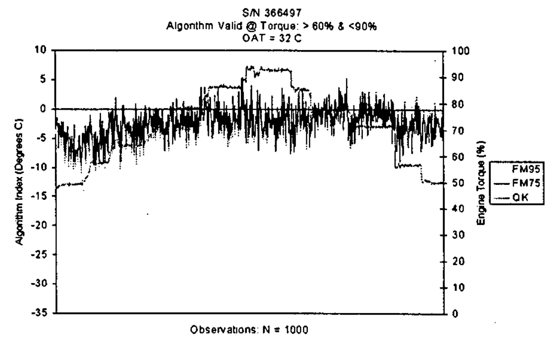


Fig. 18. T700 Engine Algorithm Results.

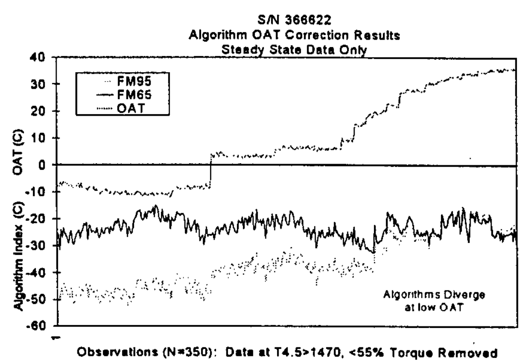


Fig. 19. Algorithm Correction Results.

6. Evaluate diagnostic system sensitivity to defects and faults in tail drive shafts and bearings. Hanger bearing assemblies are used to support the helicopter tail drive shaft. The main components of the assembly consist of a grease-packed sealed ball

bearing that is pressed into a viscous damper bladder and supported by a housing that mounts to an air-frame interface. The bearing is expected to be lightly loaded since it doesn't support any significant radial or axial loads, though those imposed from imbalance and misalignment occur in-service. Figure 20 shows the hanger bearing assembly and associated accelerometer installed at the number 2 location in the tail drive system. Since the viscous damper is in the vibration transmission path, there was concern it would inhibit the transmission of high frequency tones from the bearing to the vibration sensor.

A fleet removed hanger bearing with a very light click was installed in the HTTF. There was considerable opinion that the click was due to dirt in the bearing. 12.7 drive system operating hours were accumulated and 129 data points were acquired. Figure 21 shows a representative envelope spectral plot for the fleet rejected hanger bearing.. A fault clearly exhibits itself by the strong tones at frequencies specific to the inner and outer race defect frequencies and also at shaft speed. By comparison, fault-free hanger bearings did not generate bearing defect frequencies. The Figure 22 indicator is derived from the information contained in the spectral plot, and presents data from four different bearings which were installed in the #2 hanger bearing location. Data from the fleet rejected bearing is easily identifiable between run numbers 199 through 325. Note that the viscous damper attenuation concern did not materialize

Post test inspection of the bearing revealed that the inner ring was fractured as shown in Figure 23. Also, the bearing was found to have about 1.5 grams of grease remaining, which is within the range normally found in bearings operating to their 3000 hour overhaul life. Hanger bearings with inner race fractures have been known to eventually purge all the grease through the fracture leading to overheating, seizure, and loss-of-aircraft.

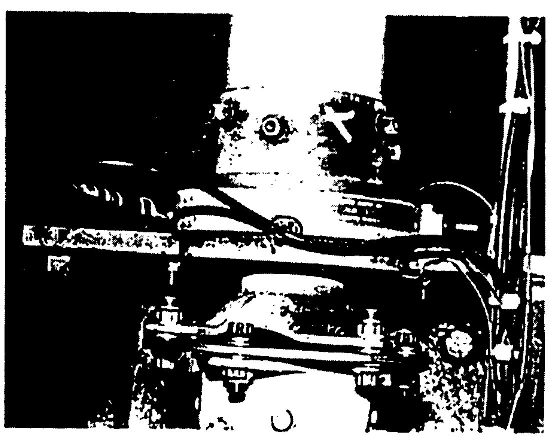


Fig. 20. Hanger Bearing Assembly.

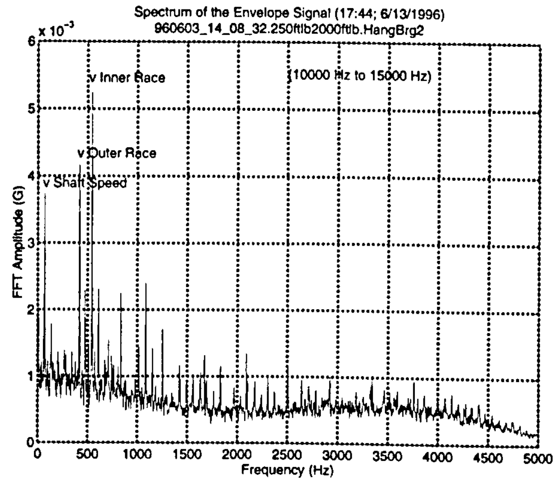


Fig. 21. Rejected Hanger Bearing Spectral Plot.

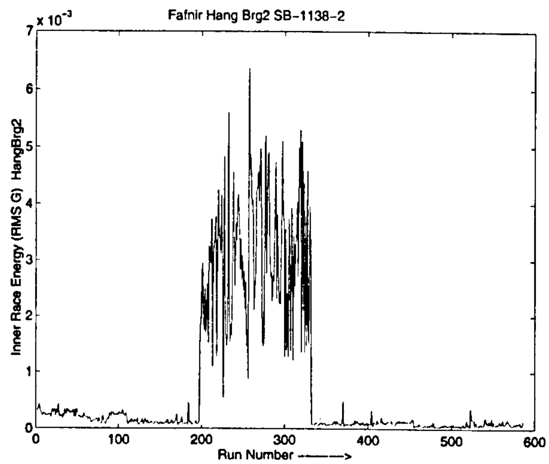


Fig. 22. Hanger Bearing Inner Race Energy.

Fig. 23. Post-Test Condition of Hanger Bearing.

7. Evaluate the diagnostic systems sensitivity to bearing defects in gearboxes. The spalled integral raceway bearing (P/N SB 2205) is the most common dynamic component cause for gearbox removal in the H-60 community. This fault is particularly challenging as it is located deep inside the main transmission, (see Figure 24) suggesting it would be difficult to detect. Figure 3 illustrates the SH-60 main transmission system and respective vibration accelerometer locations. The Figure 25 fleet rejected component was installed in the Trenton test facility starboard location. Bearing condition for the starboard and port main accelerometer locations are presented in Figures 26 and 27 respectively. The starboard main condition indicator toggles into the alarm position when the fault is implanted at acquisition number 254 and reverts back to the okay position when the fault is removed at acquisition number 300. The port main indicator is also sensitive to this fault because the sensor is located on the same structural housing member, and is rotated about 90 degrees around the housing from the starboard main sensor. The port indicator serves as a confirmation of the starboard condition. Enveloped kurtosis is the main indicator used to evaluate bearing condition for this fault. One of the keys to obtaining meaningful results with this technique is to envelope an appropriate frequency range. The frequency range used in this analysis was determined analytically as well as experientially. Figures 28 and 29 respectively exhibit the Kurtosis values of the primary (stbd main) and secondary (port main) sensors for the bearing SB-2205 fault.

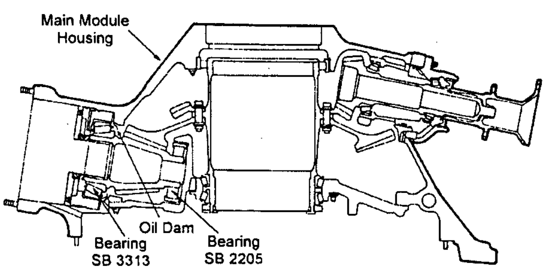


Fig. 24. Locations of SB-2205 and SB-3313 Bearings in the Main Module.

8. Evaluate the diagnostic systems ability to identify oil cooler blower faults. This test was recently performed by deliberately imbalancing the blower by attaching weights to the fan blades. The imbalance did not manifest itself in the data and recent conversations with Sikorsky Test Group suggest insufficient imbalance was implemented during the test.

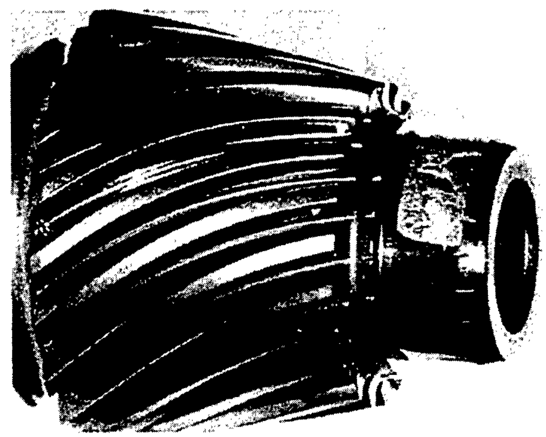


Fig. 25. Main Module Input Pinion with Spalled Integral Raceway Bearing SB 2205.

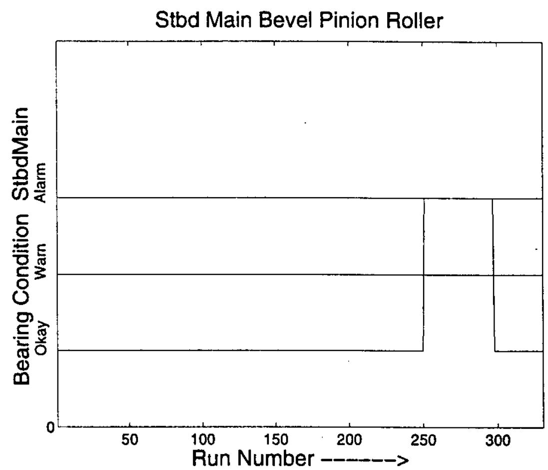


Fig. 26. SB 2205 Condition Call from Starboard Sensor.

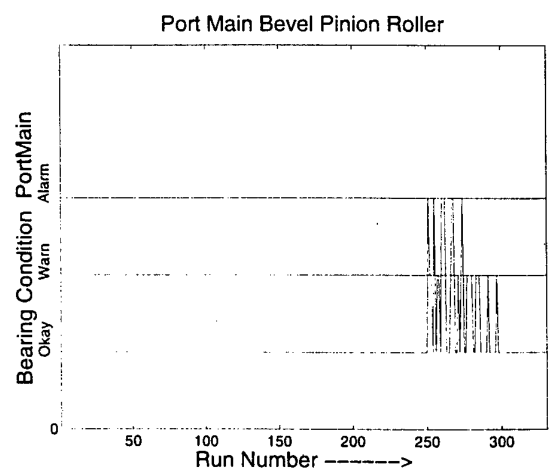


Fig. 27. SB 2205 Condition Call from Port Sensor.

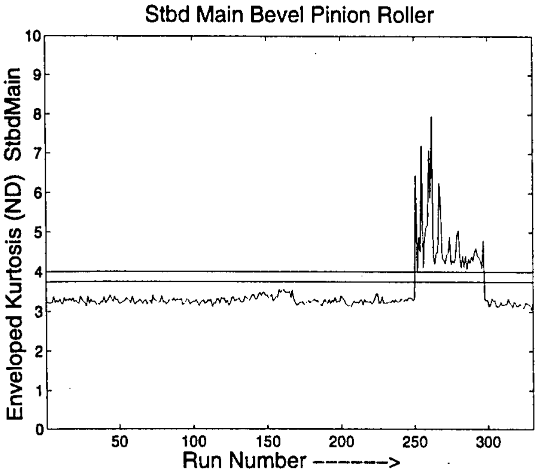


Fig. 28. SB 2205 Starboard Main Kurtosis Trend.

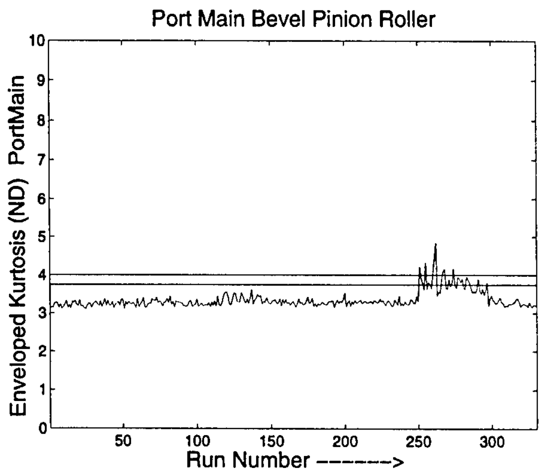


Fig. 29. SB 2205 Port Main Kurtosis Trend.

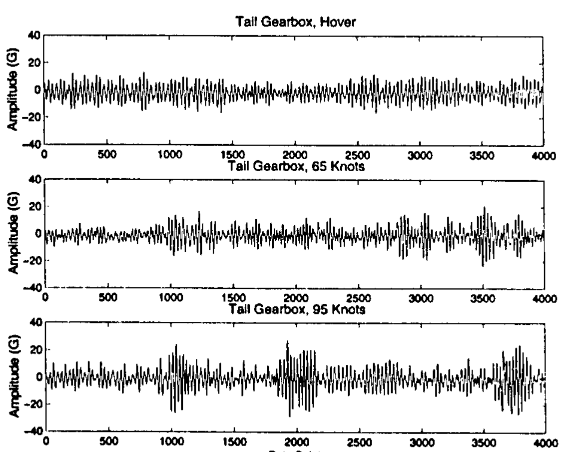


Fig. 30. Dissimilarity of Tail Gearbox Time Series Data for Various Flight Regimes.

9. Evaluate variability of data across flight regimes (including torque and weight variations). Figure 30 exhibits time domain tail gearbox vibration data at different flight regimes. There is considerable difference in the signal between forward flight and hover. This introduced considerable scatter in the algorithm indicators. It was determined a large main rotor 4/rev component (rotor wash) is interacting with the tail pylon in forward flight, which is causing this data instability. This and other flight regime nuances are being investigated.

10. Evaluate sensor placement sensitivity for the various defects. The objective is to minimize the total number of sensors required to identify faults large enough to require maintenance action and to increase robustness by verifying use of secondary sensors. The test of bearing SB 2205 provided an interesting study for sensor placement. At the time of test, the stbd main was the primary sensor for the stbd SB-2205 bearing, and the stbd input sensor was the secondary. Test results however showed otherwise. Figure 31 shows that the enveloped kurtosis of the stbd input sensor does not respond to the fault, whereas the port main sensor does (see Figure 29). Based on results from this test, the port main sensor was then mapped as the secondary sensor for the stbd SB 2205 bearing.

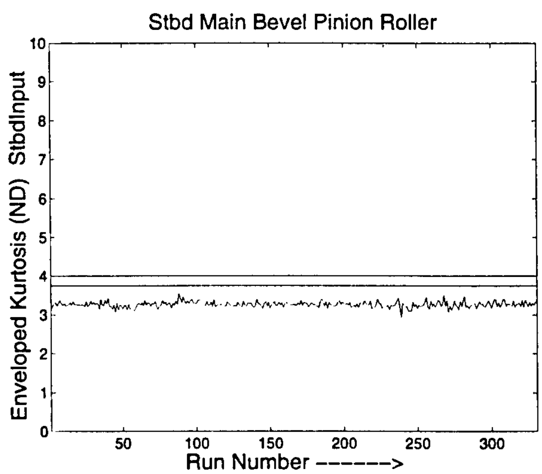


Fig. 31. SB 2205 Starboard Input Kurtosis Trend.

11. Determine ambient temperature effects upon diagnostics. The Trenton HTTF is capable of operating at temperatures from +20F to 100F. Many test configurations were tested at this temperature range. Also, data was acquired immediately upon reaching test conditions and prior to the gearbox reaching operating temperatures. For no-fault data, data acquired during cold temperatures fall within the existing "ambient" distribution. Figures 14 and 15 exhibit a

knee in the upward trend at approximately 360 minutes (acquisition 23), during the IGB cracked pinion test. This data point was the first of the day, acquired before the gearbox reached operating temperature.

12. Support The Technical Cooperation Program (TTCP) in evaluating new and emerging technologies in diagnostics. As stated, the HIDS team has coordinated with AMRL and the UK MOD to share test hardware, data, results, and engineering expertise. Digital vibration data acquired on wide band Metrum tape recorders was provided to the UK MOD for use as evaluation criteria in a recent RFP for HUMS systems on the Chinook helicopter. Tapes have also been provided to the Australian Aeronautical Maritime Research Laboratory (AMRL) for diagnostic evaluation and development. Tapes have also been provided to Sikorsky Aircraft in a reciprocate agreement in exchange for implanting faults at their overhaul facility. As stated previously, the HIDS team has coordinated with AMRL on the evaluation and development of engine performance algorithms. The oil debris monitoring evaluation was also coordinated with TTCP.

13. Evaluate the potential for detecting misalignment, bad pattern and improper shimming during assembly that may be the cause of premature damage in mechanical systems. Misalignment and imbalance testing have been performed on a number of drive system components. Specifically, the engine high speed shaft/input module assembly has been investigated under these conditions and findings were documented (Ref. 5). Other similar tests (some naturally occurring) were recorded. Gearbox gear pattern shim surveys were also performed. Test results are pending data review.

14. Develop seeded fault data library that can be used to evaluate systems in the future without repeating the test program. The HIDS program has provided a wealth of knowledge and understanding of the implementation of mechanical diagnostics. Though not immediately quantifiable, the HIDS testing has identified many optimized test methods and fleet implementation issues. Though not eliminating the need of seeded fault testing for other drive systems, the scope of work can be more precise and reduced. For the Integrated Mechanical Diagnostics Commercial Operational Savings and Support Initiative (COSSI), the HIDS data is being distributed to various institutions to develop and evaluate transmission planetary system gear and bearing algorithms.

15. Demonstrate ROTABS rotor track and balance. Four trials of ROTABS were undertaken to investigate the ability of the ROTABS concept to effect adequate control of blade track and balance. The aircraft used for these trials was a Sikorsky SH-60B, Bureau Number 162326. All operations were conducted at Patuxent Naval Air Station under the auspices of NAVAIR. The vibration and track data presented in this paper were recorded by a standard U.S. Navy Vibration Analysis Test Set (VATS) installed in the aircraft for these tests. VATS records vertical vibration at the same locations (copilot's left shoulder and pilot's right shoulder) as the single and dual-axis ROTABS sensors.

VATS vibration data are displayed as "A + B", and "A - B". These terms refer to the mean [more precisely, (A + B)/2] and difference (A - B) of the 1P vibration at the two locations near the pilot and copilot. Using rigid-body terminology, "A + B" is a measure of the vertical motion of the fuselage at the pilot's and copilot's seats, while "A - B" is a measure of the rolling motion of the fuselage about a longitudinal axis. VATS also collects blade track data from a line-scan camera aimed out of the left-side window of the aircraft. This camera is held and operated by a member of the crew. Data was collected at the following conditions: (1) on the ground, (2) Hover Out-of-Ground Effect (HOGE), (3) 120 knots, (4) 140 knots and (5) maximum forward velocity (VH).

Single Pitch Rod Adjustment Test

Table 1 shows Rotor Track and Balance (RTB) vibration and track data for the initial flight.

Table 1. RTB Data for Initial Flight

	A-B (ips)	A+B (ips)	Track (in)
Ground	0.16	0.19	1.0
HOGE	0.19	0.12	0.8
120 kts	0.15	0.25	1.5
140 kts	0.13	0.24	1.0
VH	0.06	0.21	1.2

After this flight, the Pitch Control Rod (PCR) on the blue blade was extended 10 clicks. The vibration and track data collected during the flight following this adjustment are shown in Table 2.

Table 2. RTB Data after Blue PCR Extension

	A-B (ips)	A+B (ips)	Track (in)
Ground	0.46	0.22	2.8
HOGE	0.60	0.26	2.8
120 kts	0.37	0.69	3.6
140 kts	0.43	0.78	4.7
VH	0.53	0.84	4.5

Table 3 shows the vibration and track data recorded during the flight following the implementation of the ROTABS adjustments. In every operating regime the vibration is greatly reduced, and with the exception of VH, is lower than recorded before the rotor was thrown out of balance by extending the pitch rod. The ROTABS adjustments reduced the track spread from a maximum of 4.7 inches to 2.3 inches.

Table 3. RTB Data after ROTABS Adjustments

	A-B (ips)	A+B (ips)	Track (in)
Ground	0.13	0.11	1.2
HOGE	0.08	0.05	1.2
120 kts	0.06	0.10	2.0
140 kts	0.06	0.10	2.0
VH	0.16	0.21	2.3

Single Tab Bend Test

Following this flight the tab on the red blade was bent down 10 mils. The vibration and track data collected during the flight following this adjustment are shown in Table 4.

Table 4. RTB Data after Red Tab Bend

	A-B (ips)	A+B (ips)	Track (in)
Ground	0.31	0.03	2.1
HOGE	0.25	0.10	1.7
120 kts	0.14	0.34	4.3
140 kts	0.19	0.45	5.2
VH	0.38	0.61	5.9

Based upon the vibration recorded by KT-3 during this flight, the ROTABS computed adjustments were made. The vibration and track data taken during the confirmation flight are shown in Table 5. Again the vibration is greatly reduced in all operating regimes. The track spread is reduced from 5.9 inches to 3.3 inches.

Table 5. RTB Data after ROTABS Adjustments

	A-B (ips)	A+B (ips)	Track (in)
Ground	0.10	0.05	1.4
HOGE	0.07	0.08	1.3
120 kts	0.04	0.08	2.1
140kts	0.07	0.24	2.8
VH	0.15	0.21	3.3

Based upon the vibration data collected during this flight, a second set of ROTABS corrections were computed and made. The vibration and track spread recorded on the flight following the implementation of these adjustments are shown in Table 6.

Table 6. Second ROTABS Adjustment RTB Data

	A - B (ips)	A + B (ips)	Track (in)
Ground	0.16	0.21	1.6
HOGE	0.05	0.08	1.3
120 kts	0.04	0.10	2.2
140 kts	0.07	0.14	2.0
VH	0.08	0.21	2.7

Equal Pitch Rod Changes on Opposing Blades (Track Split) Test

As the third trial, a pair of equal pitch control rod adjustments on opposing blades, in this case the red and yellow blades, were implemented. The pitch rods on these blades were both lengthened 10 clicks.

Balanced changes of this sort have minimal or no effect on vibration at odd shaft orders (1P, 3P, and so on). They do affect vibration of even shaft orders (2P, 4P, etc.), and throw out the blade track.

Table 7 shows only the track spreads before and after the adjustments and after the ROTABS corrections. In all cases, the 1P vibration was less than 0.2 ips before and after all adjustments.

Table 7. Track Spread Summary

	Prior to PCR Adjustments	After PCR Adjustments	ROTABS Corrections
Ground	1.0	2.0	0.7
HOGE	1.1	2.4	0.8
120 kts	1.5	3.7	2.5
140 kts	1.1	4.0	2.3
VH	1.7	3.6	2.0

Paired PCR and Tab Bends (Track Split) Test

The fourth trial consisted of a paired set of PCR extensions and tab bends. The pitch control rods on the yellow and red (opposing) blades were extended 8 clicks, and the tabs on the same two blades were bent up 10 mils. Four flights were conducted including two sets of ROTABS corrections. Table 8 shows the track spreads recorded on these flights.

During this trial as well, the 1P vibration was substantially unaffected by the adjustments, and on all flights was less than 0.2 ips.

Table 8. Paired PCR and Tab Bends

	Before Change	After Change	First ROTABS	Second ROTABS
Ground	0.7	2.4	2.1	1.5
HOGE	0.8	2.7	1.7	1.4
120 kts	2.5	4.7	3.3	2.0
140 kts	2.3	4.3	3.3	2.0
VH	2.0	3.9	3.1	3.1

ROTABS was able to keep vibration within limits (below 0.2 ips) on all tests and track spread within limits except for the track split paired adjustments (3.0-3.3 mils). Test results reflect the coupling between blade flapping and rotor vibration particular to this specific type of helicopter as well as the accuracy with which specified blade adjustments can be implemented using approved methods and procedures.

16. Demonstrate automated engine health monitoring by automating the HIT check and implementing a real time engine performance algorithm. See item 5 above.

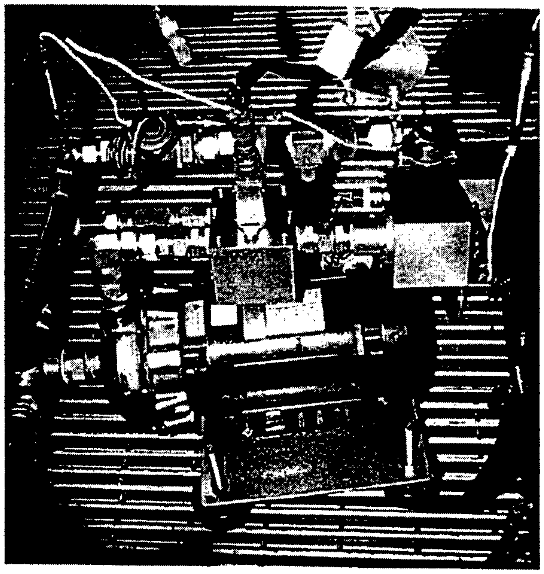


Fig. 32. Test Rig for Oil Monitoring Evaluation.

17. Evaluate as many currently available propulsion and power drive system diagnostic technologies as possible in test cell 8W and assess their relative effectiveness. Engineering evaluation testing of Stress Wave Analysis, Electrostatic Engine Exhaust Monitoring, Inductive Oil Debris Monitoring, Quantitative Oil Debris Monitoring, Optical Oil Debris Monitoring, and Acoustic Emission have been done in parallel with HIDS testing evaluation at Trenton. Two of these efforts are US Army SBIR efforts. As a means to evaluate the IDM and QDM MKII oil debris monitoring systems simultaneously, a modified main transmission lubrication scavenge apparatus was provided by Vickers Tedeco (See Figure 32). The system attaches to the main transmission module at the normal chip detector location and a positive displacement pump adds sufficient head to pump the oil through an external plumbing arrangement. Sump oil enters the pump, IDM, QDM MKII, and finally the production main module chip detector and returns to the transmission. A fine mesh screen is included to capture particles that are not captured by the QDM MKII and main module magnetic detectors. The Figure 25 main transmission input pinion with a spalled integral bearing raceway was used as a tool to generate debris for the evaluation. This test (Ref. 6) found the fault generated particles much smaller (5-20 microns) than what a typical bearing fault (>100 microns) is known to produce. This evaluation provided sensitivity and performance information.

18. Evaluate the data collected on-board the aircraft with the test cell data to validate the pertinence of test cell proven algorithms for use on-board an aircraft. As part of the HIDS program, drive system vibration data was acquired on 22 and 23 May and 30 August 1995 from SH-60 BuNo 164176 at NAVAIRWARCENACDIVPATUXENT. Data was also collected on two other SH-60 aircraft using the same data acquisition system. The data was acquired primarily to support a next generation diagnostic effort based on neural network technology and designated the Air Vehicle Diagnostic System (AVDS) program. The intent was to acquire raw vibration data on fault-free aircraft to use as a means for baselining the neural network process. For aircraft BuNo 164176 a total of 46 separate acquisitions were taken at several different flight conditions including ground turns, hover in-ground effect, hover out-of-ground effect, straight and level and descent. Torque ranged from 28-100%. Approximately one month after the May data had been acquired from BuNo. 164176, HIDS project personnel were informed that the aircraft had a history of setting off the main transmission chip detector light. The chip detector events prompted an analysis of vibration data collected from BuNo. 164176 using HIDS diagnostic algorithms. The same analysis was also conducted on one of the other aircraft , namely BuNo. 162326, to provide a baseline for comparison to aircraft BuNo. 164176. Representative envelope spectral plots of baseline and faulted aircraft data are shown in Figures 33 and 34 respectively. The fault clearly exhibits itself by the strong tones at frequencies specific to the main bevel pinion tapered roller bearing (SB 3313) both in the test cell and the aircraft. The Roller Energy indicator for the aircraft data is displayed in Figure 35.

The analysis clearly indicated a fault in the rolling elements of the starboard main bevel input pinion tapered roller bearing, P/N SB 3313 (see Figure 24 schematic for location) and represented a safety-of-flight concern. Further confirmation of fault location was provided by chip elemental analysis, conducted by Sikorsky Aircraft, which determined that the chips were CBS 600 steel indicating that this bearing was one of several possible sources of the chips. Based on the analysis, the HIDS team strongly recommended that flight operations on aircraft BuNo. 164176 cease and the main gearbox be removed and sent to NAVAIRWARCENACDIVTRENTON for installation and continued testing in a test cell environment to provide a comparison to flight test data (see Figure 36 for test cell data). Moreover, the urgency to remove the gearbox from service was a result of the HIDS team assessment that the presence of the oil dam (P/N 70351-38124-101), adjacent to the bearing was a barrier to chip migration thereby (1) preventing the chip detector from indicating the true severity of the failure development and (2) creating a reservoir of chips which may act to increase the failure progression rate. Action was taken to comply with the recommendation. Subsequent teardown and inspection confirmed that 13 of the 23 rollers in the bearing were severely spalled as shown in Figure 37. Inspection revealed a large amount of debris harbored by the oil dam, confirming the HIDS team suspicion that the oil dam acted as a chip reservoir.

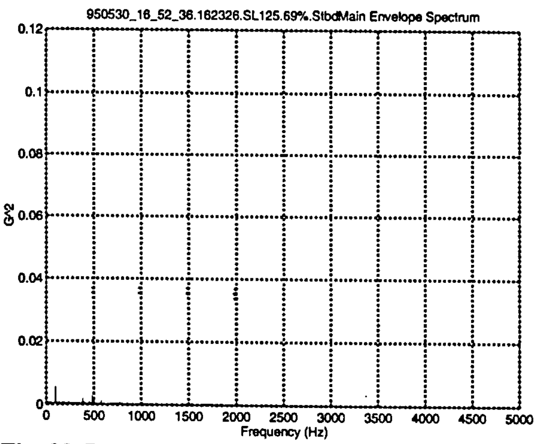


Fig. 33. Baseline Spectrum for Bearing SB 3313.

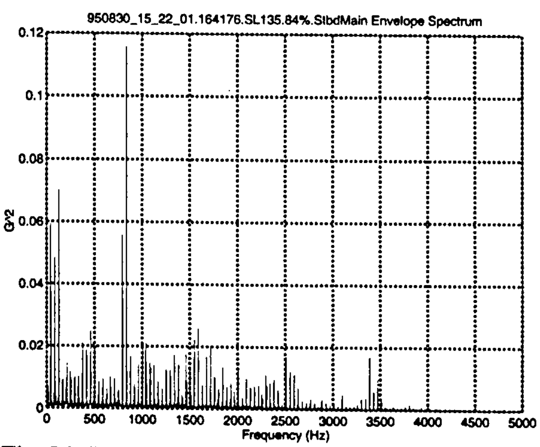


Fig. 34. Fault Spectrum for Bearing SB 3313.

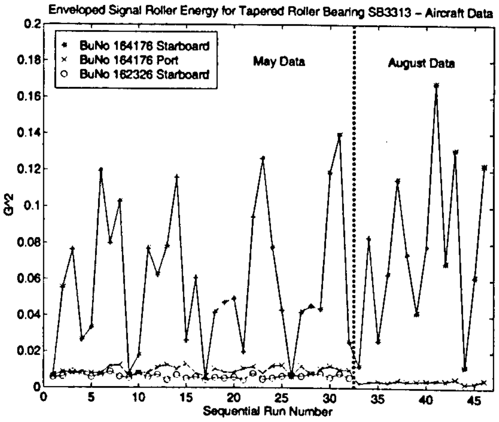


Fig. 35. Enveloped Signal Roller Energy for Bearing SB 3313, Aircraft Data.

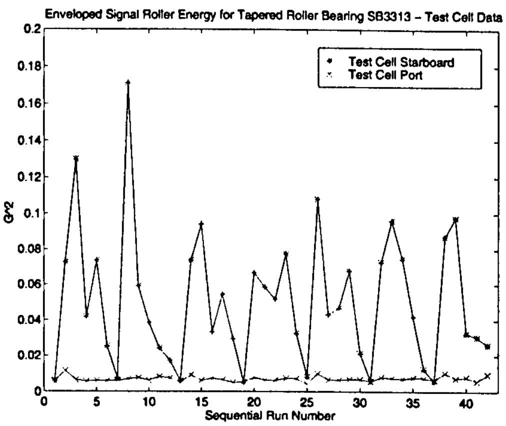


Fig. 36. Enveloped Signal Roller Energy for Bearing SB 3313, Test Cell Data.

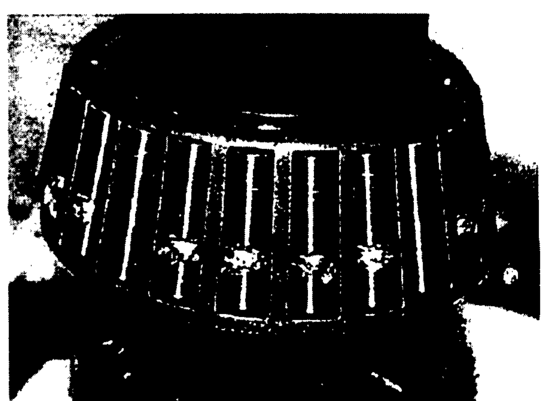


Fig. 37. SB 3313 Removed from PAX Aircraft.

19. Categorize diagnostic results with respect to aircraft flight regime to define optimized system acquisition and processing requirements. Review of Figures 35 and 36 reveals a great deal of scatter in the

value of the faulted bearing indicator. This is due to the differences in flight regime and torque. A fault must be loaded to excite a discrete frequency, and a determination of what regimes produce satisfactory results is needed.

20. Demonstrate automatic acquisition of mechanical diagnostics and ROTABS via flight regime recognition. Automatic acquisition via regime recognition of drive system diagnostics data and rotor track and balance data have been demonstrated.

21. Demonstrate real time, on-board analysis and health assessment of drive system gears, shafts, and bearings. The real time data acquisition and analysis for all channels was demonstrated in the test cell in May 1995, and in the aircraft in May 1997. The KT-3 system was found to have a hardware processing limitation on the Shamrock quad DSP which prevented it from calculating the optimum length of data for bearing analysis for all aircraft bearings in parallel. This shortfall has been overcome during the H-53E Early Operational Assessment by the implementation of a Pentium processing board on the second generation KT-3.

22. Demonstrate structural usage functionality in groundstation. Figure 38 is a view of the groundstation window which the HIDS team and BFG have worked to develop. By using flight regime recognition and structural usage calculations, component damage can be calculated in near real time. A rotor system component with flight regime, flight hours, and damage assessment to date is displayed.

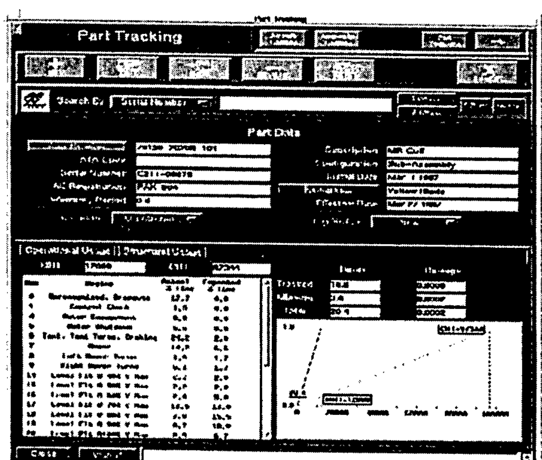


Fig. 38. Groundstation Window.

23. Demonstrate the diagnostics ability to reduce component "false removals" and trial and error maintenance practices. Several fleet removed components which were tested at Trenton were found to be fault free. Four hydraulic pumps removed for oil pressure problems were found to operate normally in the Trenton test cell. An input module removed for chip generation was tested. No debris was generated, and the diagnostics indicated a healthy component. Subsequent teardown inspection at Sikorsky revealed no dynamic component degradation.

24. Demonstrate methods that reduce false alarms and improve component condition assessment. Numerous indicators have been developed to quantify health of the drivetrain components. Rather than use each of these indicators in isolation, practicing data fusion can derive additional benefit. Multivariate Analysis is currently under investigation and has been shown to increase robustness of condition calls. Tighter control limits can be established by taking advantage of underlying correlation among the indicators while developing a composite indicator that changes by orders of magnitude in the presence of a fault.

Conclusions and Recommendations

1. This collaborative effort has provided significant benefit to the US, Australia and UK, in the form of a rich vibration database, diagnostic reports and integrated HIDS lessons learned.

2. The U.S. Navy has taken an aggressive approach in the evaluation and validation of propulsion and power drive system diagnostics through the HIDS effort.

3. Raw digital time series data files are a valuable asset for evaluating the performance of diagnostic algorithms, and are necessary to identify system problems that result in false alarms. The data allows for development of system built in test features to negate potential false alarms, and provide system maintenance direction.

4. Technology to monitor and diagnose aircraft systems exists today, but reliable vibration diagnostics requires the capability to record raw data for baseline development of aircraft types to establish production system algorithms and thresholds. Raw data collection capability and detailed analysis prior to release of aircraft from overhaul is a necessary part of system development and fleet support.

5. Testing needs to continue in the HTTF to expand the database and refine the correlation of defect

size to algorithm output level for alarm threshold settings on the SH-60 and H-53E. Continue refinement of vibration diagnostic algorithms and QA/QC routines and implement into aircraft system. Expanded testing to include the following:

(a). Testing of fleet gearboxes rejected for vibrations or chips. Support from the Class Desk and Depot has been coordinated for identification and testing of components.

(b). Continue testing of EDM notched gears and bearings for fault propagation testing at HTTF.

6. NAVAIRWARCENACDIVPAX needs to continue flying the HIDS system to continue evaluation of functional capabilities while developing recommendations and requirements for a fleet system.

(a) Ongoing work is required to improve correlation of engineering diagnostic outputs with component conditioning to effect meaningful fleet information and recommended actions.

(b) Expand diagnostic system data base for regime recognition and structural usage monitoring algorithms for the H-60.

(c) Validation and implementation of ROTABS technology in flight test aircraft. Survey other aircraft to expand database. Recommend procuring portable ROTABS system for aircraft survey to expand database. Maintenance procedures to minimize functional check flights need to be developed, allowing for regular rotor system improvement without maintenance down time. Small adjustments to the system on a regular basis is the maintenance concept that could negate the need for a dedicated functional check flight.

(d) Altitude flight testing and validation of T700 Power Performance Index algorithm to expand the data base for additional refinement of the on-board monitor of performance.

7. Expand system demonstration to leverage off the HIDS propulsion and power drive efforts to include the additional functions required by a fleet health monitoring and maintenance system, i.e. logistics and structures.

(a) Add an automated NALCOMIS interface that will update upon HIDS system download into the ground station. Expose fleet maintenance and NAESU personnel to capabilities for development of fleet friendly interfaces and functions. Incorporate the existing H-60 Integrated Electronic Technical Manuals (IETM) and develop a connectivity between maintenance actions recommended by diagnostics and the procedure in the IETM.

(b) Utilize the existing data acquisition system which records all of the required parameters for regime recognition and structural usage monitoring by including algorithms to calculate these functions in the HIDS system demonstration.

8. Testing for vibration analysis evaluation and validation in the NAVAIRWARCENACDIV-TRENTON HTTF has provided a tremendous foundation for a thorough understanding of the vibration characteristics and transmissibility between dynamic components of the SH-60 drive system. Future HTTF test efforts should require vibration databases to be established using the KT-3 raw vibration data system. Upgrade the HTTF to allow for testing of the CH-53E at full power. Provide vibration test facilities at overhaul as a quality assurance check and initial aircraft baseline for when the component is installed. These data records will provide component level baseline prior to installation on the aircraft.

9. Demonstrate groundstation interface with USCG Aircraft Computerized Maintenance System.

References

[1]Mission Need Statement for Integrated Diagnostic System, 3501 Ser 723/9197 of 25 Oct 93 and 3501 Ser NO2X/03100 of 18 Aug 93.

[2]NAVAIR AIRTASK A536360/052D/3W135-50000.

[3]Emmerling, W. C., "Helicopter Drive System Seeded Fault Test Program", AHS Rotary Wing Technical Specialists' Meeting, 26 Oct 93.

[4]Hardman, W. and Frith, P., "Analysis of a Severe IGB Tooth Fault Implanted in the 8W SH-60 Drive Train Rig", NAVAIRWARCENACDIVTRENTON-LR-PPE-95-7, Aug 95.

[5]Neubert, C., and Mimnagh, M., "Results of H-60 Helicopter Engine High Speed Shaft Assembly Imbalance Testing", NAVAIRWARCENACDIV-TRENTON-LR-PPE-96-4, Jun 96.

[6]Neubert, C., "Performance of QDM and IDM Oil Debris Monitors in a Full Scale Helicopter Transmission Test", NAVAIRWARCENACDIVTRENTON-LR-PPE-96-3, Mar 96.

Development and Validation of Algorithms for Engine Usage Monitoring Systems

M B Henderson and G F Harrison
Structural Materials Center, Griffith Bldg (A7), Room 2008
DERA, Ively Road, Farnborough, Hants GU14 0LX, UK

1. Summary

The design techniques used to provide the required component life with the safety levels imposed by the regulatory authorities are discussed. The development and validation of materials algorithms to determine the rate at which this life is consumed under service usage for the usage recorders are described. For lifing highly stressed engine components two of the most important variables are the piece of material that an individual component is made from, and the way in which the engine is used.

Engines contain a number of highly loaded parts which are only released for limited service usage. At the end of the design and development process each part is qualified to the extent that a service life is 'declared' by the design authority. When a component reaches this declared life it is withdrawn from use. Life limitation is based on assessment of the low cycle fatigue, thermal fatigue and/or creep service capability of the component under consideration and engine life consumption is based on the stresses, strains and temperatures experienced at critical areas depending on the mission profiles, engine intake conditions, individual pilot reactions and several other parameters.

2. Introduction

The primary objective of any lifing procedure is to ensure aircraft safety. Design techniques for highly stressed engine components, exposed to widely varying stresses under transient speed, thermal and metallurgical conditions, are needed to provide the required component life with the safety levels imposed by the regulatory authorities. The costs of failure can be measured in terms of the loss of missions, unplanned maintenance and increased manpower and ultimately the loss of human life. At the end of the design and development process each safety critical structural component is qualified to the extent that a service life is "declared" by the design authority. When a component reaches this declared life it must be withdrawn from use. In addition, the development and validation of materials algorithms are necessary to determine the service life consumption rates for the engine usage recorders. Engine usage monitoring is limited to damage mechanisms that are life limiting, and in current designs these include low cycle fatigue (LCF), crack growth and creep.

Two of the most important variables for consideration by the design engineer are the material from which an individual component is made and the way in which the engine is actually used during service. Qualification is the process whereby airborne equipments demonstrate that they meet the minimum specified safety requirements and these test methods, such as explosion tests and cutting the power turbine shaft, are developing as quickly as the science behind new component design and development.

The following paper describes briefly the failure point definitions and the methods available for determining the rate of damage accumulation. Engine usage monitoring systems and the simplified stress analysis and damage accumulation algorithms necessary to determine real time life consumption rates for fracture critical components are also discussed.

3. Component design and failure

The engine designer requires information about the intended engine duty cycle and the materials used. Inputs to the detailed component design come from associated thermodynamic, heat flow, materials and mechanical models, plus the mission specification(s) provided. After the engine and component cycles have been decided, and the physical space allotted to each component, the detail design is completed. The basic steps are: i) stress modelling, ii) materials characterisation, iii) component testing, iv) field experience, and v) component management and retirement policy. The main problems still to be answered centre on how the engine will really be used in service, as experience has shown that most military aircraft undergo changes in the way that they are operated. Figure 1 illustrates the complex nature of the loading sequence experienced and the effects of apparently small differences in engine usage cycle on the amount of damage accumulated during nominally the same mission. The service mission experienced by the engine of aircraft #8 is significantly more damaging than that for aircraft #1[1].

3.1 Defining a Safe Failure Point

Regardless of the details of the lifing methodology used, it is crucial that all declared lives are to a common safety level. In most cases, fracture critical component service lives are specified, such that with a confidence level of 95%, not more than 1 in 750 (or 1/1000) will achieve the defined failure point, traditionally based on a Life To First Crack (LTFC) criterion using a "first crack" depth of 0.38 mm. This was originally based on the effectiveness of crack detection techniques, since exceeded, but the definition provides a standard. This means that each turbine disc is removed from service at

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

the point in its life when it is calculated that a 0.38 mm deep crack (0.75 mm surface length) would be present in the weakest member of the total service population and each disc is treated as though it were the weakest in the fleet. This approach has ensured that service failures of engine discs are extremely remote. Most discs are retired, however, after consuming just a fraction of their available life (only ~40% on average). Even if inspection shows that no crack exists it is assumed that one will appear on the next engine cycle. The crack propagation phase is not considered.

Evaluation of specimen test and rig test failures is based on conventional log-normal distribution statistics which is used to estimate the mean, range and variance for the population. The main variable affecting the accuracy is sample size and due consideration enables a confidence interval (i.e., statistical correction factor) to be determined. The sample must be of adequate size to obtain enough information to show statistically that the sample itself is of a consistent and identifiable distribution and can be assumed to be representative of the original population.

3.2 Fracture Mechanics Based Component Failure

To achieve increases in fleet disc life beyond the point at which the first crack is deemed to occur, additional component design, inspection and management techniques are needed. These are generally based on fracture mechanics or damage tolerance methods and require predictable crack growth behaviour within the disc. By designing components to plastically deform at stress concentration features and thereby redistribute loads, the crack growth phase can be sufficiently large before disc burst occurs. Within the safe life context, many components are now designed on a "2/3 Dysfunction Life" to provide a constant safety ratio.

In the UK database approach [2], fracture mechanics is used to calculate stress intensities to combine both specimen and full scale component tests on different design features into a common database. An effective initial defect size is determined for each fatigue result and statistical analysis of the size distribution allows a maximum probable flaw size to be established. This is incorporated within a forwards crack growth calculation and minimum component lives are established for each design feature. In the US ENSIP (Engine Structural Integrity Programme) [3] approach for defect tolerant designs, declared lives are based on quantifying the available safe propagation life of fatigue cracks whose initial size is defined as the maximum defect size associated with a detection probability of 90% to a confidence level of 95% and appropriate to the particular equipment used in the NDI inspection. The inspection interval is set to 1/2 of this dysfunction life, which must be physically demonstrated in full scale accelerated engine tests (ASMET). This approach allows advantage to be taken of new inspection techniques which are one of the major limiting factors.

In most cases the LTFC and the "2/3 Dysfunction" concepts provide closely similar declared lives. The constant safety margin of the 2/3 Dysfunction criterion is preferred because the use of higher strength materials at increased operating stresses, has the effect of reducing the initiation crack size associated with crack growth to below the 0.75 mm surface length of the engineering "first-crack". Where the 2/3 dysfunction life exceeds the LTFC, fracture mechanics crack propagation methods may be used to determine the available service life beyond "first crack". For potential sub surface locations, such as the disc mid-cob region, probabilistic crack growth methods are used to account for the effects of defects of specific sizes at specific locations.

4. Overview of Engine Usage Monitoring

Cumulative damage assessment is conducted to ensure that engine components do not fail in service, either without giving adequate warning or before their design life is reached. The initial thrust to improve life usage predictions came as a response to the management and logistic needs of in-service fleets. Simple monitors were fitted to engines and data was recorded and analysed by the manufacturer, often as a product support task, rather than a design activity.

Engines contain a number of highly loaded parts which are only released for limited service usage. Life limitation is based on assessment of the low cycle fatigue, thermal fatigue and/or creep service capability of the component under consideration. Using the measured time histories of engine operating parameters (such as spool speeds, intake and gas path temperatures and pressures), the algorithms calculate the thermal and mechanical boundary conditions for the engine components to determine the transient temperatures and stresses or strains at critical areas. These are then used to predict the related damage due to low cycle fatigue, thermal fatigue and creep which is summed over all engine runs to establish life consumption records for all monitored parts. Engine life consumption is counted in terms of engine flying hours or engine flight cycles depending on the mission profiles, engine intake conditions, individual pilot reactions and several other parameters. High cycle fatigue and corrosion are not currently monitored but could be incorporated if required.

Damage algorithms have been developed that are capable of application in real time life usage monitoring systems to calculate the consumed life directly from measured engine signals. The monitoring system must respond to rapid transitions in input signals during complex aircraft and engine manoeuvring. On-board systems process the results continuously as they are developed, whilst ground based systems process recorded parameters immediately after the end of a flight. The life consumption associated with each particular flight is measured in damage related physical or technical units.

4.1 Calculation Process

The life consumption calculation and monitoring process is controlled by engine running history and can be divided into three phases that correspond with engine start, engine run and engine shut down. In the first phase, system and algorithms are initialised, followed by the second phase during which the main monitoring tasks are active and computations repeated for every time step (e.g., every 0.5 second). In the third phase all the monitoring tasks are finalised and the results stored. The performance calculation is comprised of determination of the temperatures and pressures in the gas path and in the cooling air paths and also includes the computation of torque, bending moments and normal forces in shafts. These form the boundary conditions for the damage assessment steps.

4.2 Metal temperature Calculation

The metal temperature distribution is used to determine the thermal stresses and the temperature influence in damage assessment. Calculation at a given point should take account of the heat transfer to the temperature point from the surrounding gas and cooling air, and vice-versa, and heat conduction between any temperature differences in the component. Heat balancing for each of these points may be described by regular differential equations optimised such that typical deviations from the given basis data are less than 10 K. The equation's coefficients depend on current engine operating parameters (e.g. spool speed) and the time step length. The local transient metal temperature and temperature distributions in the component are calculated during the three phases of an engine run. The initial temperatures, determined in phase 1 for all temperature points, depend on the temperature distribution at the end of the last engine run and the time which the engine has been stopped. During phase 2, the temperatures of each point are updated every time step using the thermal boundary conditions and the temperature distribution at the beginning of the current time step. In the final phase the temperatures corresponding to shut down stress peaks are computed.

4.3 Stress Calculation

Transient stresses are calculated for each critical area by summing up the centrifugal, thermal and pressure induced stresses and any additional stresses (bolt clamping, residual stresses, etc.) for every time step during the main phase. In the initial and final phases stress peaks occurring during engine start and engine shut down are determined. If shafts are monitored, stresses due to torque and bending can also be included. The stress model's coefficients are optimised to account for the fact that under cyclic damage the major cycle contributes most to the accumulated damage and the accuracy is weighted so that maximum and minimum stresses are satisfied, whereas for intermediate stresses wider tolerances are acceptable. Fracture mechanics based damage assessment methods require stress intensity factors or other parameters to be derived from the stresses.

4.4 Damage Assessment

As illustrated in Figures 2 and 3, the stress-temperature histories are assessed with respect to the relevant damage mechanisms. For fatigue damage, the cycles (i.e. major reference cycle and subcycles) may be extracted using Rainflow analysis. The cycles obtained are converted into equivalent damage according to the S-N curves, or for fracture mechanics assessment the stress intensity range is used in conjunction with the relevant crack growth law (da/dN curve). In both cases, subcycle and major cycle damage are accumulated over the whole engine run. In addition, if creep is an important damage mechanism then a creep damage increment is also evaluated in every time step.

4.5 Data Check and Substitution

To ensure accurate operation of the monitoring algorithms, the input data range and rate are checked and corrective actions taken if faults occur. For short drop out periods interpolation of the respective signal can be used, however, for longer periods substitutes derived from other genuine input signals are taken. If these are not available the monitoring process must be interrupted and life consumption estimated from the flight time or engine run time.

4.6 Stress Modelling

The object of stress modelling is to ensure that the component will survive the duties imposed on it. Modelling forms the basis from which the designer works and generally involves finite element analyses of physical and thermal boundary conditions, heat flow calculations and the step-by-step assessment of the stresses induced throughout a large number of flights. All subsequent service experience and design changes are referred to this model.

4.7 Materials Characterisation

The materials properties considered in the analyses go far beyond Elastic Modulus and simple proof stress design limits. The design of critical features involves consideration of how material properties change with temperature, time, stress level, damage histories and strain rates. Due to the high costs associated with engine redesign, materials selection tends to be conservative so that information about the linear (elastic) and non-linear (plastic/creep) performance of the material under various static and cyclic loadings and temperatures are available as input to the stress model.

4.8 Component and Engine Testing

Once a new engine design programme is launched in its own right, a number of component models are evaluated. These may range from simulated components in rigs through to full scale, instrumented flight engines. In each case the objective is to increase the designer's knowledge and confidence in the design. As experience with real components is accumulated and the materials database grows, a feedback loop to earlier materials experience and the stress model assumptions is created. Final feedback from the qualification test programme may, as with ENSIP, seek to prove the structural integrity of the engine hot end parts through full scale engine tests to demonstrate that a full service life can be achieved. These tests are extremely costly and cannot be used to validate statistically the declared service life for fracture critical components. This can only be achieved by statistical assessment of a series of component test results. Hence, basic LCF programmes are used to provide the larger element of proof by analysis with the model tests providing confirmation and feedback.

In parallel with the engine component structural integrity tests the airborne and ground based monitoring and computational system must be qualified. It must be shown to give, within close limits, the same results with the same data as the design models.

4.9 Field Experience

Field experience represents the end point at which the design is proven. By this stage, however, a number of the original design assumptions may require modification. Almost invariably, the way in which the engine is used will differ significantly from that specified. From original conception, a new aircraft or engine programme can take up to eight years to complete. In this time the operational scenario would have probably changed and the recipient will want to use the vehicle in a different way to exploit fully new or unique capabilities.

Components now have a modified service usage model and a growing operational usage database which may differ from and have a much greater volume than that assumed in the design model. The damage accumulation rate collected by the engine monitoring system is not always as expected and the cumulative damage picture across the fleet will identify special cases where very high or very low levels of damage occur. These may lead to changes in pilot practice or to changes in spares provisioning rates. The information being gathered provides a model of component usage based on direct measurements and computations of the airborne system. Without such monitoring, as components continue in service until they are rejected or achieve their declared service lives, there would be would be no feedback about how components are consuming life.

The monitoring system results only work according to the rules built into it. Hidden flaws or errors in the original model will not reveal themselves unless a component fails, or unless a sample of components is removed from service and tested to destruction under controlled conditions, which is expensive. Any significant differences between the design predictions and the interrogation of the engine monitoring system usually result in an examination of the most appropriate things to change. This often relates to the usage rules, algorithms, or materials properties.

5. Lifing Concepts for Fracture Critical Components

5.1 Basic Damage Algorithms

The damage algorithms are based on the following assumptions:

1. Major stresses are due to centrifugal loads and are proportional to the square of the component's rotational speed.
2. Under normal service usage, rotating components are most likely to fail (due to fatigue).
3. Induced stresses consist of both steady and alternating loadings. The allowable alternating stress must be reduced as the steady stress component increases and statistical analysis is necessary due to the wide variation in test results, as indicated in Figure 4.
4. Fatigue damage is cumulative according to Miner's law (life fraction summation for different loading conditions).

More advanced algorithms take account of thermal loads due to temperature gradients, creep, stress relaxation and recovery mechanisms within the component during its duty cycle. To ensure acceptable levels of accuracy and safety the design computations should "mirror" the life usage computations that reflect service practice. Measured data may include air temperature and pressure, altitude, engine rotor speed, compressor delivery pressure and outlet temperature. Stress and strain cycles and damage are established using procedures similar to those used in the design stage. However, these require substantial computing power and can only be used effectively on ground based installations. On-board processing requires simplified analysis routines verified by formal qualification procedures.

5.1.1 Mechanical Stresses

Stress analysis is used to determine the critical areas and peak stresses within the disc at a reference shaft rotational speed. For all other loading conditions, the peak stresses can be related to the previously identified peak stress or reference stress, σ_r by an expression of the form:

$$\sigma_i = \sigma_r \left(\frac{N_i}{N_r} \right)^2 \quad \ldots 1$$

where σ_r is the peak stress under imposed engine reference shaft speed conditions Nr (revs/sec), and σ_i is the stress corresponding to the speed N_i (revs/sec). This expression enables the stress-time profiles at critical locations to be determined throughout any mission profile. Fatigue damage, however, is related to the cyclic stress range and the significant stress cycles from the stress histories of the sortie profiles are extracted via a Rainflow procedure [4] as shown in Figures 2 and 3. The method takes no account of cycling rate or of dwell effects and assumes that interruption of large stress cycles to complete smaller cycles does not affect the damage imposed.

The effects of mean stress level are assessed using a Goodman diagram by estimating the zero-to-max stress range which would impose an equivalent amount of fatigue damage as that for the high mean stress range test. Mean stress level is plotted against stress range and a line drawn between the tensile strength of the material and the stress amplitude of the push-pull test having "N_f" cycles to failure. It is assumed that all combinations of mean stress and stress range that lie on the line have a common life to failure, and hence, high R-ratio stress ranges can be expressed in terms of their Goodman equivalent zero-to-max stress, $\Delta\sigma_E$, via the expression:

$$\Delta \sigma_E = \left[\frac{\sigma_{max} - \sigma_{min}}{1 - \dfrac{\sigma_{min}}{\sigma_{UTS}}} \right] \quad \ldots.2$$

Equivalent expressions can be constructed for strain based fatigue results, and an alternative formulation which accounts for maximum stress and mean strain is the Smith Watson Topper Parameter [5]:

$$\sigma_{SWT} = \sqrt{\sigma_{max} \Delta\varepsilon E} \quad \ldots.3$$

Having identified the equivalent stress range, Miner's law can be used to convert the damage imposed to its equivalent amount of major cycle damage. The Miner's expression has the form:

$$\frac{\Delta \sigma_R}{\Delta \sigma_E} = \left(\frac{n_{RF}}{n_{EF}} \right)^m \quad \ldots.4$$

where n_{RF} is the number of reference cycles to failure at the reference stress, $\Delta\sigma_R$, and n_{EF} is the number of cycles to failure under the equivalent zero to max. minor cycle, $\Delta\sigma_E$. The slope of the fatigue design curve is given by "m".

For single minor cycles, damage can be expressed in terms of equivalent reference cycles via the expression:

$$D_R = \left(\frac{\Delta \sigma_E}{\Delta \sigma_R} \right)^{\frac{-1}{m}} \quad \ldots.5$$

Hence for each mission the exchange rate, β, can be expressed as:

$$\beta = \sum D_R = \sum_1^{n_c} \left(\frac{\Delta \sigma_E}{\Delta \sigma_R} \right)^{\frac{-1}{m}} \quad \ldots.6$$

where n_c is the number of cycles extracted via the Rainflow analysis and, D_R equals the equivalent number of reference cycles that have been consumed.

When using the basic LCF algorithm a "cut-off" stress range is commonly used below which the LCF life is assumed to be infinite. A stress level at or close to the endurance limit is chosen and leads to an expression of the form:

$$D = \frac{1}{n_{RF}} \sum_1^{n_s} \sum_1^{n_c} \left[\frac{\Delta \sigma_R - \Delta \sigma_c}{\Delta \sigma_E - \Delta \sigma_c} \right]^{\frac{1}{b}} \quad \ldots.7$$

Where $\Delta\sigma_R$ is the reference stress range, $\Delta\sigma_c$ the asymptotic stress range, $\Delta\sigma_E$ the total equivalent zero - to-max stress range and n_{RF} is the number of reference cycles to failure.

5.1.2 Thermal Algorithms For Low Cycle Fatigue In Discs

During normal engine running, temperature profiles are determined from the temperature transfer matrix, which describes heat conduction and radiation between hot gas, cooling air and component surfaces derived from previous mission distributions, calculated for the current operating conditions. Fixed time steps are generally used, which can result in large differences in temperature change per second between transient and near steady state conditions. Changes per time step can range between about 10°C after slam acceleration, to 0.001°C in the asymptotic phase. When the time between successive missions has not allowed the components to reach ambient temperature, the initial temperature distribution is interpolated from the previous shutdown at the appropriate elapsed time.

Transient temperatures can induce significant thermal stresses, predominantly during take-off and landing and a full analysis involves predicting component transient thermal and mechanical stresses from flight recorded data. For the centre regions of a large turbine disc, the characteristic response time can approach 20 minutes, during which time some 20 major throttle movements may have occurred. Therefore, most transient events commence from a non-stabilised state which are interrupted by further changes before stability occurs. Coupled transient temperature/transient stress finite element analyses require significant computing power due to the short time steps needed to identify both the instantaneous transient thermal stresses and the turning values associated with changes in throttle setting. This involves computation of gas stream temperatures, pressures and velocities from engine characteristics, determination of appropriate gas-metal heat transfer coefficients and transient temperatures throughout the component and throughout the flight, and calculation of the thermal stress distribution profiles from these calculated temperatures. Simplified thermomechanical stress analysis can be used which is then matched with full thermo-mechanical finite element computations [6]. The critical parameters can be established by curve fitting to the thermo-mechanical analysis results. The temperatures are assumed to decay exponentially towards their asymptotic value (if current conditions were held indefinitely) and the characteristic constant of the decay depends on the engine conditions. This can be modelled via a small number of thermal masses (typically 3), each having 4 or 5 dependable constants to control the temperature at a particular location, and hence can be used in calculating transient temperatures and their associated thermal stresses.

Given the spool speed (N), the local gas temperature (T) and pressure (P) for each individual mass the asymptotic temperatures T_{ai} can be determined:

$$T_{ai}(t) = K_i^1 T + K_i^2 \quad \ldots .8$$

The heat transfer coefficients $H_i(t)$ can be obtained from:

$$H_i(t) = K_i^3 \left(N \, {P \over T^{0.7}} \right)^{K_i^4} \quad \ldots .9$$

and the temperatures of the masses $T_i(t)$ from the numerical solution of the differential equation:

$$\frac{dT_i}{dt} = H_i (T_{ai} - T_i) \ldots .10$$

With an appropriate initial condition for the start of the flight, these equations lead to a thermal stress equation of the form:

$$\sigma_{(t)} = \sum_i K_i^5 T_i(t) \ldots .11$$

where 'i' is summed over the number of thermal masses that significantly influence the selected location.

In addition to calculating the thermal stresses, the corresponding temperatures are retained as material property data can be highly temperature dependent. The model should be validated against the stress and temperature results from a full analytical solution, including features not in the original conditions, transients to and from different engine states and non-stabilised conditions.

6.1 Experimental Support for Basic Damage Algorithms

Figure 4 presents over 70 notch fatigue test results for the titanium alloy Ti 6/4. The number of tests, the spread in results and the mean fatigue lives for each loading condition are shown, which can be used to estimate the scatter in fatigue for the total population. Figure 5 shows the high R-ratio results in terms of their Goodman equivalent stresses. These fall within the scatter bands obtained for the initial zero-to-max tests and supports the Goodman approach. Figure 6 shows the Goodman diagram constructed from Figures 4 and 5. For ease of interpretation only the mean data points have been transferred to this diagram. The lines of constant life drawn through the high R ratio data intersect the zero-to-max 45° line in approximately the correct positions. However, the true tensile strength of the material has been chosen as the focal point on the mean stress axis. The specific value has little effect on where the constant life lines intercept the zero-to-max data, but selection of true tensile stress makes the handling of high R-ratio data easier.

To provide experimental support for the LCF summation algorithms, a series of simplified military aircraft mission simulation tests was applied to a multi-specimen rig, modified to enable a repeatable ten-cycle sequence to be performed. Using the combined Goodman/Miner summation approach adopted in the fatigue algorithms, the results can be expressed in terms of "equivalent major cycle damage", as shown in Figure 7. These results all fall within the zero-to-max scatter bands and supports the theoretical concepts employed.

6.2 Analytical Support for Simplified Thermal Algorithms

To illustrate the adequacy of the model, analysis of the transient temperatures and thermal stresses occurring within the HP turbine of a military engine during a typical combat mission has been used to construct simplified models. These have been applied to representative missions to check the predictions against the detailed analysis for the mission. Comparison of the transient metal temperature results for a military engine turbine disc is shown in Figure 8. Differences between thermal stresses calculated using the simple algorithm and base-line data obtained via complex programs running on main frame computers are also shown with excellent agreement.

7. Crack Propagation Lifing Algorithms for Discs

An approach increasingly favoured for new disc designs is the "Life to 2/3 Dysfunction" which depends heavily on developing crack growth behaviour algorithms. Application of damage tolerance concepts contrast with crack initiation lifing concepts as the crack size now represents a measure of life consumption. Prediction of possible crack size considers the "worst case" conditions, a practical complication being that predicted crack sizes significantly exceed those of any identified cracks. Crack size itself is of no practical use to the customer and so, based on the correlation between accumulated cycles and crack size, the monitoring system can easily convert the "crack size" damage into reference cycle damage and enable only a single unit to be used as the measure of life consumption. i.e. internally, both LCF and damage tolerance lifing concepts can be combined within the monitoring system. Each factor affecting crack growth must be identified and modelled statistically to ensure that the approach can be implemented without compromising safety. The procedures described below are equally applicable to life extensions beyond "first engineering crack" and to the ENSIP damage tolerance concept, where safe crack growth lives are based on the maximum defect size associated with NDI inspection and a detection probability of 90% to a confidence level of 95%.

Crack initiation and propagation in laboratory specimens under constant load amplitude are reasonably well understood and are generally quantified via linear elastic fracture mechanics. Variable amplitude loading, however, as occurs in service requires identification of a scaling parameter. A simple approach is to assume that the full elastic load range normal to the crack contributes to the effective stress intensity cycle providing the driving force for crack growth. This can lead to inaccurate estimates of the available safe service life because the higher stresses imposed under laboratory test conditions can lead to crack closure on unloading and compressive residual stresses that subsequently retard crack growth rates during the minor mission cycles that immediately following a major cycle. The position and magnitude of the maximum stress, in the missions flown, is dominant in identifying the boundary of the plastic zone and the associated compressive residual stress field when this cycle is completed. For sufficiently high loading conditions that cause yielding in the region of a stress concentration feature in an uncracked component any subsequent crack growth will only occur when the crack is fully open. The crack opening stress level depends on the magnitude of the local compressive residual stress field created by a preceding overload. The effective stress intensity range driving the crack growth process is simply the peak elastic value associated with the maximum stress less the crack opening stress. Where local plastic yielding can occur, due account must be taken of the relative effects of the different plastic zone sizes created under spin pit component life determination and under service conditions and the lifing methodology for crack growth must therefore be more complex than that for the crack initiation situation.

In any fracture mechanics based crack growth methodology, it is essential to identify an appropriate statistical model to ensure that current safety levels are not compromised.

8. Identification of Criteria for Total Safe Cyclic Life

Evaluation of a large number of disc spinning test results has shown that the dysfunction life is frequently about 1.5 times the LTFC. However, when the crack is growing into a rapidly decreasing stress field, the crack propagation phase can account for a considerably larger portion of the overall life. To provide suitable criteria for these circumstances service life can still be defined as 2/3 of the overall failure life, irrespective of the initiation/propagation ratio. This preserves the existing safety levels in those cases which do not conform to the standard 2 to 1 ratio, but allows life extensions for situations where the ratio is greater. The only obvious alternative is to consider the whole of the initiation plus some fraction of the propagation life. For high strength disc materials the US military standard specification, Mil Spec 1783, adopts an approach based on declaring half of the crack growth life, ½Np. However, since the latter is based on the use of mean crack growth rate data, the "2/3 burst" criteria is generally more conservative. Consequently the "1/750 quartile and 2/3 burst" criterion is adopted in most European methodologies. (A quirk of the statistical models is that where the $+3\sigma$ dysfunction life is a factor of 4 times the -3σ life, the 1 in 750 "2/3 burst" declared life will equal the ½Np declared life and hence calculates the same value as from the ENSIP approach, however the additional application of a 95% confidence factor results the "2/3" criterion yielding a more conservative prediction.).

9. Conclusions

1. The optimisation of service lives for fracture critical components is dependent on the adoption of fleet wide service usage monitoring systems and represents a key factor in the reduction of life cycle costs.
2. The effectiveness of component usage monitoring depends on the accuracy of measured engine parameters and on the capabilities of the numerical algorithms and computing equipment.
3. Simplified life usage prediction algorithms have been developed to provide real time monitoring of fracture critical components. These are now well established.
4. Verification and application of the simplified algorithms has been achieved through engine service usage, however, further experimental validation of transient predictions and non-linear loading mission sequences is necessary.

10. References

1. G.F. Harrison, M.E.F.Smith and J.Nurse "Procedures for Aeroengine Component Life Usage Prediction." Proc. 5th European Propulsion Forum, Paris, April 1995.

2. G.F. Harrison and D P Shepherd "Lifing Philosophies for Aero Engine Critical Parts." Aero Engine Reliability Integrity and Safety. Proceedings of the Royal Aeronautical Society, 1991.

3. US Mil Std 1793 (USAF) ENSIP, 1984.

4. P.Watson and B.J. Dabell "Cycle Counting and Fatigue Damage." Jnl. of Soc. of Environmental Engineers, Sept.1976.

5. K.N.Smith, P.Watson and T.H.Topper "A stress-strain function for the fatigue of metals." J.of Materials, **5**, No.4, 1970, pp. 767-778.

6. T.M.Edmunds and R.A.Lawrence "Monitoring Engine Thermal Stresses." AGARD CP386 Engine Cycle Durability by Analysis and Testing, Netherlands, 1984.

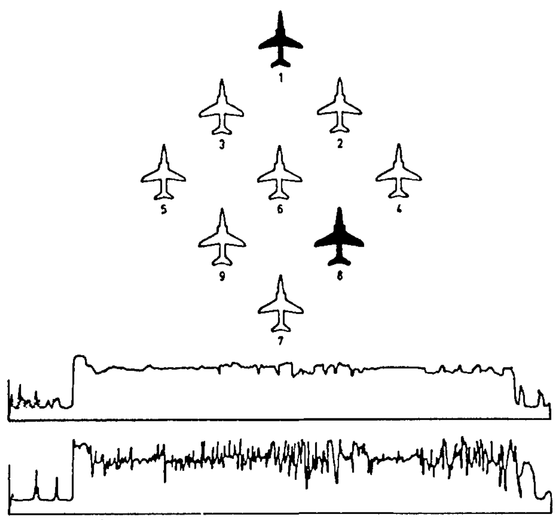


Figure 1. Effects of throttle movements during typical formation flying by the "Red Arrows" display team.

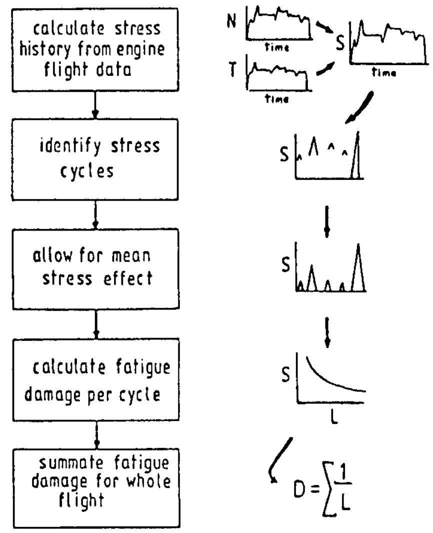


Figure 2. Procedure for calculating low cycle fatigue usage.

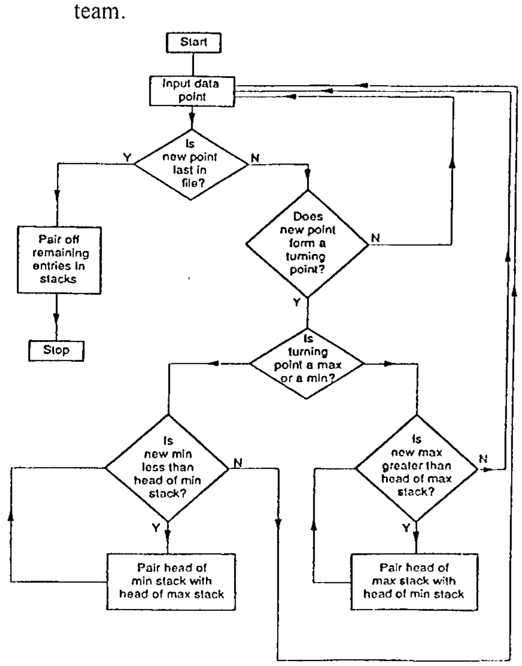


Figure 3. Basic structure of Rainflow sub-routine.

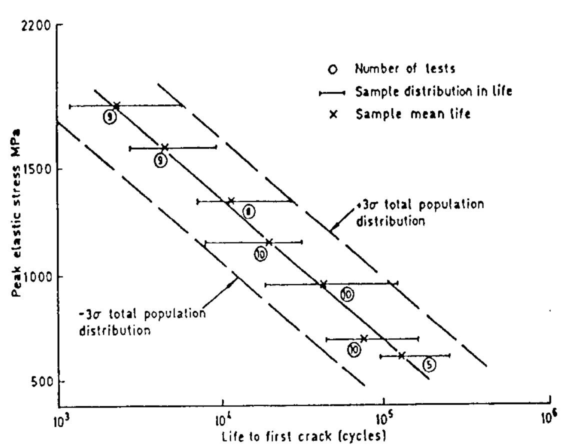


Figure 4. Notch fatigue data for Ti 6/4.

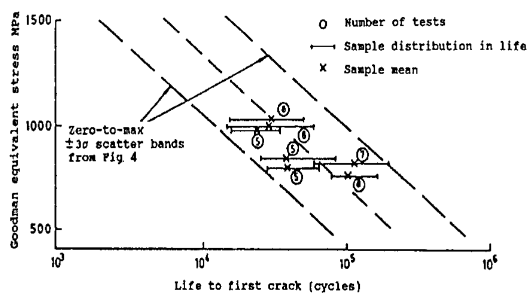


Figure 5. High mean stress notch fatigue data for Ti 6/4 plotted in terms of Goodman equivalent zero-to-max stress amplitude.

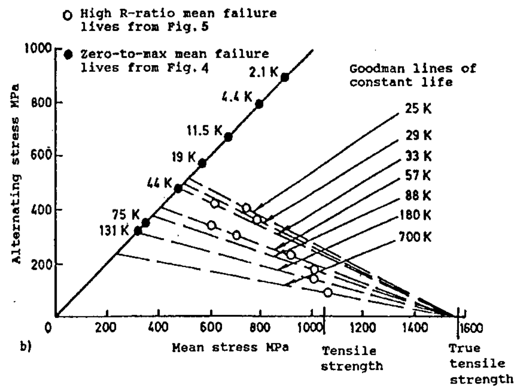


Figure 6. Modified Goodman diagram for Ti 6/4 notch fatigue data.

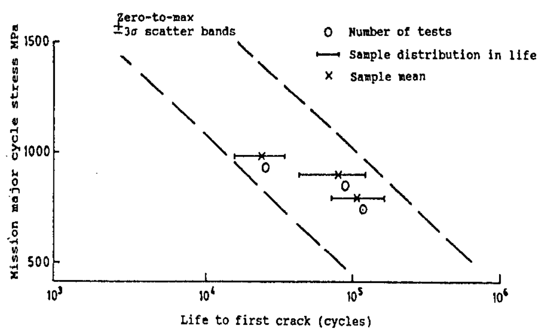


Figure 7. Simplified mission profile notch fatigue for Ti 6/4 data plotted in terms of equivalent reference stresses and equivalent reference cycles.

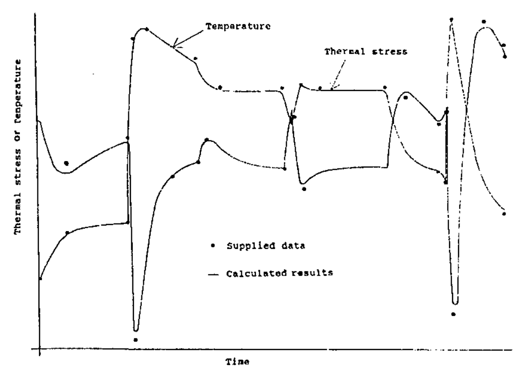


Figure 8. Correlation of the calculated and supplied transient temperatures and thermal stresses for a military engine turbine disc during typical mission cycle.

HUMS LOADS MONITORING AND DAMAGE TOLERANCE: AN OPERATIONAL EVALUATION

Michael L. Basehore
Federal Aviation Administration
NASA Langley Research Center
Hampton, VA 23681
USA

William Dickson
Bell Helicopter Textron, Inc.
P.O. Box 482, Dept. 8
Fort Worth, TX 76101
USA

SUMMARY

This paper describes the results of a research program to evaluate structural usage monitoring and damage tolerance methodology using data collected concurrently during a helicopter flight program. The helicopter, a Bell Model 412 equipped with a Health and Usage Monitoring System (HUMS) and data recorder, was operated by Petroleum Helicopters Inc. (PHI) during the 1996 Summer Olympic Games in Atlanta, Georgia, as a part of Project HeliSTAR. This effort was conducted by Bell Helicopter Textron Inc. (BHTI) under the cognizance of the Federal Aviation Administration (FAA), the U.S. Army, and NASA. The helicopter was flown in what is referred to in this paper as the Atlanta Short Haul Mission (ASHM). This mission involved numerous short flights to pick up and deliver packages and freight. Data recorded during the period, together with pilot flight records and maintenance records were furnished by PHI to BHTI for analysis. The results of the analysis of the ASHM were compared to results from an offshore oil support Gulf Coast Mission (GCM) which involved longer level flights at cruise airspeed.

The purpose of the program was to acquire usage data for the ASHM and to perform component fatigue life calculations and damage tolerance evaluations for selected critical dynamic components, referred to here as Principal Structural Elements (PSE's). Although the usage was more severe for the ASHM than the GCM, the results of the comparison showed that usage monitoring would provide benefits in extending retirement times or inspection intervals, compared to certification, especially if high/low altitude effects were considered. In addition to usage monitoring evaluations, guidelines for HUMS certification are discussed along with simplified "mini-HUMS" approaches to provide low cost systems with high return on investment. The lives and inspection intervals determined for purposes of this study should not be used to draw any conclusions concerning certification or continued airworthiness of the Model 412 helicopter.

1. MISSION DESCRIPTIONS

HUMS data recorded during project HeliSTAR covered the period from July 19, 1996 through August 1, 1996 and contained a total of nine flying days. It should be noted that the data sample for the ASHM is limited (approximately 17 hours of flight data) compared to the approximately 450 hours of flight data processed from the GCM. Flight data were not recorded during the afternoon on two of the mission days, resulting in the loss of approximately 10 hours of flight data. An investigation indicated that the recorder was not operating during the missing 10 hours but did not reveal a reason for the data loss. The Quick Access data Recorder (QAR) used for the ASHM was separate from the HUMS and not representative of an integrated data recorder as would be used in a production system. Statistical methods need to be developed to account for unrecorded or corrupted data. Because of the limited amount of data, care should be exercised regarding the mission characteristics presented and any analysis resulting from the use of the ASHM data.

The ASHM consisted mainly of flights of short duration with a large number of maneuvers. The ASHM consists of a significantly higher percentage time in low to moderate speeds (0.8 and $0.9V_h$) and in turning than either the GCM or certification spectra. The Gulf Coast mission consisted primarily of high-speed level flight. Both the ASHM and GCM indicate more time spent at 324 rpm than at 314 rpm while the certification spectrum assumes more time at 314 rpm. the time at condition comparison is emphasized in Figure 1.1, which represents the data sorted by descending time at condition for the ASHM.

2. SELECTED COMPONENTS

This section discusses the four PSE's that were selected for analysis. The PSE's selected comprise the following components:

- Rephase lever
- Collective Lever

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

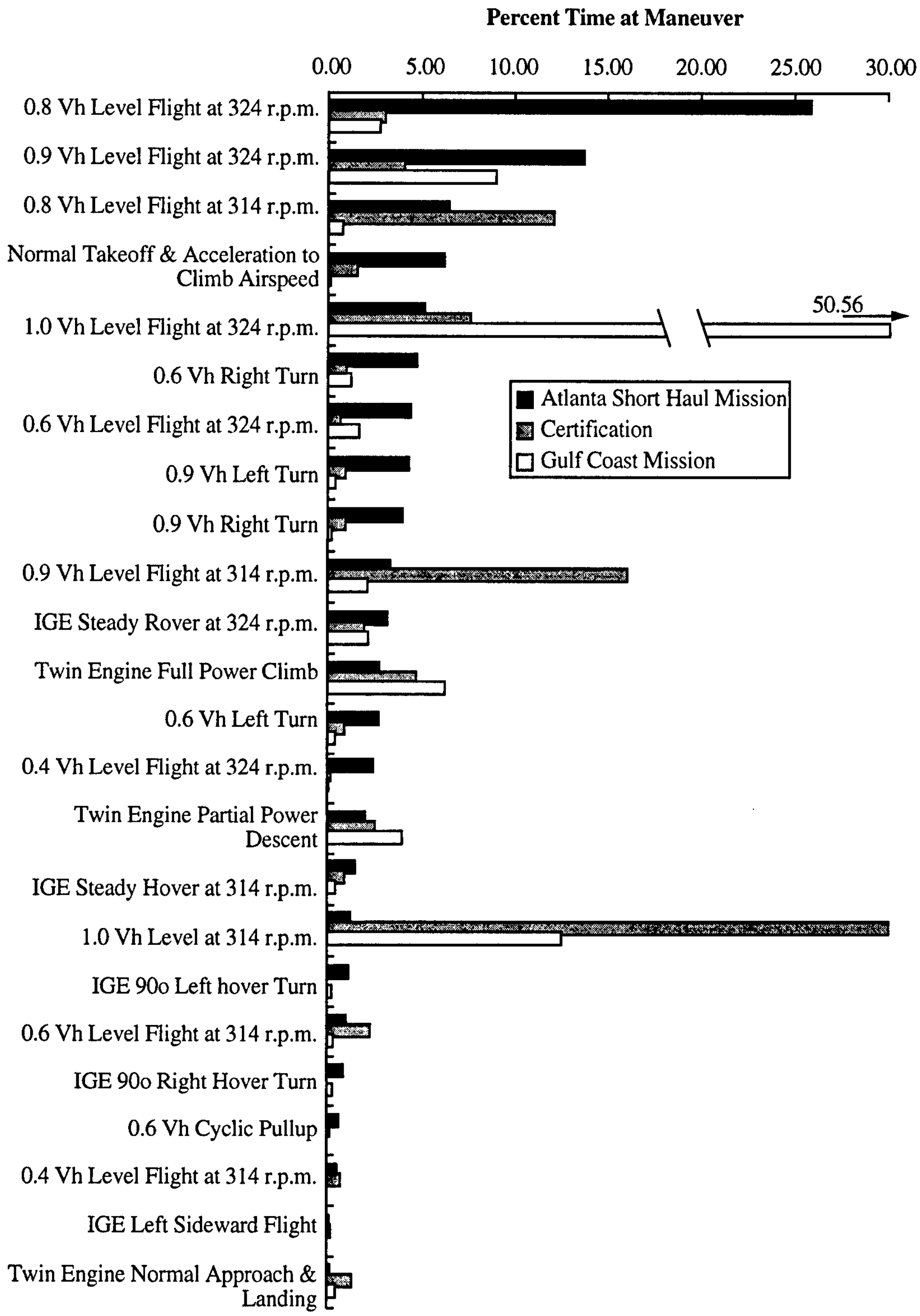


Figure 1.1 Spectra Comparison

- Main Rotor Spindle
- Main Rotor Yoke

The part service history of the PSE's is presented in support of the assumptions made for the initial flaw sizes used for the damage tolerance analysis and includes the service history, e.g., failures, redesigns, configuration changes, process changes, Advisory Service Bulletins (ASB's), Airworthiness Directives (AD's), reports and other design and manufacturing actions.

As part of this study, the documented service history of the four PSE's was reviewed for premature component removal. The source of the data for this study was either the customer Discrepancy and Malfunction Report (DMR) or documentation of service returned components using BHTI Field Investigation Reports for all design derivatives. In the case of DMR's, BHTI maintains a computer database that summarizes the information from the written document. A total of 877 DMR's were reviewed by this method beginning with the introduction of the Model 412 helicopter in 1981. A request was forwarded to the Field Investigation Laboratory to provide reports on any of the four study components that had been evaluated during the same period.

Table 2.1 is a summary of the findings of this inquiry. The reasons for component removal are divided into broad categories for the purposes of this study. Generally, an attempt has been made to separate and note categories involving physical discrepancies/damage to the component whether manufacturing induced or service induced. The total number of DMR's reviewed may not represent all components that were removed prematurely, although all component and component design derivatives are included in this study. Generally, a DMR is written by the customer as a means of obtaining warranty credit towards a replacement part. In the case of the yoke, a large number of components were removed in response to a manufacture's bulletin or an FAA Airworthiness Directive or both. In the case of the spindle, a large number of the parts in the "other" category were removed due to premature deterioration of the elastomeric feathering bearing or replacement with an improved part.

The rephase lever is manufactured from a 7075-T73 aluminum forging, Figure 2.1. The rephase lever pivots on a rotating hub and provides a reindexing of pitch link to the swashplate by offsetting the attach points. Swashplate motion is imparted to the rephase lever via a tubular link or a drive link. This motion is then transferred to the rotor by the pitch link with the rephase lever as the intermediate mechanism. The majority of the DMR's for the rephase lever resulted from bulletins, which provided an improved version of the design.

The collective lever is manufactured from a 7075-T73 aluminum forging, Figure 2.2. The collective boost actuator attaches at the apex of the lever. The lever pivots about an axis common to a lug situated on the swashplate support. The ends of the legs attach to the collective sleeve to impart mean blade angle changes. The majority of the DMR's for the collective lever involved joint wear as the cause of replacement. Parts returned to manufacturer that would not install correctly due to accumulation of adverse tolerances are included in the Table 2.1. No corrosion reports were received.

The original spindle design (Figure 2.3) was manufactured from SAE 4340 alloy steel and was protected from corrosion by an applied surface finish. The elastomeric feathering bearing was mechanically attached to the spindle by means of a bonded inner race. The pitch horn is splined to the end of the spindle. The spindle exhibited corrosion in the pitch horn attachment area as a result of the corrosion protection wearing away. Four of the 51

Table 2.1 Part Service History

Topic		Rephase Lever	Collective Lever	M/R Spindle	M/R Yoke
Removal Hours		0	0-6424	0-3159	0-4980
Manufacturing Problem		4	5	1	6
Metal Fatigue					2
Bulletin or AD		11			204
Mechanical Damage	Scratches				1
	Wear		7	1	1
	Corrosion			51	
Other		0		582	0
Total Parts		15		635	214
CR & O Limit-Inches	Scratches	0.005	0.010	0.005 to 0.010	0.005 to 0.010
	Wear	.002	–	–	– 0.002
	Corrosion	0.0025	0.005	0.005 to 0.010	0.005 to 0.010

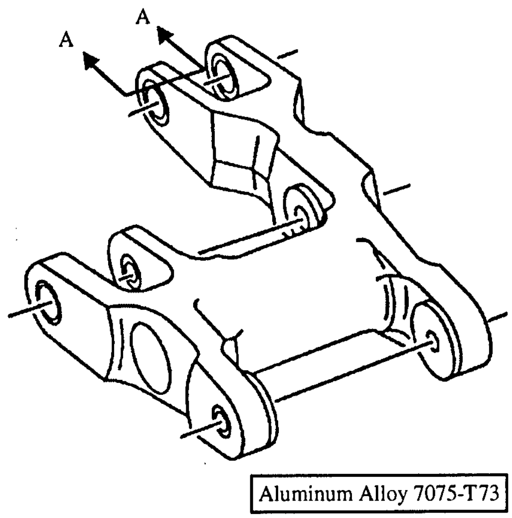


Figure 2.1 Rephase Lever Geometry

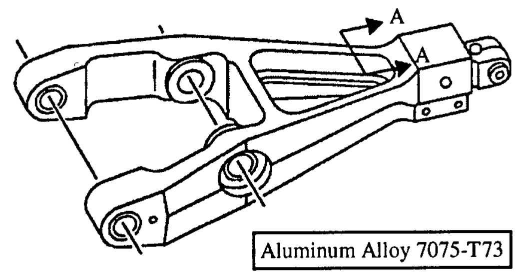


Figure 2.2 Collective Lever Geometry

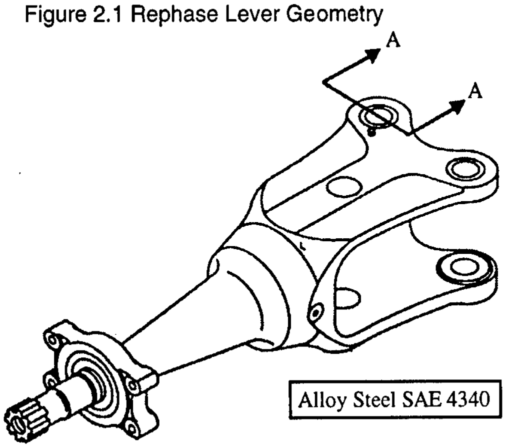


Figure 2.3 Main Rotor Spindle Geometry

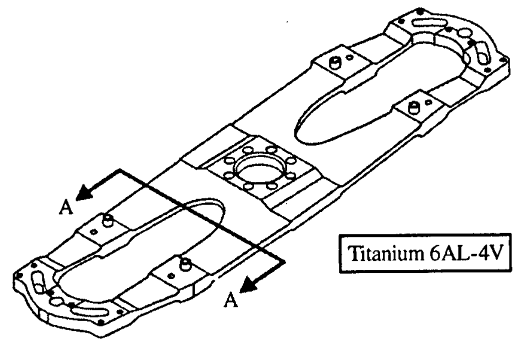


Figure 2.4 Main Rotor Yoke Geometry

DMR's reported corrosion on the order of 0.1 mm (0.0039 inch) in the pitch horn attachment area of the spindle where no corrosion was allowed per the Component Repair and Overhaul Manual (CR&O). Mechanical or corrosion damage of 0.005 inch is allowed around the blade attachment lugs while 0.010 inch mechanical or corrosion damage is allowed elsewhere. The majority of the 582 DMR's in the "other" category resulted from a gradual deterioration of the elastomeric feathering bearing that was detected either visually or as a change in rotor vibration characteristics. Later designs of the spindle were made from 15-5PH stainless steel to eliminate the corrosion problem. The elastomeric feathering bearing is molded directly to the spindle surface allowing the elastomeric element to be increased in size to reduce strains.

In the case of the main rotor yoke (Figure 2.4), the original design was initially certificated with a 5000 hour life. In two separate incidents, the yoke sustained a partial flexure fatigue crack (non-catastrophic) after ground static compressive overloads due to high surface winds. The high loads compressively yielded the shotpeened surface of the 6AL-4V annealed titanium flexure, nullifying the benefits of the peening. A 700 hour service life was established for these early yokes by manufacturer's bulletin and FAA AD. The yoke was redesigned to solve this problem. The yoke flexure was lengthened, the material changed to 6AL-4V BSTOA and a dynamically activated droop stop incorporated to protect the yoke flexure against high beamwise loads due to natural winds or winds generated by other helicopters operating nearby when the rotor was not operating.

In summary, this study of the 877 DMR exhibits of the four subject components revealed several interesting facts. In the 15 years since the Model 412 was fielded, not one accident has been caused by fatigue. The maintenance surveillance currently in place can detect potential problems such as wear, corrosion, etc., before they become serious. The damage limits published in the CR&O manual are realistic with respect to damage tolerance or crack growth thresholds. This data supports the 0.005 inch flaw size used in the crack growth study presented in this report, particularly as it applies to corrosion damage.

3. FATIGUE LIFE ANALYSIS

3.1 Analysis Procedure

The fatigue analysis procedure of the ASHM data was performed on a basis that is consistent with the certification of the selected PSE's. The methodology remains unchanged from that used in the certification process. The only variation in assumptions from the certification procedure is the use of measured time-at-condition in place of the estimated time-at-condition. In addition to the certification procedure, component lives were calculated that include altitude effects.

- Time-at-condition is determined from analysis of the measured flight parameters using flight condition recognition (FCR) software (see Reference 1 for FCR description).
- The loads for each condition are taken from the FAA certification load survey. No additional loads are used in the HUMS data processing.
- Component damage is calculated by combining the loads with the time-at-condition using FAA certification endurance limits.

The certification methodology uses an assumed worst case spectrum of time-at-condition to determine the life of helicopter components. When the FCR software processes recorded data, there is a small percentage of flight time that is not within the parameter set associated with any of the defined conditions. This time is considered to be unrecognized and is assigned the most damaging condition within the domain in which the event occurred.

3.2 Life Limitations

As shown in Figure 3.1, a potential benefit from usage monitoring is part retirement extension if the actual usage severity is milder than the basis for certification. However, recommended retirement lives derived for HUMS-equipped aircraft may be subject to limiting factors other than fatigue calculations. For example, maximum lives or minimum usage rates may be restricted due to reasons of practicality, including, but not limited to, corrosion, wear and component sensitivity to load variation.

3.3 Analysis Results

Analysis results comparing fatigue safe lives for ASHM, GCM and certification data are summarized in Figure 3.2 through Figure 3.5. The component safe lives were calculated without regard to altitude for direct comparison to the certification data. Certification does not employ an altitude breakdown because the operating altitude is unknown. Components are certificated using the most severe altitude within any condition. However, in this study, pressure altitude (H_p) and Outside Air Temperature (OAT) are recorded by the HUMS system allowing for the calculation of Density Altitude (H_d), which is required to take credit for altitude. Load level survey data, used as the basis for

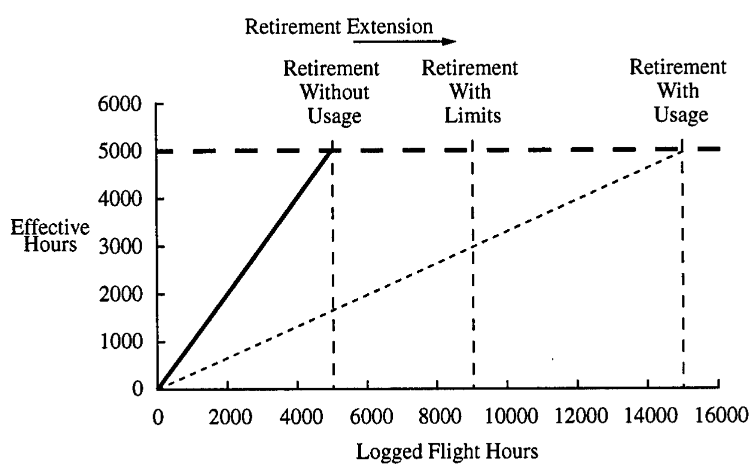


Figure 3.1 HUMS Usage

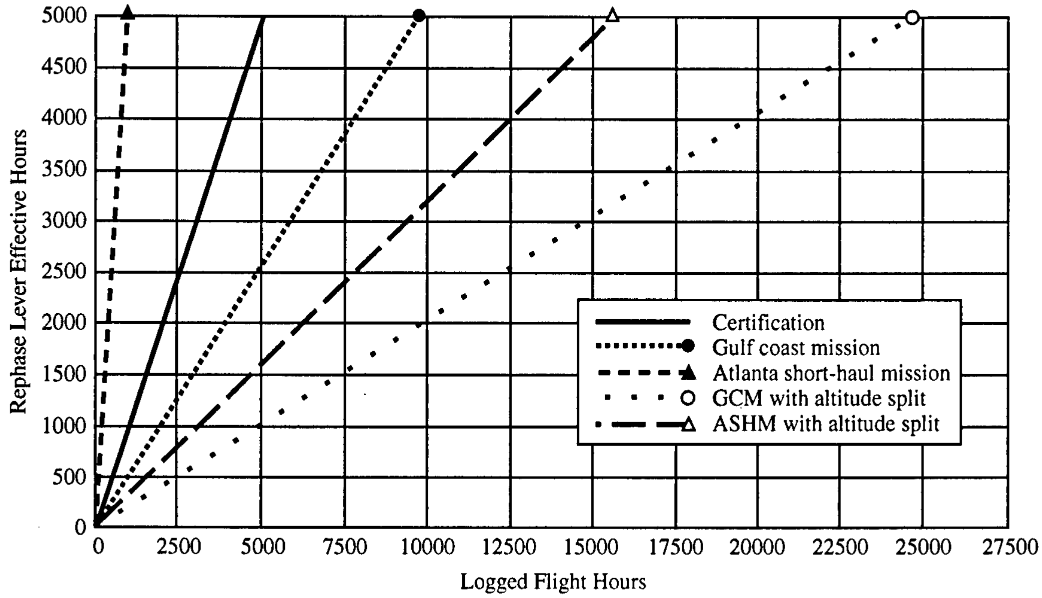


Figure 3.2 Effective Usage Rephase Lever

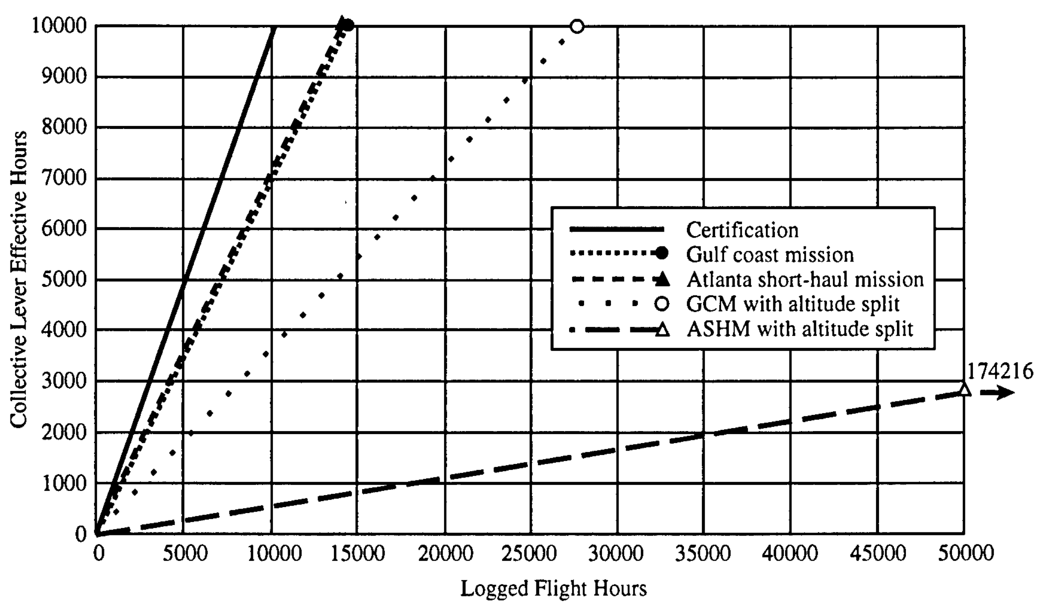


Figure 3.3 Effective Usage Collective Lever

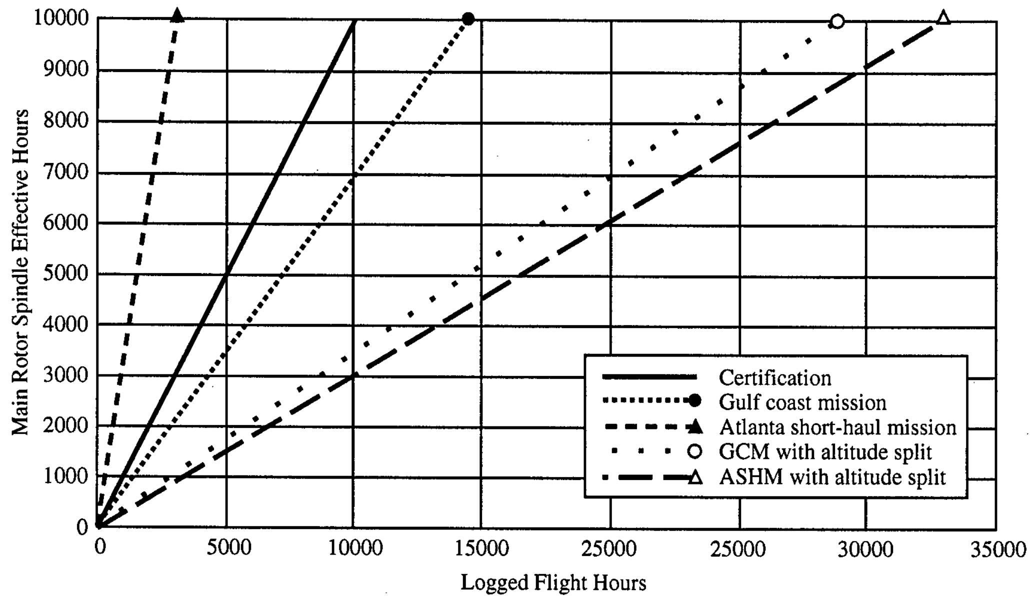


Figure 3.4 Effective Usage Main Rotor Spindle

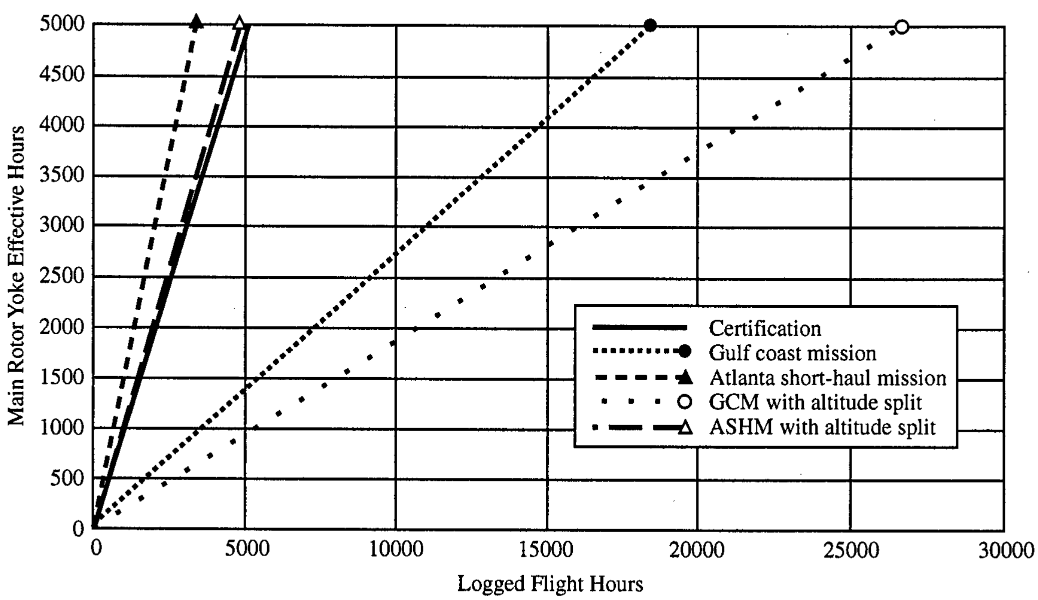


Figure 3.5 Effective Usage Main Rotor Yoke

all life calculations, does not contain all data at all altitudes. For each condition, the survey contains records at 3000 ft and records at 6000 ft and/or 12000 ft for each of the Gross Weight, CG combinations flown. Therefore safe lives were also calculated using a split between high (>3000 ft H_d) and low ((3000 ft H_d) altitude data to ensure multiple records from which to select the most severe condition. This approach deviates from results previously published for the GCM data (Reference 1) which employed a full altitude breakdown. Calculations performed without an altitude split compare directly with certification data. Comparison of spectra with and without an altitude split indicate additional potential benefits due to HUMS.

The results of the comparison of the ASHM and GCM fatigue lives to the certification mission are as follows:

- Rephase Lever - With no altitude split, GCM calculated lives are higher and ASHM lower than the certification, but with altitude split, both are much higher.
- Collective Lever - With no altitude split, both GCM and ASHM lives were about 40% greater than certification and much higher with altitude split.
- Main Rotor Spindle - With no altitude split, GCM is higher, and ASHM is lower, than certification and both are higher with altitude split.
- Main Rotor Yoke - With no altitude split, the GCM is higher, and the ASHM lower, than certification. With altitude split, the GCM is higher and the ASHM about the same as certification.

4. DAMAGE TOLERANCE ANALYSIS

The critical locations and critical flaw sizes were established for each of the PSE's, as well as the maximum probable initial flaw size. The service history of the PSE's is provided in Section 2 of this paper.

This is only a preliminary analysis to determine relative crack growth rate for three different spectra. The analysis was performed for the time-at-condition spectra from the Certification Spectrum and spectra generated from the HUMS data collected during the Gulf Coast Mission and Atlanta Short Haul Mission. Analysis was generated for initial flaw sizes (I_0) of 0.005 inch representing a manufacturing durability limit and 0.015 inch to represent an in-service detectable flaw.

The individual part fatigue test reports were used to determine the critical locations for the crack growth analysis. Analysis was performed at the failure location as indicated by test results. The certification load/stress spectrum and crack growth based analysis methods, CRKGRO (Reference 2) were used to intervals.calculate the inspection threshold and the subsequent inspection intervals.

4.1 Rephase Lever

Figure 4.1 presents the Rephase Lever section; the geometry was described in Figure 2.1. Crack growth analysis was performed for the Rephasing Lever at Lug 2, section A-A. Loads normal to the lug were not considered in this analysis, therefore a damage tolerance life only applies to the loads in the plane of the lug. Mean and oscillatory Pitch Link loads were used to generate the loading spectra for the crack growth analysis.

4.2 Collective Lever

Detail of the analyzed section is presented in Figure 4.2. The Collective Lever part geometry is presented in Figure 2.2. Crack growth analysis was performed at section A-A of Figure 2.2. The Collective Boost Tube mean and oscillatory load spectrum was used to derive the crack growth spectra.

4.3 Main Rotor Spindle

Main Rotor Spindle section, geometry and part detail are presented in Figure 4.3. Crack growth analysis was performed for the Main Rotor Spindle at the blade attachment lug (Sta 32.0) section A-A of Figure 2.3. Blade beam and chord mean and oscillatory bending moments were the reference loads used to generate the crack growth spectra.

4.4 Main Rotor Yoke

The analyzed section is presented in Figure 4.4 and the Main Rotor Yoke geometry is presented in Figure 2.4. Crack growth analysis was performed at blade station 4.8, section A-A.

4.5 Crack Growth Analysis Results

Results for the analyses for the four components are given in Table 4.1 for an initial crack length of 0.005 inch and in Table 4.2 for an initial crack length of 0.015 inch.

5. MEASURED LOAD COMPARISON

A very limited set of oscillatory loads data were measured during the ASHM. These data comprise the Collective Boost Tube, Left Cyclic Boost Tube, and Right Cyclic Boost Tube. These data were collected to provide reference data to indicate the level of conservatism that is built into the analysis. These data were analyzed to determine the frequency of occurrence at various loads. Recorded data were extracted from the load level survey database and processed with the time at condition measured for the three available missions. The curves represent the number of times per hour a given oscillatory load will exceed a given level, e.g. 47 cycles/hour exceeded 200 lb for the collective boost tube (Figure 5.1). This comparison indicates that the measured cumulative load data are approximately two orders of magnitude lower than that predicted by the flight load survey data in the region at and above the endurance limit.

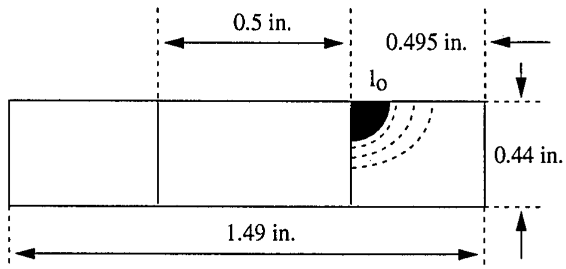


Figure 4.1 Rephase Lever Section A-A

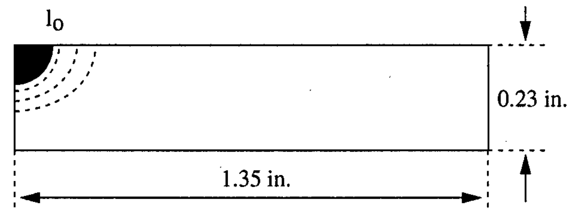


Figure 4.2 Collective Lever Section A-A

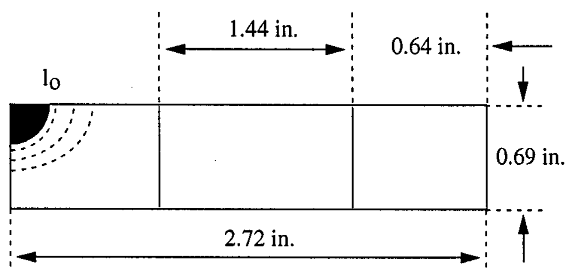


Figure 4.3 Main Rotor Spindle Section A-A

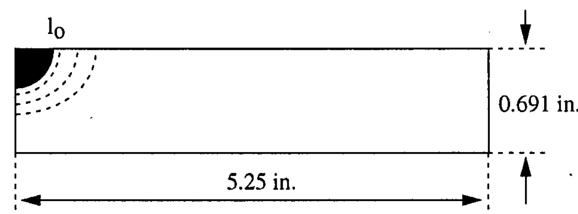


Figure 4.4 Main Rotor Yoke Section A-A

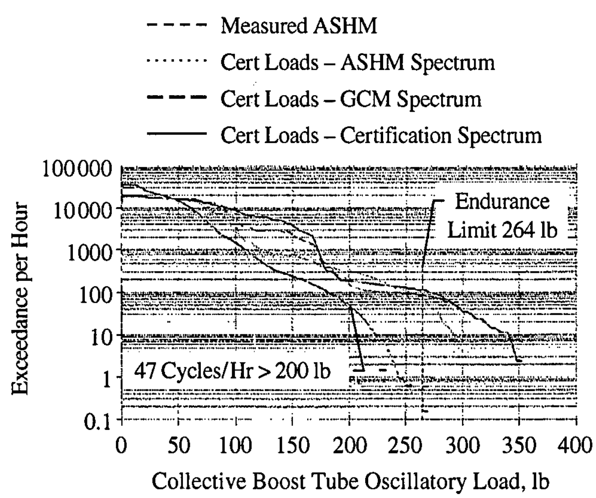


Figure 5.1 Collective Boost Tube Load Comparison

Table 4.1 Flight Hours to Critical Crack Length—0.005 Inch Initial Crack

	Certification Mission	Gulf Coast Mission	Atlanta Short Haul Mission
Rephase Lever	No Growth	No Growth	No Growth
Collectiver Lever	192	271	554
Main Rotor Spindle	No Growth	No Growth	No Growth
Main Rotor Yoke	160	7,790	2,910

Table 4.2 Flight Hours to Critical Crack Length—0.015 Inch Initial Crack

	Certification Mission	Gulf Coast Mission	Atlanta Short Haul Mission
Rephase Lever	78	259	154
Collectiver Lever	13	16	31
Main Rotor Spindle	143	104	2,557
Main Rotor Yoke	20	50	70

6. GUIDELINES FOR CERTIFICATION

For transport category rotorcraft governed by FAR 29, the requirement is that all new rotorcraft be equipped with a flight data recorder (See Paragraph 29.1459 of Reference 3). At the present time, the FAA has no specific regulatory requirement that makes a HUMS mandatory. There is a draft of an advisory circular currently being worked by a joint FAA/JAA task force that outlines what constitutes a HUMS and contains suggested certification methods.

In accordance with FAR 21 the system may be certificated by the manufacturer as part of the Type Certificate (TC) of a production helicopter or as a Supplemental Type Certificate (STC) by the manufacturer, a modifier, an equipment manufacturer, or an operator. If the system is to be retrofitted to existing aircraft, the most logical method would be an STC as a kit. This would not preclude the system from being installed on the production line in a new aircraft by the manufacturer.

No matter what the certification vehicle, TC or STC, a complete set of engineering drawings and specifications must be submitted to the certifying agency. The applicant must show that the addition of the onboard equipment would in no way be a hazard to the safe operation of the aircraft. The FAA has suggested using AC No. 25.1309-1A (Reference 4) as a guideline for safety and hazard analysis in connection with the installation of a HUMS. The hazard analysis covering both airborne and ground based aspects should be submitted to the certifying agency.

The certification process for a HUMS differs somewhat from current processes because of the use of ground based equipment including computers and software. Certification involves addressing the installation of the equipment, maintenance credit validation, and continuing airworthiness. These aspects are discussed in some detail in an American Helicopter Society paper (Reference 5).

The certification and implementation of a commercially viable HUMS will require the close cooperation of the applicant, the certifying agency, and the manufacturer. The HUMS concept is relatively new on the scene and must be approached cautiously especially regarding life extensions. The system design and installation, validation of the procedure for obtaining credit, and continuing airworthiness aspects including operator procedures and training must be complete and thorough. Of utmost importance is the need to clearly establish airborne and ground based software criticality levels and provide rationale and justification for the level chosen.

7. MINI-HUMS

Two possible simplified or "mini-HUMS" configurations were investigated. The first configuration of a simplified HUMS attempted to reduce the number of sensors and therefore the complexity and the cost of the system by allowing larger groups of conditions to be lumped together. The second configuration took a simplistic approach, based upon other analysis within this project. The certification spectrum was applied to time at altitude with the FCR reduced to high or low altitude determination, essentially the HUMS became a recording altimeter. This had the added advantage that unrecognized conditions did not contribute to the damage as the certification spectrum is fully defined.

7.1 Simplified HUMS

The simplified methodology involves broadening the conditions that are recognized by the system. The suggested configuration and parameters are listed in Table 7.1 and a broad category breakdown is shown in Table 7.2 . The safe lives resulting from the implementation of this analysis (Table 7.3) did not agree well with the results obtained from the full-up HUMS. This is due to the lack of correlation between the broad categories and the certification spectrum. The indications therefore are that the categories need be to refined and that broadening them does not provide sufficient useable data. An overview of the simplified procedure follows:

1. Measure time in broad condition types,
2. Accumulate certification damage for broad conditions, and
3. Factor damage sums from 1 and 2 by the ratio of measured time to the time from the certification spectrum.

7.2 Recording Altimeter

It may be possible to use the certification time at condition with the recorded time at altitude to determine the usage. This is equivalent to producing two certification data sets, one for below 3000 ft and another for at or above 3000 ft. The Fatigue Life calculations were reprocessed with the above assumptions and the results presented in Table 7.4.

This method has the advantage that there is very little required equipment, little or no deviation from the certification methodology and demonstrates significant life extension. Simplicity is the key to this system as it makes no attempt to measure any time at condition, only time at altitude, and therefore is simple to verify. Failures of the system would involve reverting to the existing certification data, i.e. no credit for altitude.

8. CONCLUSIONS

The usage monitoring of the Atlanta Short Haul Mission (ASHM) during the summer Olympics using a HUMS was effective. Several significant conclusions can be drawn from this study.

- The ASHM usage data indicates a significantly different type of mission from the Gulf Coast mission and are as follows:
    a. Much shorter flight duration
    b. Many more maneuvers
    c. Lower cruise airspeeds
    d. A large portion of the operating time spent on the ground
- The FCR software was able to recognize the maneuvers associated with the ASHM operation. The percentage of unrecognized data was extremely low.
- The recorded cyclic and collective boost oscillatory loads verify the conservatism of the certification loads.
- No anomalies associated with sensors were observed in the data.
- The scripted flight was useful in trouble shooting and verifying the enhanced FCR algorithms.
- Historical data for the four study components indicated that the current maintenance procedures are adequate to catch corrosion, scratches and wear. In the 16 years since the Model 412 was certificated, no catastrophic fatigue failure has occurred in any of the PSE's.
- To realize the maximum benefit from the FCR technique, it is recommended that a more refined load level survey is required. For example, the use of a low/high altitude split is justified from the current load level data. However, there are too many conditions that were not recorded during certification to consider a detailed altitude breakdown. Loads measured during the ASHM also suggest that the load level should include less severe categories of maneuvers and that the FCR should be refined to recognize the severity of maneuvers.

In summary, the usage function of HUMS performed acceptably for the ASHM using the FCR technique. This study presents comparisons of significantly different mission scenarios that must be covered presently by a single certification spectrum and has indicated that a HUMS with the usage function can be used to monitor a wide range of spectrum types.

Table 7.1 Simplified Mini-HUMS Configuration

Parameter	Status	Note/Requirement
Gross Weight	Add	Measured or Pilot Input
Nz	Add	Load Factor and Symmetric Maneuvers
Roll	Add	Asymmetric Maneuvers
Squat Switch	Add	Ground/Air Time
Airspeed	Add	
Altitude	Existing	
Rotor RPM	Existing	Torque Cycle Count
Engine Torque		

Table 7.2 Simplified Mini-HUMS

Full up HUMS	Mini Hums
Hover Side Flight Rear Flight Etc.	Hover time
Level Flight .4Vh Level Flight .6Vh Level Flight .9Vh Level Flight 1.0Vh	Level
Right Turn .6Vh Right Turn .9Vh Left Turn .6Vh Left Turn .9Vh Etc.	Maneuver
Take Off Landing Engine Start Etc.	Events

Table 7.3 Simplified Mini-HUMS Fatigue Life

	Certification (Hours)	GCM (Hours)	ASHM (Hours)
Rephase Lever	5,000	3,360	2,280
Main Rotor Spindle	10,000	6,060	5,400
Main Rotor Yoke	5,000	18,870	6,720

Table 7.4 Recording Altimeter Fatigue Life

		Hours	Life	Clock
Rephase Lever	No altitude split (Certification)	5,000	100%	100%
	Gulf Coast Altitude Split	12,910	258%	39%
	Atlanta Short Haul Altitude Split	80,320	1606%	6%
Collective Lever	No altitude split (Certification)	10,000	100%	100%
	Gulf Coast Altitude Split	20,730	207%	48%
	Atlanta Short Haul Altitude Split	45,170	452%	22%
Main Rotor Spindle	No altitude split (Certification)	10,000	100%	100%
	Gulf Coast Altitude Split	19,000	190%	53%
	Atlanta Short Haul Altitude Split	33,090	331%	30%
Main Rotor Yoke	No altitude split (Certification)	5,000	100%	100%
	Gulf Coast Altitude Split	5,760	115%	87%
	Atlanta Short Haul Altitude Split	5,460	109%	92%

The crack growth lives calculated in this study indicate relatively short inspection intervals for components that were designed to safe-life methods. This is not unexpected since the crack growth threshold stresses at initial crack length of 0.015 inch used for damage tolerance are significantly lower than endurance limit stresses used for safe life. The recently certificated Model 430 was designed from the outset to be damage tolerant and uses a zero growth philosophy (no crack growth from a 0.015 inch flaw for any flight condition). No special inspections are required for the 430 components between normal overhauls.

9. REFERENCES

[1] Dickson, B., Cronkhite, J., Bielefeld, S., Killian, L., and Hayden, R., "Feasibility Study of a Rotorcraft Health and Usage Monitoring System: Usage and Structural Life Monitoring Evaluation." DOT/FAA/AR-95/9

[2] "A User Manual for a Detailed Level Fatigue Crack Growth Analysis Computer Code," AFWAL-TR-81-3093.

[3] Federal Airworthiness Regulations, Part 29 - Airworthiness Standards: Transport Category Rotorcraft.

[4] Federal Aviation Regulations, Advisory Circular No. 25.1309-1A System Design and Analysis.

[5] "The Certification Process for Health and Usage Monitoring Systems" presented at the 53rd American Helicopter Society Forum, VA Apr. 29 - May 1, 1997.

CP140 (P3) STRUCTURAL DATA RECORDING SYSTEM

M Oore
IMP Group Limited
Aerospace Division
Halifax, NS B3L 4T1
CANADA

DH Crocker
Department of National Defence
National Defence Headquarters
Ottawa, ON K1A 0K2
CANADA

SUMMARY

Each of the Canadian CP140/CP140A aircraft, variants of the Lockheed P-3C, has been equipped with a Structural Data Recording System (SDRS) to facilitate Individual Aircraft Tracking (IAT). One of the objectives of the system is to provide usage monitoring data that will enable the Canadian Forces (CF) to quantify individual aircraft fatigue usage and crack growth rates from which optimized inspection times can be calculated. Thus inspection frequency and costs can be reduced while the safety of the aircraft is ensured.

This article provides a brief description of the SDRS and related experience acquired during the five years of system usage and data collection. An overview of the parameters recorded by the SDRS is presented as well as examples of data recorded in flight and their significance. Strain sensor recording rise/fall criteria are discussed in the context of minimizing the volume of recorded data while capturing significant data. A rise/fall criteria sensitivity study, conducted to optimize selection of the triggering gate value, is presented.

Since the SDRS zeros strain sensor readings at the beginning of each flight, a strain offset determination method was developed in order to calculate absolute strain values. This method has been substantiated by calibration tests that included validation of a Finite Element (FE) model of the wing and verification of the SDRS strain measuring system. Studies performed to assess the adequacy of the SDRS strain resolution are also presented.

Overall it is demonstrated that SDRS data can be used to generate sufficiently accurate stress spectra for fatigue and crack growth analyses.

1. INTRODUCTION

The CF has an established program for IAT of the CP140/CP140A fleet. This program uses data recorded manually on Flight Engineer Logs and represents tracking methods that are conservative, reactive, and lack timeliness of feedback. The Fatigue Life Index (FLI) and inspection intervals derived are thought to be overly conservative.

In 1993, the CF established a policy to assign airframe punishing training missions to three training aircraft, designated CP140A Arcturus, specifically configured for this role. Such severe usage results in high crack growth rates yielding short inspection intervals for these aircraft when calculated using the methods of the existing IAT program. To address this problem, the CF established an aircraft tracking system based on actual aircraft data. To this end, each of the CP140A aircraft was equipped with a SDRS to record flight parameters and structural loads.

Based on the experience with the SDRS on the three CP140A, the CF decided that it was warranted to have the system installed on all CP140 aircraft. A fleet fit installation of the SDRS on all CP140 Aurora aircraft was completed in 1997.

Support is provided to the CP140/CP140A SDRS program by means of an Aircraft Structural Integrity Program (ASIP) contracted to IMP Group Limited.

2. SDRS OVERVIEW

The heart of the SDRS is the AN/ASH-37A Structural Data Recording Set developed by Systems & Electronics, Inc. (SEI). The AN/ASH-37A is an advanced airborne structural recording system consisting of a twenty channel microprocessor based recorder-converter, a removable memory unit, a Data Entry Keyboard (DEK), a multi-axis motional pickup transducer (accelerometer), temperature compensating strain sensors and a recorder-reproducer for system ground support and data transfer. The SDRS is able to record the following flight information: engine on/off, weight on wheels (WOW) state, flap deployment, centre of gravity acceleration, wing and horizontal stabilizer peak/valley strain, altitude and airspeed.

The AN/ASH-37A system is currently in use by the United States Navy (USN) on the A-6E and E-2C fleets and on the CH-46 and AH-1W helicopter fleets.

The SDRS on the CP140/CP140A is configured to record all significant data for a 30 day period or 100 hours of aircraft operation before download. Details on SDRS recorded parameters are presented in Tables 1 and 2.

3. OPERATIONAL EXPERIENCE WITH SDRS DATA

3.1 Overstress Cases

During the period between 1993 and 1998 there were numerous instances where the vertical acceleration of the centre of gravity of the aircraft (Nz) exceeded the Aircraft Operating Instructions (AOI) limit of 3.0g. During this period, maximum Nz readings ranging from 3.06g to 3.85g were recorded. Such overstresses, when noted by the aircrew, are entered in the aircraft flight log and result in the performance of a costly overstress maintenance check before the aircraft is returned to service. However, the aircrews are often unaware that an overstress has occurred as the cockpit accelerometer indicator reads 0.3g to 0.5g

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

below the Nz reading recorded by the SDRS. As a result, it is often not until the aircraft has landed and the SDRS memory module downloaded that there is the first indication of a possible overstress.

3.2 Typical Overstress Incident

A typical flight profile (Figures 1a and 1b) illustrates what appears to be an overstress incident at approximately 2150 seconds. A zoom in on the area where the overstress occurred shows that the vertical acceleration of centre of gravity of the aircraft fluctuated between 2.20g and 3.26g. The following evidence supports the possibility of an overstress. There is a high correlation between, the Nz parameter and the wing root strain recorded by strain sensor 1. Wing root strain is proportional to wing root bending moment, which is in turn related to the magnitude of loading on the wing. However, in the present case, the moderate level of the over-stress (3.26g) coupled with an aircraft weight of only 86,000 lb at the time of the incident indicates that there may be little likelihood that any structural damage resulted from this incident. This type of incident has raised a number of questions:

How should aircrew be alerted of the potential for an overstress? Indication that an overstress incident has occurred may not happen until the SDRS memory module is downloaded and the data examined. From an operational sense this is less than satisfactory. Investigations are currently underway between the CF and SEI to establish a manner of alerting aircrews of possible overstress incidents. One possibility is to modify the DEK to include a cautionary light to indicate when preset levels of centre of gravity acceleration have been exceeded.

What constitutes an overstress? The CF, in consultation with IMP Group Limited, is in the process of defining criteria that will be used to determine whether an overstress has occurred and hence, whether there is a requirement for an overstress inspection.

How can the potential for overstress incidents be reduced? The CP140/CP140A fleet is presently operated to very aggressive limits that are defined by maximum g level. Investigations should be undertaken to examine the effect of redefining fleet operational limits in terms of bank angle. The following example illustrates that such restrictions may be beneficial.

In August 1997, a 2g manoeuvre restriction was imposed at Canadian Forces Base (CFB) Greenwood in an effort to reduce the number of costly aircraft inspections required as a result of overstresses. For the purpose of this study, overstresses were defined as Nz in exceedance of 3g. The CF examined SDRS data for the CP140/CP140A fleet stationed at CFB Greenwood for the period 01 July 1997 to 01 September 1997 to assess whether the operational restriction had any effect on the manoeuvre Nz exceedance distribution. The results of the study are presented in Figure 2, which demonstrates a significant reduction in Nz exceedances in the period after the restriction was imposed when compared to the period before the restriction. While it is premature to draw any definitive conclusions, it appears that the restriction reduced the number of Nz exceedances as well as the number of overstresses experienced.

3.3 Relationship Between Damage Severity and Flap Over speed

CP140A data from two sources were combined and analyzed to investigate the hypothesis that a relationship existed between the severity of flap overspeeds as recorded by the SDRS and flap damage severity as recorded by the CF maintenance database system. The method used to test this hypothesis was to rate the damage incidence according to an increasing scale of severity, where the highest damage severity was given the highest rating. There was a high correlation coefficient (0.99) between the mean flap overspeed value and the mean flap damage severity for the three CP140A aircraft. Statistical analysis indicates that there is a 0.087 probability that the relationship between the amount of flap overspeed and the severity of flap damage as reported in maintenance data for CP140A aircraft is due only to chance. Hence, there is a reasonable probability that a causal link exists between the two parameters. This link will be difficult to establish with a high degree of certainty given the small size of the fleet (three aircraft) being monitored. However, there is enough evidence from other sources to conclude that flap deployments outside the operational envelope are costly and should be avoided whenever possible. The focus given to the flap overspeed problem through this discovery was enough to cause operators to monitor the situation resulting in a decreased incidence of flap overspeed damage.

4. SDRS BASED FATIGUE AND DAMAGE TOLERANCE ANALYSIS

4.1 Scope and Objective

At present, each of the eighteen CP140 and the three CP140A aircraft is equipped with a SDRS to facilitate IAT. One of the objectives of the SDRS program is to provide usage monitoring data that will enable to quantify individual aircraft fatigue usage and crack growth rates from which optimized inspection times can be calculated. The goal is to reduce inspection frequency and maintenance costs while ensuring the safety of the aircraft.

Fulfilling these objectives in a practical/economical fashion required several steps. Of primary importance is verification of the SDRS recorded data. While this is crucial for any subsequent fatigue, damage tolerance and inspection interval calculations, it also impacts the confidence in conclusions obtained using SDRS data during the 5 years of experience with the system.

4.2. Calibration and Verification

4.2.1 Calibration Tests

The SDRS is configured such that the strain sensors are zeroed at the beginning of each flight. This "zero reading" is used by the system as a reference for any sensor reading recorded during that flight. This is an effective way to avoid the problem of the sensor zero position drifting over time. However such a system configuration necessitates a suitable method for strain offset determination before each flight in order to obtain correct values of absolute strain.

This implies that the actual level of strain of the fueled aircraft on the ground (before flight) need be known very accurately at each sensor location. The appropriate value of offset strain is then added to the SDRS recorded strain readings to obtain the corresponding value of absolute strain. A NASTRAN Finite Element (FE) model of the wing has been developed at IMP Group Limited to accurately determine these offset strains (see Figure 3). However, due to the complexity of the structural configuration, bending near inflection points (see Figure 4), tires stiffness and questions regarding fueling sequence and fuel distribution inside the wing and in the fuselage, it was decided to conduct strain sensor calibration testing. The steepness of the bending moment slope in Figure 4 implies that small changes in modeling or mass distribution can greatly change the bending moment in the areas near the inflection point. Calibration testing would serve to validate recorded strain data generated by the SDRS and verify the analytical tools (FE model) used in the follow up analysis.

During the installation of the SDRS two identical strain sensors, a primary and a secondary, are installed at each sensor location. Such an arrangement serves to provide an alternate sensor (the secondary) should the primary sensor fail and also facilitates SDRS calibration. By connecting a calibrated, external strain measurement system to the secondary sensors and comparing these readings to the corresponding primary sensor readings recorded by the SDRS, strain sensor readings can be validated.

Three aircraft were subjected to calibration tests. Calibration testing of each aircraft consisted of two phases. Phase I testing involved the application of measured upward and downward point loads near the wingtips. Phase II testing involved the measurement of wing strain at sensors locations for various fuel loadings.

4.2.2 Summary Of Results From Calibration Tests

Phase I results established the linear bending moment/strain relationship at strain sensor locations and verified the analytical FE model. Figures 5a and 5b illustrate this linear relationship. Figure 5b demonstrates the close correlation between the test results and the analytical model.

Phase II results are presented in Figures 6a, 6b and 6c which illustrate the complexity of this structural loading case. Figures 6b and 6c demonstrate the ability of the analytical model to provide acceptable results within the scatter band of the test data. Note that the SDRS readings can change only in 50 microstrain ($\mu\varepsilon$) increments due to the resolution of the system.

4.3 Spectrum Generation

As part of the SDRS program, the development effort also focused on software for generation of stress spectra at critical aircraft locations. Such spectra are required for fatigue and crack growth analysis from which inspection intervals are calculated. The spectra generation software includes algorithms, which use the analytical FE element model verified by calibration test data, to convert the recorded SDRS data into stress at critical aircraft locations. The software also incorporates a rainflow cycle counting algorithm to provide the proper counts of cycles. The final output is in a format that lends itself to fatigue and damage tolerance analysis using both proprietary software developed at IMP Group Limited and commercially available software packages. Damage tolerance analysis (DTA) results, based on generated spectra, provide the basis for inspection intervals and maintenance times determination.

As part of the SDRS program assessment, this spectra generation software was used in combination with the fatigue and crack growth software, to provide SDRS system data evaluation by means of sensitivity/parametric studies. These studies are described in the following sections.

4.4 Rational for Rise/Fall Criteria and Their Determination

The recording of peak/valley strain readings is governed by the rise/fall criteria of the SDRS.

The rise/fall criteria are the amount of change in strain necessary to validate the recording of a peak or a valley strain recording. The SDRS allows for specification of the rise/fall criteria value in its software configuration. The selection of a small rise/fall criteria (gate) value will ensure that small cycle amplitudes are recorded resulting in a very accurate and comprehensive database. However, this results in an overwhelming amount of data, much of which may be insignificant "noise", and may cause memory module overflow leading to loss of substantial data blocks. Optimized rise/fall criteria need to be established to ensure the recording of all significant data but without excessive data storage requirements.

To establish the optimized triggering gate, the SDRS rise/fall criteria were first set to record a very wide field including very small cycle ranges (95 $\mu\varepsilon$). The resulting data were run through software that simulates the SDRS rise/fall triggering criteria. Multiple software runs were performed each corresponding to a specified triggering gate (see Figures 7a through 7f). The resulting files were used to generate stress spectra at a wing lower front spar root fitting. These spectra were then used to conduct the fatigue crack growth analysis presented in Figure 8. The intent was to establish the rise/fall criteria value beyond which reducing the triggering gate had no significant effect on the fatigue crack growth results. It is observed from Figure 8 that there is no significant difference in crack growth results between the 142 $\mu\varepsilon$ and the 95 $\mu\varepsilon$ values of the rise/fall criteria. Spectra generated using the optimized rise/fall criteria value and the next highest value were subjected to fatigue crack growth testing using a centre cracked tension (CCT) test specimens (see Figure 9). This analysis and testing sequence reduced the number of tests normally required in spectrum truncation tests. The tests presented in Figure 9 confirmed that including cycles with rise/fall differences below the analytically determined optimum triggering point had no significant effect on the crack growth test results.

This study also allowed an assessment of the savings in data storage. Summaries of the rise/fall criteria sensitivity crack growth life analysis results and the corresponding number of strain pairs are presented in Figures 10 and 11.

4.5 Study of Resolution Effects

4.5.1 Background

The strain resolution limit of the SDRS is inherent in the system design. To facilitate practical memory requirement, all measured values of strain that fall within the same resolution limit of 50 $\mu\varepsilon$ are assigned the same strain value. For example all strain values between 75 $\mu\varepsilon$ and 125 $\mu\varepsilon$ are recorded as 100 $\mu\varepsilon$; all strain values between 125 $\mu\varepsilon$ and 175 $\mu\varepsilon$ are recorded as 150 $\mu\varepsilon$ and so on. Thus all measured strain values are rounded within a 50 $\mu\varepsilon$ range (25 $\mu\varepsilon$ amplitude). The question arises as to the effect of the resolution limit on fatigue and crack growth results. Since the resolution parameter is built into the system and cannot be reduced for the sake of a parametric sensitivity study (as with the rise/fall criteria), a Monte Carlo simulation approach was used to assess the effect of the system resolution on fatigue and crack growth results.

4.5.2 Analysis and Results and Conclusions

SDRS generated data were used to generate a stress spectrum for a critical front lower spar cap and web location. This spectrum was then used to conduct analytical fatigue and crack growth analyses. Because any monitoring system has inherent measurement errors associated with its resolution and rounding errors, a random error analysis was conducted to establish the effect of the resolution error on fatigue and crack growth life predictions. The strain readings resolution errors were simulated with computer generated random numbers within the bounds of the resolution specification of the system. The simulated errors were superimposed on the basic spectrum derived from the SDRS recorded data.

This revised spectrum, which incorporates resolution error effects, was then used to repeat the fatigue and crack growth analyses. The results were used to quantify the potential effect of the resolution random errors on the fatigue life predictions as elaborated in the following. Figure 12 presents a comparison of several fatigue analyses with results normalized by the basic fatigue result. Series 1 in Figure 12 presents fatigue life results for a conservatively adjusted spectrum in which the maximum resolution limit is added to each peak strain and subtracted from each valley strain in the recorded data. It shows that for a resolution range of 50 $\mu\varepsilon$, this conservative approach gives fatigue life that is 35% shorter than the life obtained with the basic spectrum. Hence, this approach may be too conservative for practical evaluation of the error effect. Another approach, using a spectrum that incorporate uniform random error simulation of the resolution, gave the fatigue life results shown by series 2 in Figure 12. It is observed that for resolution errors within 50 $\mu\varepsilon$ the fatigue life result decreased by only by 4%, whereas for resolution of 100 $\mu\varepsilon$ the resulting life is 12% below the basic fatigue life. Additional scenarios are presented in Figure 12 to demonstrate various simulations of the resolution effects. Series 2 is deemed to be the most realistic as the resolution error is expected to be of uniform random distribution within the resolution limits. Since the analysis generating series 2 represents a random effect, fatigue analysis was repeated with several sets of uniform random numbers (Monte Carlo simulation) in order to establish confidence bands. The results for these repetitions displayed a small variation within each resolution range group. The frequency distribution of the results for the 50 $\mu\varepsilon$ cases gave a histogram that rationalized confidence intervals calculation using a t- distribution. Very high confidence (99.99%) bounds were established that the mean population of the fatigue life is between 25,000 to 26,000 hours whereas the basic fatigue calculation gave 26,500 hours. This demonstrates that the 50 $\mu\varepsilon$ resolution effect on the basic fatigue life results could be at most a 6% reduction in fatigue life. Larger resolution error ranges will have a greater effect on fatigue life (series 2) while a 25 $\mu\varepsilon$ resolution will have an effect of significantly less than 6%. Series 3 in Figure 12 presents the effect of increasing the mean strain on fatigue life. The strong effect observed confirms the requirement for the offset strain determination and the calibration testing described in Section 4.2.

The potential effect of the resolution error on crack growth was assessed in a similar manner to that described above for fatigue life. The basic spectrum derived from SDRS generated data was used also to conduct crack growth analysis for multiple crack growth paths at a critical lower spar web location on the wing. Initial crack lengths of 0.05" and 0.25" were considered. To assess the effect of resolution errors on crack growth life predictions, the resolution effect was simulated as before with uniform distribution random errors within ± 50 $\mu\varepsilon$. These were superimposed on the basic spectrum derived from the recorded data in the same manner as for the fatigue analysis. This superposition was repeated for 12 sets of random numbers simulating resolution errors of 50 $\mu\varepsilon$ and the resulting 12 spectra were used in crack growth analyses. Results of these analyses show only small variations, due to resolution error, from the basic crack life results. A statistical confidence analysis was conducted which demonstrated that resolution errors within 50 $\mu\varepsilon$ could reduce the crack growth life results at most by 5% when compared to the results using the basic spectra.

It was concluded from the above that the 50 $\mu\varepsilon$ resolution limit of the SDRS is satisfactory. The system resolution limit does not have a significant effect on fatigue and crack growth analyses using SDRS derived data.

Acknowledgements

The authors wish to thank BK MacLeod of IMP Group Limited for his outstanding technical support in the preparation of this paper.

Parameter	Units	Sampling Rate	Range	Resolution	Filter
Elapsed Time	seconds (s)	n/a	167,772	10^{-1}	n/a
Indicated Altitude	feet (ft)	2 Hz	-500 to +40,000	0.0098 psia/count	6 Hz / 3 pole
Indicated Airspeed	knots (kt)	2 Hz	0 to 400	0.0195 psid/count	6 Hz / 3 pole
Centre of Gravity Acceleration (Nz)	g	64 Hz	-1.5g to 4.0g	0.098g	12 Hz / 5 pole
Strain Sensor 1 (WS 92R)	microstrain (με)	32 Hz	-3500 to +3500	50 με	12 Hz / 5 pole
Strain Sensor 3 (WS 147R)	microstrain (με)	32 Hz	-3500 to +3500	50 με	12 Hz / 5 pole
Strain Sensor 3 (WS 223R)	microstrain (με)	32 Hz	-3500 to +3500	50 με	12 Hz / 5 pole
Strain Sensor 4 (HSS 30R)	microstrain (με)	32 Hz	-3500 to +3500	50 με	12 Hz / 5 pole
Weight on Wheels (WOW)	n/a	per flight	0 or 1	n/a	n/a

Table 1 SDRS Recorded Parameters

Mission Data Entered on SDRS Data Entry Keyboard (DEK)	
Parameter	**Format**
Date	6 digits (ddmmyy)
Time	4 digits (hhmm)
Mission Code	(1, 2, 3 or 4)
Takeoff Fuel Weight	3 digits (000s lb)
Takeoff Gross Weight	3 digits (000s lb)
Airframe Hours	6 digits (hours)
Landing Fuel Weight	3 digits (000s lb)
Landing Gros Weight	3 digits (000s lb)

Table 2 Manually Entered SDRS Parameters

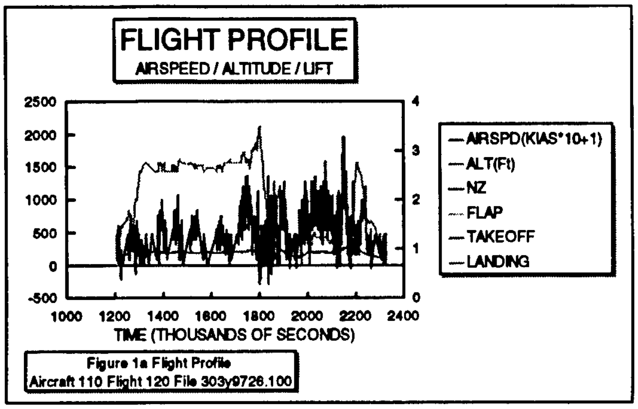


Figure 1a Flight Profile of an Overstress Incident

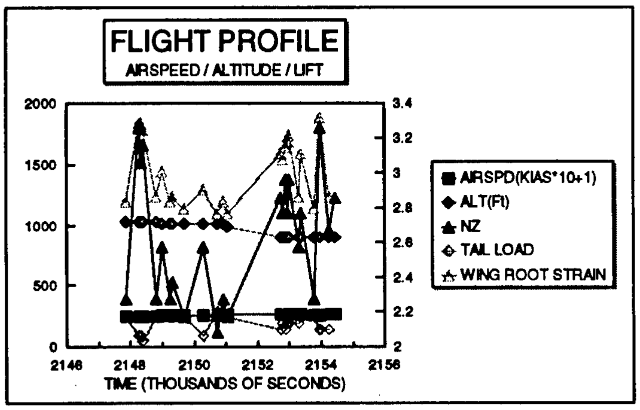


Figure 1b Zoom on Figure 1a Overstress Incident

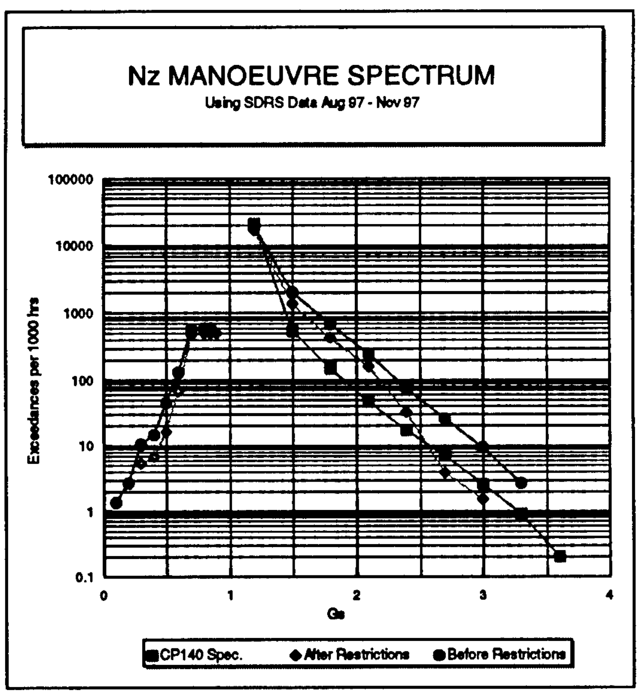


Figure 2 Vertical Acceleration Manoeuvre Spectrum

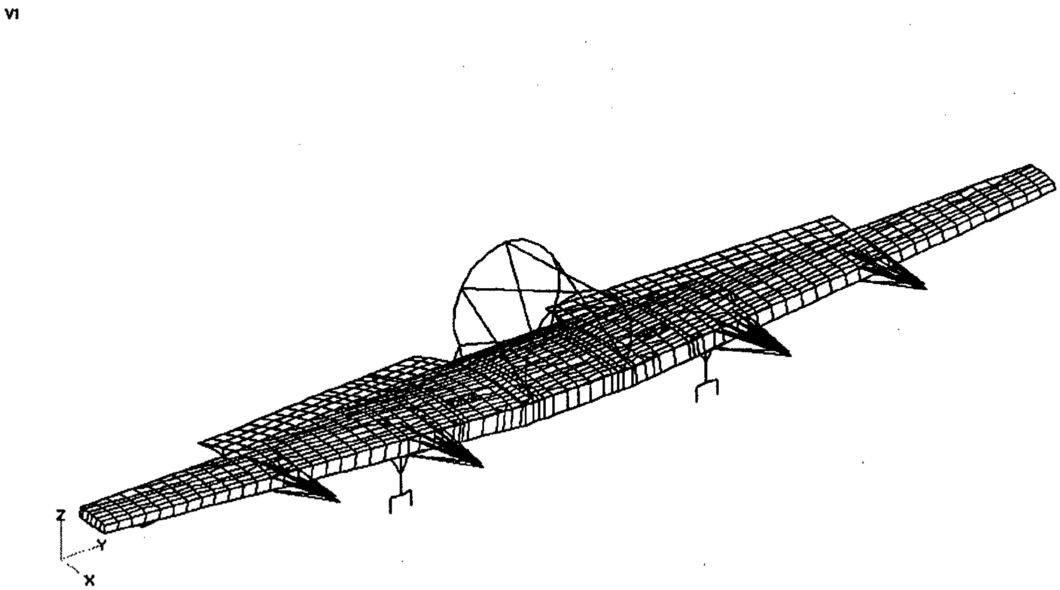


Figure 3 Wing Finite Element Model

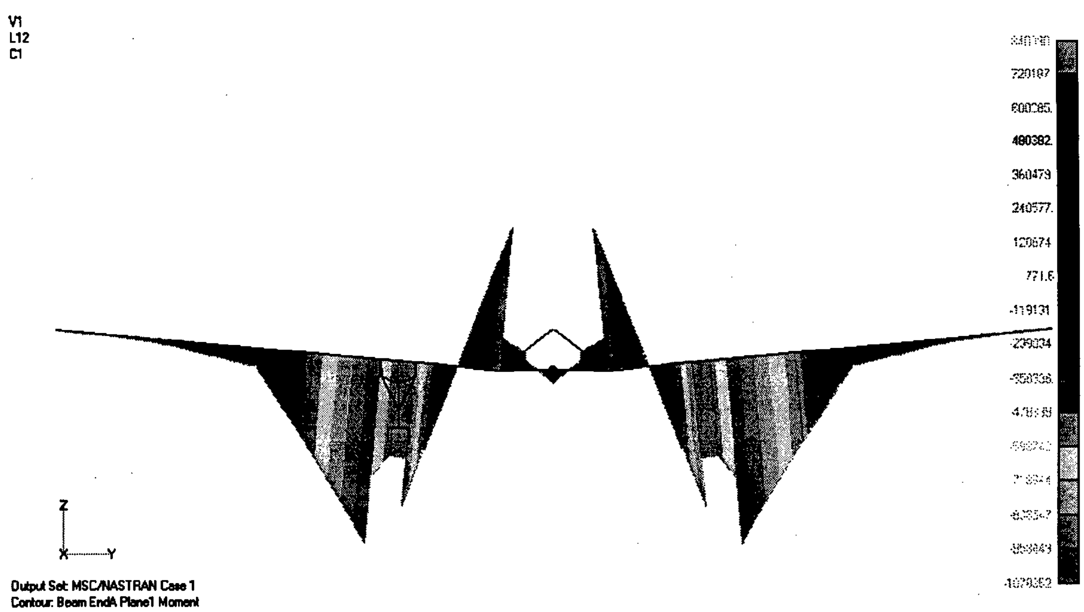


Figure 4 Wing Bending Moments

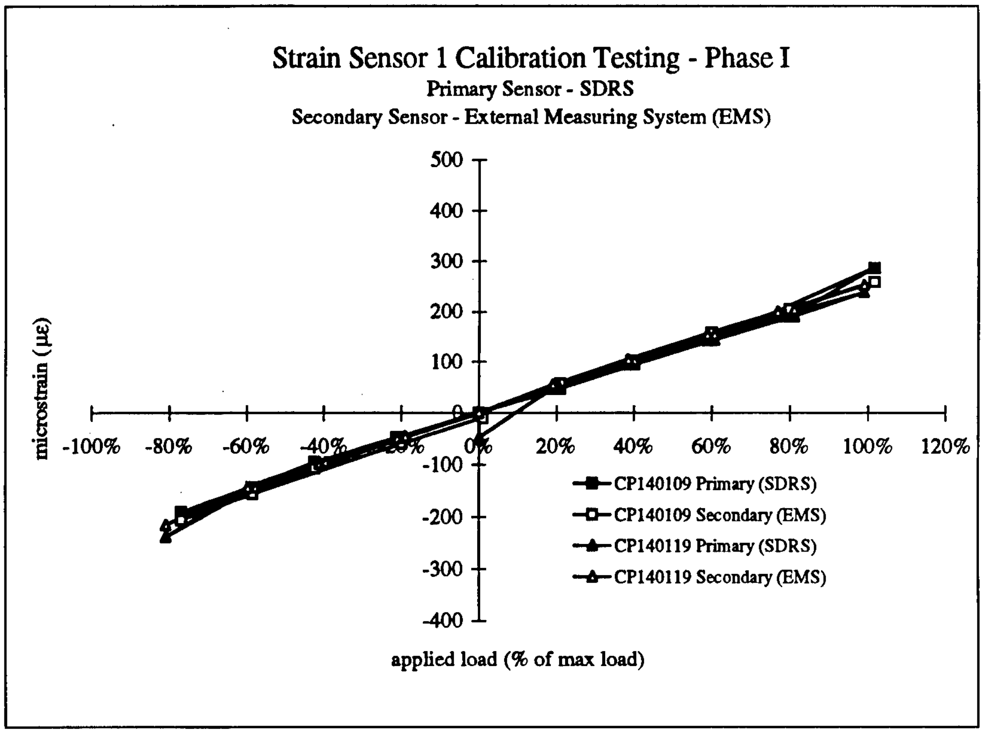


Figure 5a SDRS Calibration Phase I Test Results

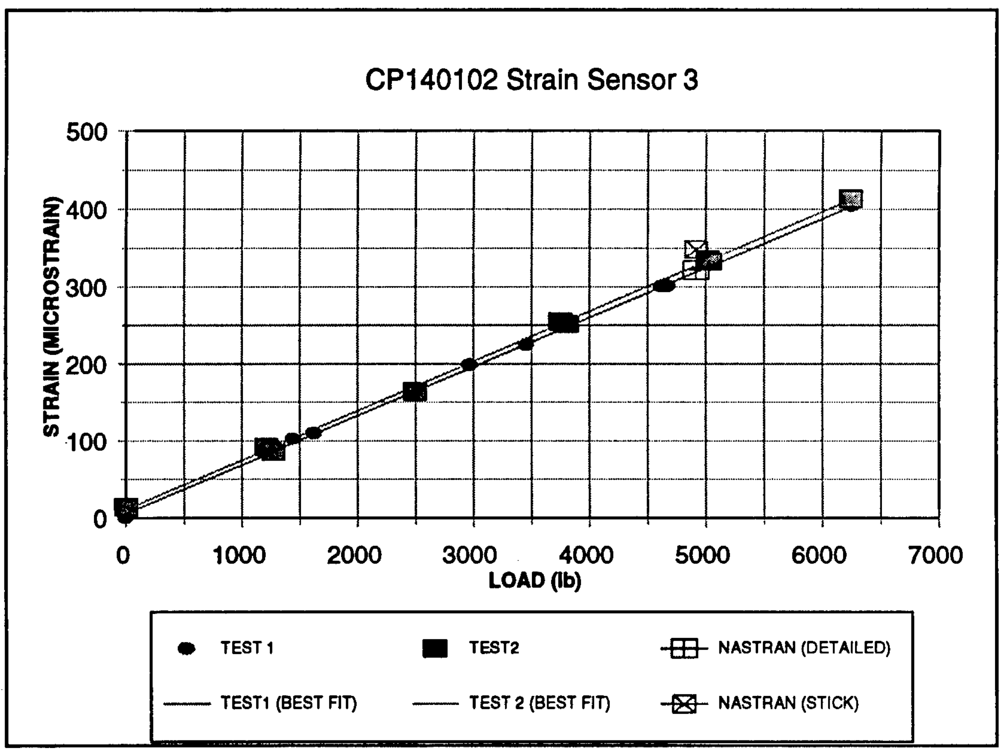


Figure 5b SDRS Calibration Phase I
Test Results and Analytical Model

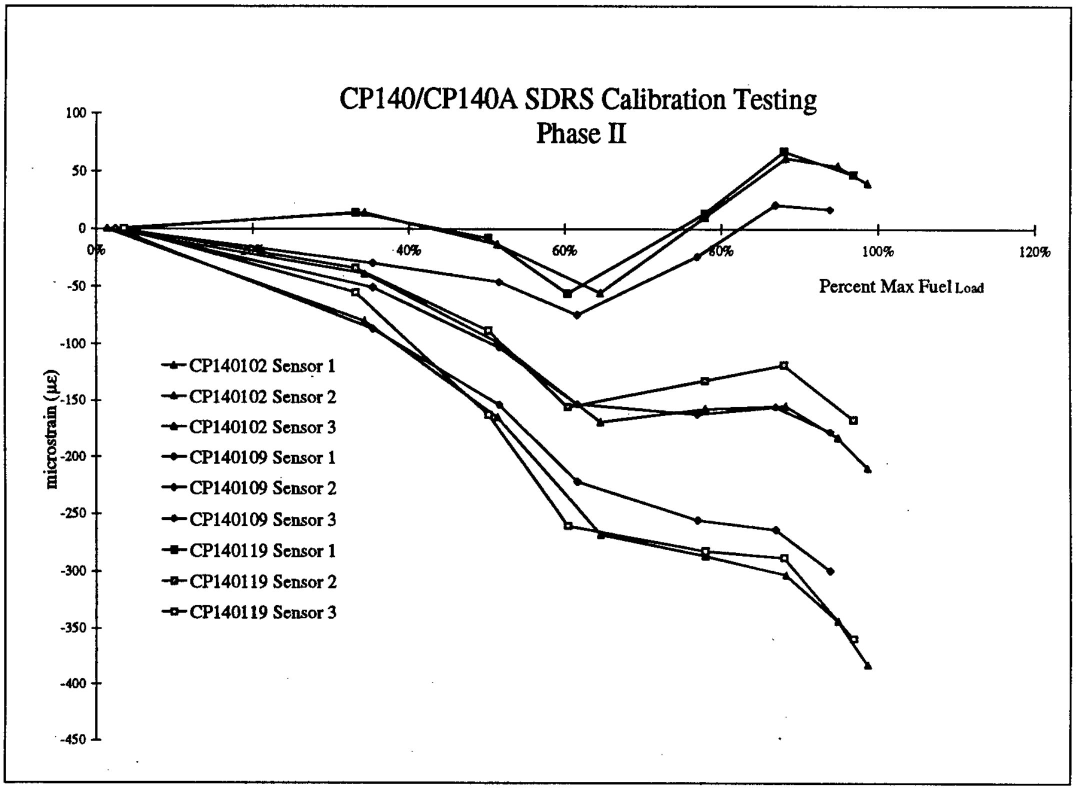


Figure 6a SDRS Calibration Phase II Test Results

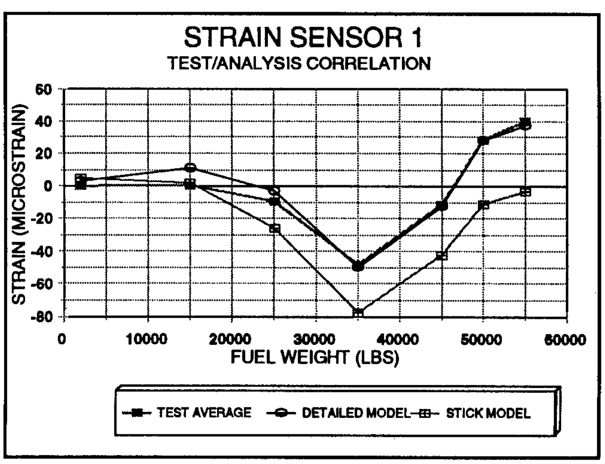


Figure 6b SDRS Calibration Phase II Test Results and Analytical Model

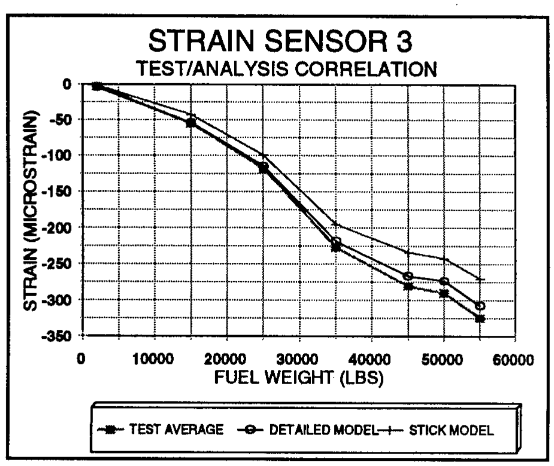


Figure 6c SDRS Calibration Phase II Test Results and Analytical

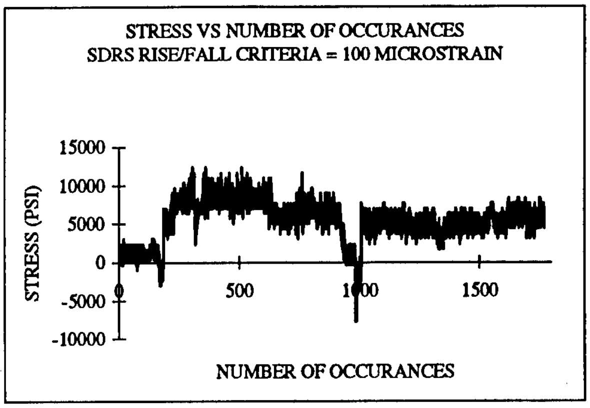


Figure 7a

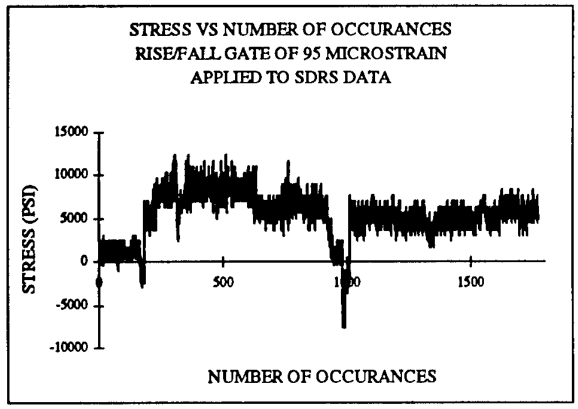


Figure 7b

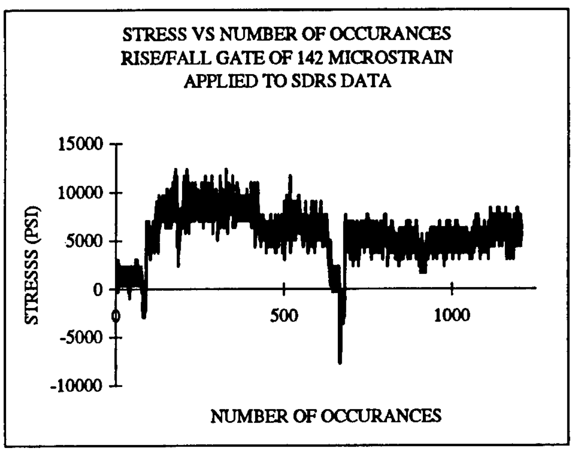


Figure 7c

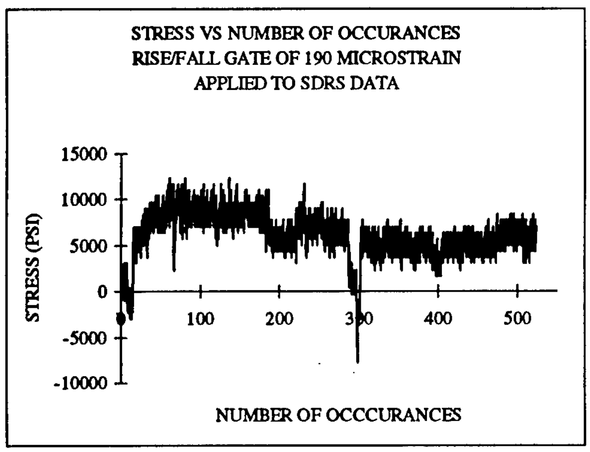


Figure 7d

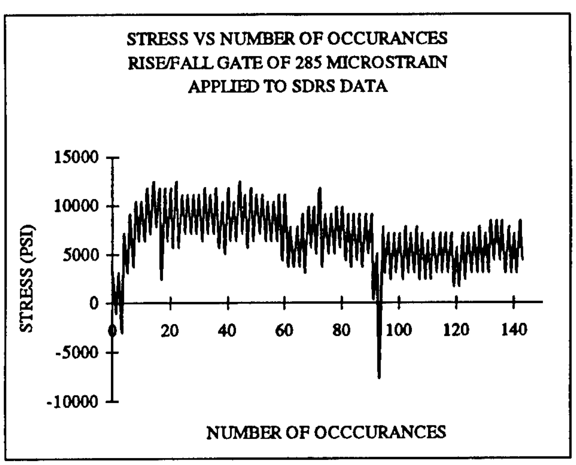


Figure 7e

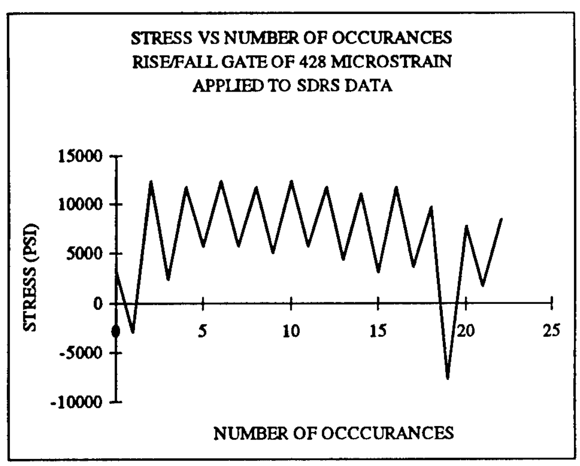


Figure 7f

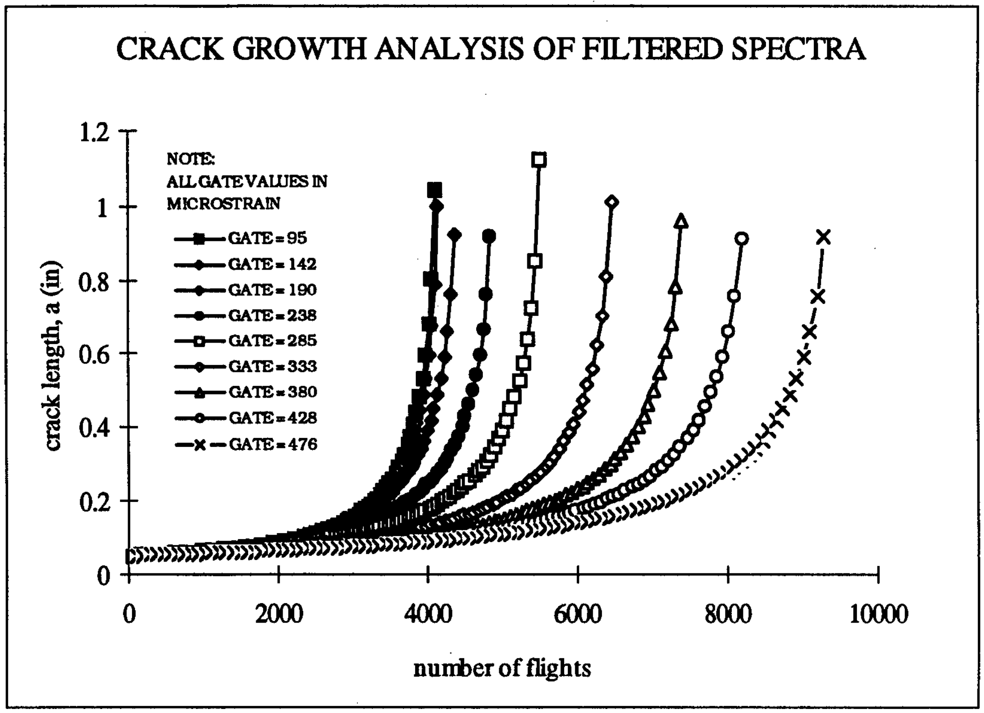


Figure 8 Rise/Fall Gate Effect on Fatigue Crack Growth

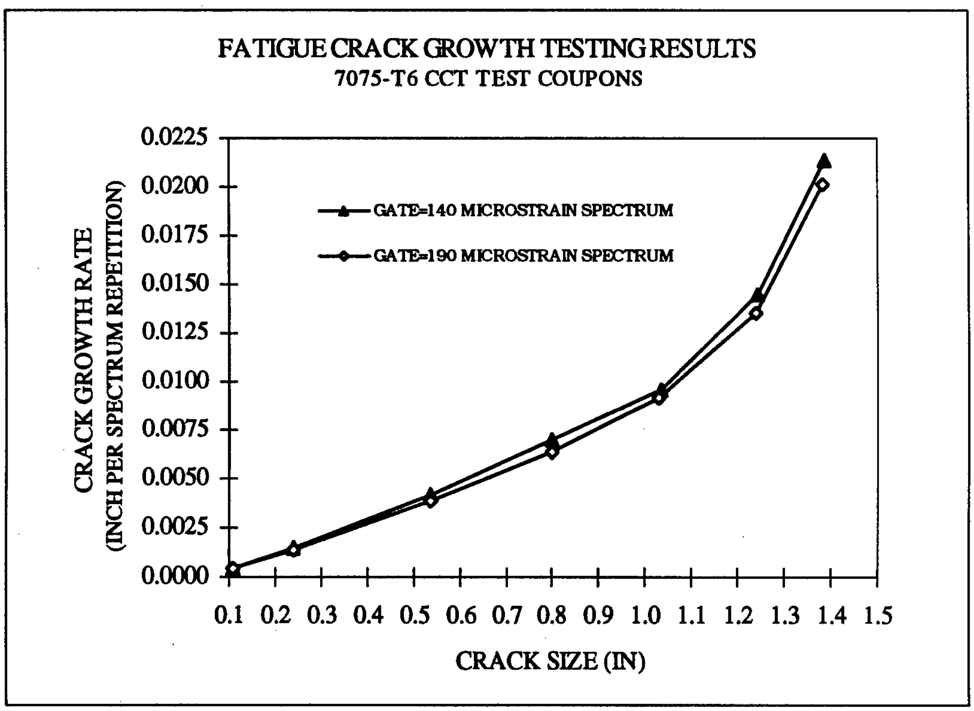


Figure 9 Fatigue Crack Growth Testing Results

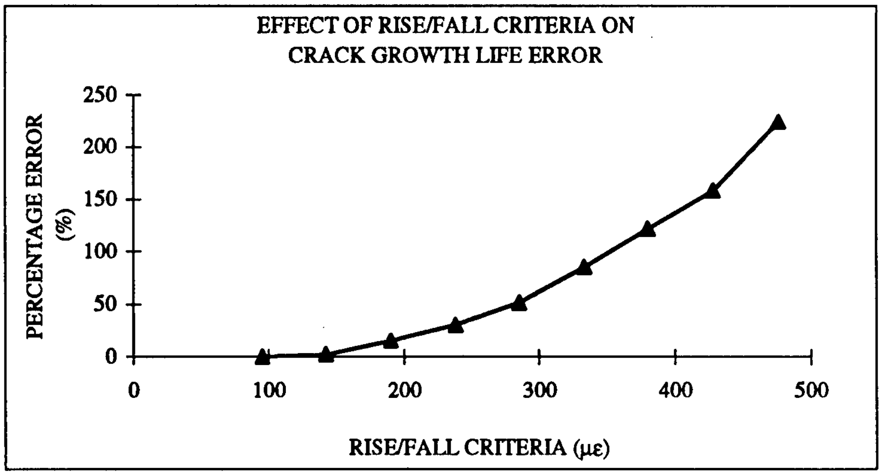


Figure 10 Effect of Rise/Fall Criteria on Grack Growth Life Error

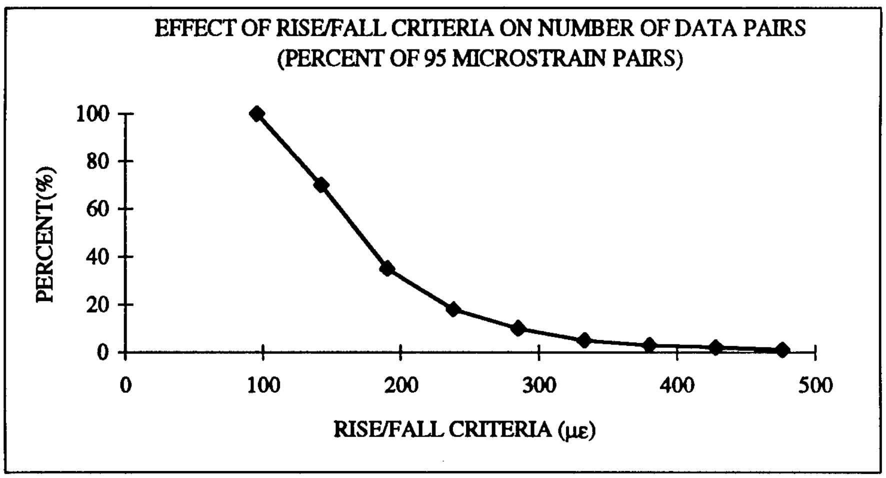


Figure 11 Effect of Rise/Fall Criteria on Number of Data Pairs

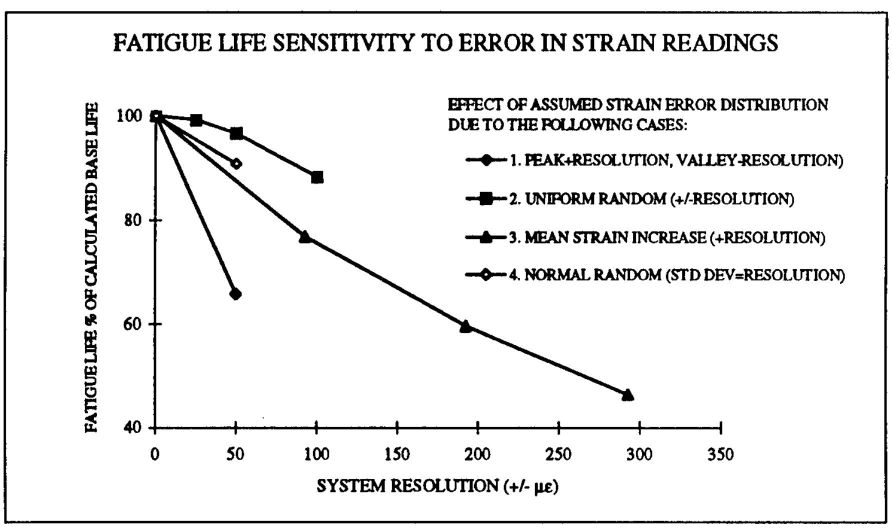


Figure 12 Fatigue Life Sensitivity To Error in Strain Readings

NATO TCA Cycle Counting Study and its applications

ir. E. Moyson
Engineering General Manager, Sabena
Brussels National Airport 12/018
B-1930 Zaventem, Belgium

ABSTRACT

Three former Sabena 707-329c's were modified and transferred to NATO as AWACS trainer and transport aircraft.

Compared to commercial B707's the Trainer Cargo Aircraft (TCA) accumulate more cycles including touch and go landings. One engine out landings and aerial refueling training are significantly more damaging in some structural areas.

For operational flexibility, no mission profiles are defined. TCA missions are variable combinations of different mission events.

Available data in the area of loads, stresses, fatigue and fracture mechanics are used to develop a comparative crack growth relation between TCA events and commercial B707-320c flights.

After adjustment of cycle intervals i.a.w. the cycle counting study, all structural inspection requirements were consolidated in a calendar based integrated maintenance program.

The design service objectives are extended and major structural modifications can significantly be postponed.

0. ABBREVIATIONS

AD	Airworthiness Directive
A/R	Aerial Refuelling
CPCP	Corrosion Prevention and Control Program
CR	Cruise
DADTA	Durability and Damage Tolerance Analysis
DSO	Design Service Objectives
E/O	One Engine Out
FAA	Federal Aviation Administration
FI	Fatigue Index
GAG	Ground Air Ground cycle
M/A	Missed Approach
MS	Maintenance Schedule
SB	Service Bulletin
SSD	Significant Structural Detail
SSI	Structural Significant Item
SSID	Supplemental Structural Inspection Document
STG	Structures Task Group
TCA	Trainer Cargo Aircraft
T/G	Touch and Go
T/O	Take-off
TOL	Take-off and Landing

1. DESIGN SERVICE OBJECTIVES

The minimum design service objectives of the B707 airframe to operate with a low probability of costly non-accidental damage, occurring in primary structure on the majority of airplanes in the fleet, were:

20 years
60,000 flying hours
20,000 flight cycles.

2. DESIGN PHILOSOPHY

2.1. KC-135 versus B707

The military KC-135 jet tanker-transport was initially identified by Boeing as Model 717. While generally similar to the commercial 707, the KC-135 was

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

designed to an entirely different service life philosophy.
Considering that the military model would fly only one-tenth as much as a corresponding commercial model in a year, the KC-135 was designed to a safe-life requirement while the commercial model was designed to be fail-safe.
These differences also called for different materials: 7178 aluminum in the KC-135 and the more damage tolerant 2024 alloy in the 707. In 1995 the KC-135 tanker fleet has accumulated only an average of about 13,000 flight hours, 3,200 cycles, and 36 years.
On the B707, Boeing performed full scale fatigue testing up to 50,000 cycles, or 2.5 times the minimum design service objective

2.2. Fail-safe
Structures not practical to design or qualify as damage tolerant, are removed from service before fatigue cracking is expected. Safe-life of a part subjected to fatigue loads is determined by fatigue tests. The effective safe-life is then calculated by means of a safety factor.

Structural fatigue was first addressed in 1956, for pressure cabins only after the 1954 Comet accident. The Aloha B737, who was the catalyst of the aging airplane activity, accumulated 80,090 total cycles, compared with the 1,290 of the Comet.

The Comet was the beginning of the fail-safe approach, extended to the overall structure in 1959. Fail-safe is the ability of a damaged structure to withstand an infrequent high load (usually design limit) without placing the aircraft in an uncontrollable condition.
Fail-safe design (multiple load path) with adequate inspection is the basis for damage tolerant structure.

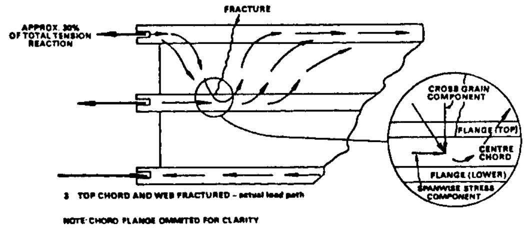


Figure 1

2.3. Damage Tolerance
The failure of a B707-300C horizontal stabiliser (Fig. 1) in 1977 with 16,723 accumulated flight cycles questioned the validity of the fail-safe concept for potential continued operation beyond the initial design life.
The resulting damage tolerance approach to detect damage before it becomes critical had to be adopted in the USA since 1978.

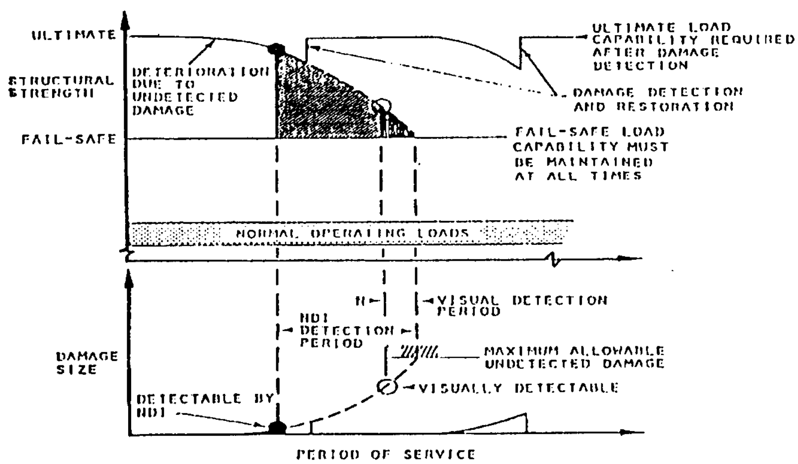


Figure 2

Damage tolerance is the ability of a structure to sustain anticipated loads in the presence of damage such as cracks and corrosion, until damage is detected through inspection or safe malfunction, and repaired, before it reaches its critical size defined by the residual strength requirements (Fig. 2). A damage tolerant structure requires that inspections are part of the maintenance program.

3. AGING AIRCRAFT

3.1. Beyond design service objective

Economic and market conditions have resulted in the use of airplanes beyond their original economical design service objectives.
These design service objectives of transport aircraft are no hard service lives as for fighter aircraft, but are supported by maintenance programs

There is no limit to the service life of damage tolerant airplane structure provided that necessary inspections are accomplished along with timely repair or replacement of damaged structure or with preventive modifications for airplanes exceeding economical design service objectives.
Aircraft operating beyond their economical design objectives, may need adjustment to their maintenance programs to continuously meet the required ultimate strength, fail-safe and damage tolerance.

Requirements for ensuring continuing airworthiness of aging jet transports were first addressed in the mid-70's. A basic assumption (Fig. 3) was that existing maintenance programs were controlling corrosion below levels that could affect airworthiness.

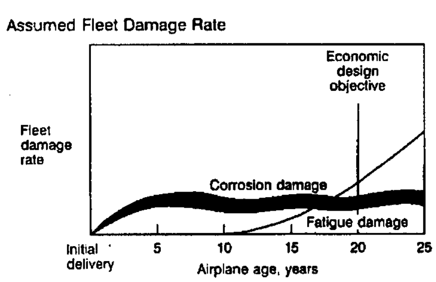


Figure 3

Evaluations centred on requirements for controlling fatigue damage, resulted in the SSID programs.

3.2. Corrosion Prevention and Control Program

Surveys and events in the late-80's have indicated an increasing number of corrosion problems in aging airplanes (Fig. 4). Some operators allow corrosion to progress to levels that require major repairs or replacement of parts.
As airplanes age, corrosion is more likely to be associated with other forms of damage, such as fatigue cracking.
This can lead to an unacceptable degradation of structural integrity that, in an extreme instance, can result in the loss of an airplane.

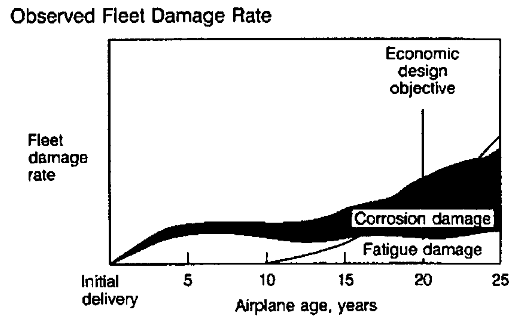


Figure 4

As airplane age, the probability of both fatigue and corrosion damage occurring increases, and maintenance requirements for controlling potential combinations of such damage are not analytically predictable. Corrosion accelerates fatigue damage, increases the potential for multiple-site damage and can significantly reduce available strength.
The effectiveness of a corrosion control program is determined for a given airplane area by the level of corrosion found. An effective program is one that controls corrosion to level 1 or better. The maintenance program needs to be amended when there are typical level 2 (exceeding allowable limits) findings during subsequent inspections, or when there is a level 3 (airworthiness concern).

3.3. Modification Program

Contrary to some maintenance programs that rely solely on inspections to maintain structural airworthiness, the SB modification program emphasises the accomplishment of structural modifications, based on the criteria of a potential safety problem, high probability of occurrence and difficulty of inspection.
For service bulletins with insufficient history of in-service failures for the STG to mandate structural modifications, inspections were mandated in areas with

know structural distress which could impair the structural airworthiness.

3.4. Repair Assessment Program

A repaired structure is not the same as originally certificated. Therefore certain assumptions may not be valid, particularly for damage detection. Large repairs, covering main structure may require additional inspection to maintain the damage tolerance of the obscured structure.

The purpose of the repair assessment is to establish damage tolerance. Meanwhile, it has been standard policy to have all structural repair designs reviewed by Boeing. This should ensure the repair quality to be of the highest standard possible. Non-obscuring flush repairs with internal doublers were installed whenever possible.

4. CYCLE COUNTING STUDY

4.1. TCA Service Objectives

Presently, the corrosion DSO (20 years) of the TCA is exceeded by more than 50%, and LX-N19996 exceeded the 20,000 cycles DSO. The plan is to operate the TCA up to the year 2020 or 2.6 times the corrosion DSO and 2.2 times the fatigue DSO.

Since the calendar design goal is largely exceeded, corrosion damage will be very likely. To control corrosion, the CPCP is being implemented, and reporting is provided to the NATO Corrosion Prevention and Control Board.

Additional considerable fatigue damage may be expected. While the world-wide highest time may have accumulated more flying hours, there is no airframe in the 707 community that has accumulated such a high number of years (corrosion) and flight cycles (fatigue).

4.2. SSID Adjustment

Supplemental Structural Inspection Document (SSID) repeat inspection intervals may be increased (decreased) for airplanes which operate at a lower (higher) than average flight length, except for intervals of Significant Structural Details (SSD) controlled by an AD. The adjustment is only applicable to selected wing details.

These guidelines were developed for standard commercial airplane operations, and not to handle certain unique aspects of training missions.

Adjustments for training flights may be made to sampling details by considering full stop landings during training as one full flight cycle, and two touch and go landings as equivalent to one flight cycle. This adjustment is not applicable to fatigue problems and details controlled by an AD.

The SSID applies to standard commercial airplane operations. If an airplane is subject to a significant number of touch and goes, as is the case for the TCA, the above is not applicable anymore.

Additionally, TCA missed approaches, one engine out operations and aerial refuelling are not covered by commercial programs.

4.3. EC-18B DADTA

A Durability and Damage Tolerance Analysis (DADTA) was developed by Boeing for the USAF EC-18B, ex-American B707-320c.

Three mission profiles were defined :

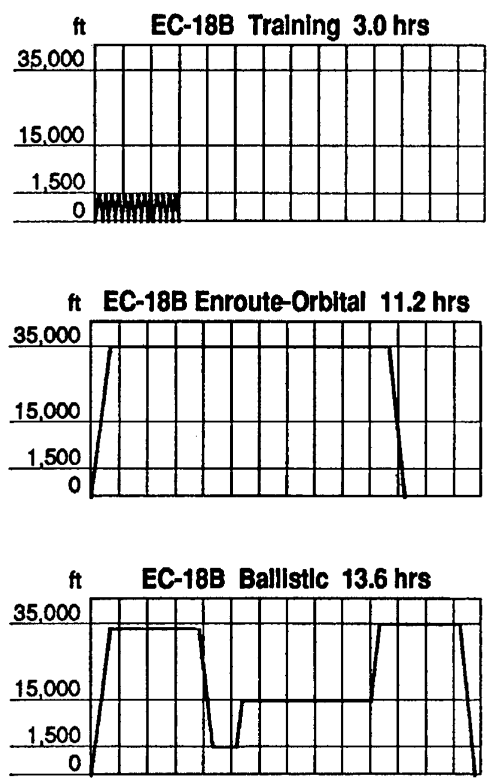


Figure 5

Comparative durability and damage tolerance between B707-320c and EC-18B airplanes is developed from crack growth analysis using the same methodology that was used in the B707-320c analysis to define inspection thresholds and repeat intervals for the SSID. Considering the number of missions flown per year, the relative severity between the commercial B707-320c and EC-18B usage was assessed.

4.4. TCA Mission Profile

TCAs were maintained per the commercial maintenance program, based on an average flight length of 2.8 hours.

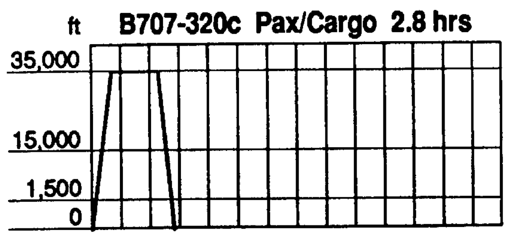


Figure 6

Compared to commercial B707's, the TCAs accumulate more cycles including touch and go landings. One engine out and aerial refueling training are significantly more damaging in some structural areas. For operational flexibility, no mission profiles are defined.

Almost any TCA mission is a combination of cruise, training and aerial refuelling without a full stop in between, and with variable cruise and aerial refuelling length:

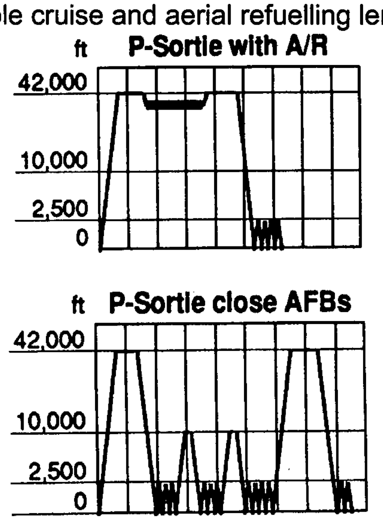


Figure 7

4.5. TCA Events

The TCAs are utilised for passenger/cargo flights, and training.

For operational flexibility, events are defined to model any TCA missions as a combination of different events : touch and go (T/G), missed approach (M/A), one engine out (E/O) and aerial refuelling (A/R):

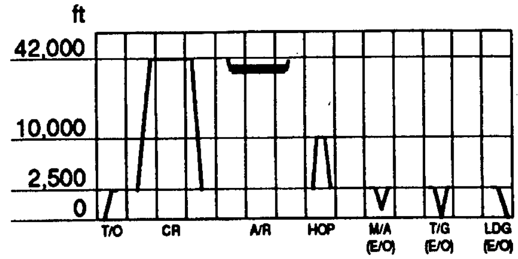


Figure 8

These events are subdivided in flight segments and conditions consistent with the segments and conditions used for previous B707-320c and EC-18B analyses.

FLIGHT SEGMENT	Flight Condition	Events						
		CR	A/R	HOP	E/O	M/A	T/G	TOL
GROUND	1 Unloaded							■
	2 Controls Check							■
	3 Tow							■
	4 Pivot							■
	5 Ground Turn							■
	6 Braked Roll							■
	7 Taxi Out							■
TAKE OFF	8 Take Off Roll						■	■
	9 Rotation						■	■
	10 Lift Off						■	■
	11 Gear Retraction						■	■
CLIMB	12 Flaps Down Departure					■	■	■
	13 Flap Retraction	■		■				
	14 Initial Climb	■		■				
	15 Final Climb Low Cruise			■				
	15 Final Climb High Cruise	■	■					
CRUISE	16 Cruise Gust Training					■	■	■
	16 Cruise Gust Low Cruise			■				
	16 Cruise Gust High Cruise	■	■					
	16 Cruise Maneuver Training					■	■	■
	16 Cruise Maneuver Low Cruise			■				
	16 Cruise Maneuver High Cruise	■	■					
DESCENT	17 Initial Descent High Cruise	■	■					
	17 Initial Descent Low Cruise			■				
	18 Final Descent	■		■				
APPROACH	19 Flap Extension					■	■	■
	20 Flaps Down Approach					■	■	■
	21 Roll Maneuver					■	■	■
	22 Yaw Maneuver					■	■	■
LANDING	23 Gear Extension						■	■
	24 Flare						■	■
	25 Touch Down						■	■
	26 Landing Roll Out/Flaps Retraction						■	■
	26 Buffeting						■	■
CREW TRAINING	27 One Engine Out				■			

Figure 9

4.6. Structural details

The DADTA developed by Boeing for the USAF EC-18B, ex-American B707-320c, is used as a basis.

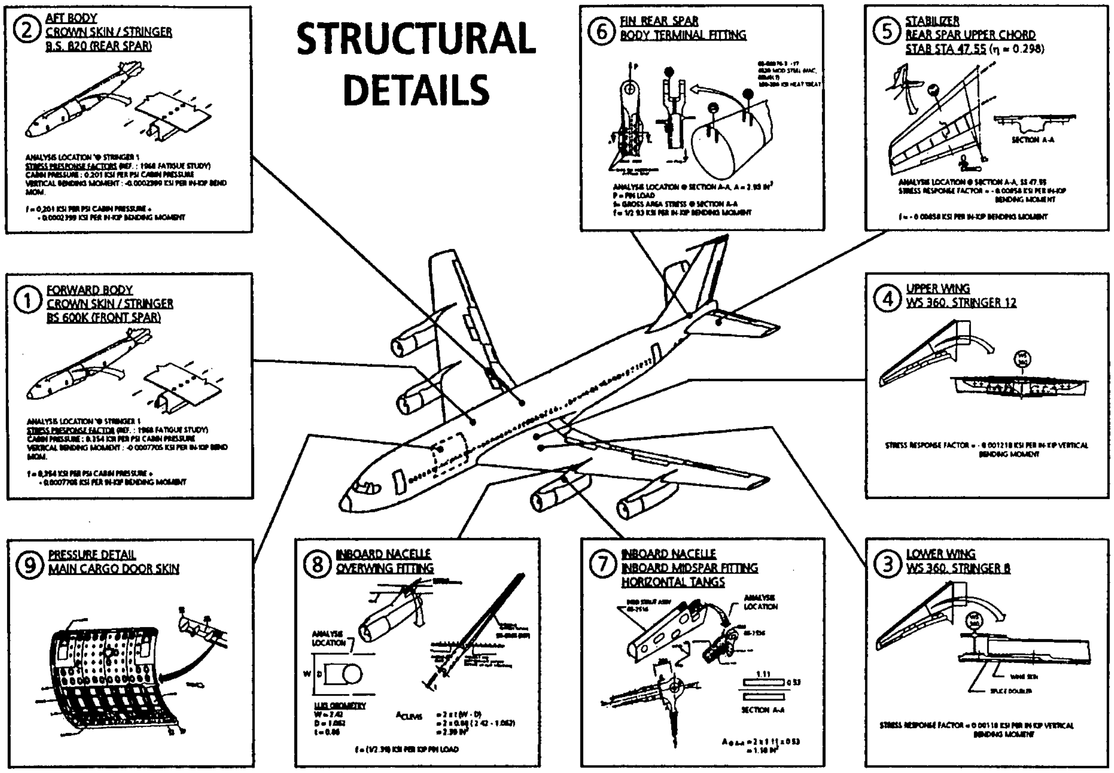


Figure 10

A comparative crack growth relation between B707-320c and TCA airplanes is developed from crack growth analysis using the same methodology that was used in the B707-320c and EC-18B analysis to define inspection thresholds and repeat intervals for the SSID.
Nine (9) structural details (Fig. 10), including a pressure only loaded detail, were selected for analysis. Detail 3/C lower center wing is added for differences in severity during aerial refuelling.

At least one detail from each component was selected to provide the relative Crack Growth relation to be applied to all details within that component.

4.7. Stresses

Flight segment stresses, ground air ground (GAG) cycle stresses and crack growth data for the nine selected analysis locations were obtained from previous B707-320c, and EC-18B training, damage tolerance analysis. The take-off and landing weights for the TCA and EC-18B training mission are similar to the typical weights used for the B707-320c.

Stresses for the longer EC-18B enroute/orbital (11.2 hours) and ballistic (13.6 hours) with consequently more fuel and a higher take-off weight, are therefore not representative for the TCA operation.

4.8. Aerial Refueling

Aerial refuelling is significantly more damaging than cruise in some E-3A areas:

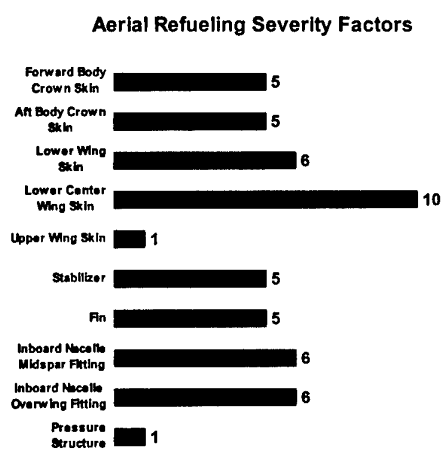


Figure 11

All other areas were shown to be about the same for cruise and aerial refueling.

These severity factors (Fig. 11) apply to the cruise flight segment only.
The type of refueling vehicle is also important. USAF E-3A data are applicable for refuelling from a KC-135 tanker. Severity factors for aerial refuelling from KC-10 refuelling vehicles will be higher because of the more powerful engines, and the presence of a center engine. The loading on the horizontal stabiliser and vertical fin is also more severe during aerial refuelling than for cruise.
The higher vertical fin of the E-4 (military B747) needed reinforcement. E-3A data do not show similar higher loading : the rotodome is masking the stabiliser and fin. Damage rates on the fin are estimated similar as for the body crown: severity factor 5. The same damage rate is considered for the horizontal stabiliser in this cycle counting study.

4.9. Crack Growth factor

The crack growth factor is obtained by dividing the total relative crack growth "T" by the "T-value" for the corresponding commercial flight cycle (2.8 hours), and indicates the relative severity between B707-320c and TCA events.
The composite crack growth factor "C" indicates how much faster (or slower) a crack will grow when subjected to a TCA GAG cycle rather than the 2.8 length hours commercial B707-320c flight.
The composite crack growth factor "C" developed for each structural detail is assumed to be applicable to all locations included in the particular airframe component common to the analysis detail.

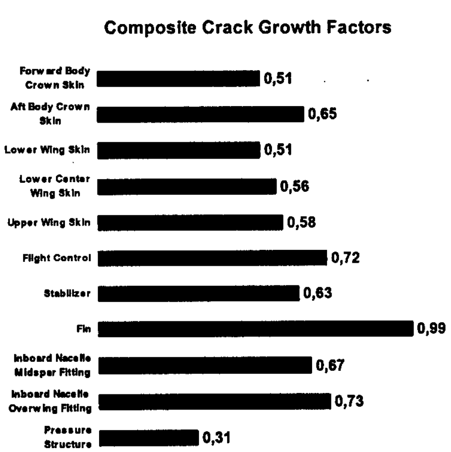


Figure 12

The reciprocal of the composite crack growth factor "C" (Fig. 12) is used to convert recommended maintenance schedule inspections and modifications from commercial flights to TCA GAG cycles.

5. CONSOLIDATED INTEGRATED MAINTENANCE PROGRAM

5.1. Applicability

Initially, the cycle counting study was only applicable to the adjustment of SSID intervals.
As more confidence was gained, its applicability was expanded to include the aging aircraft service bulletin structural modification and inspection programs.

After adjustment of cycle intervals i.a.w. the cycle counting study, and conversion to calendar time i.a.w. the anticipated utilisation, all structural requirements were integrated in a calendar based maintenance program using the CPCP as a basis.

5.2. Equalised Maintenance

To equalise maintenance (Fig. 13) for a fleet of three aircraft, the D-check items were distributed over the IL-check (every 3 years) and D-check (every 6 years), resulting in an S1- and S2-check at 6 years intervals, but staggered over 3 years. The 2C-check items were distributed over the C-check (every year) and 2C-check (every 2 years), resulting in an C1- and C2-check at 2 years intervals, but staggered over 1 year.

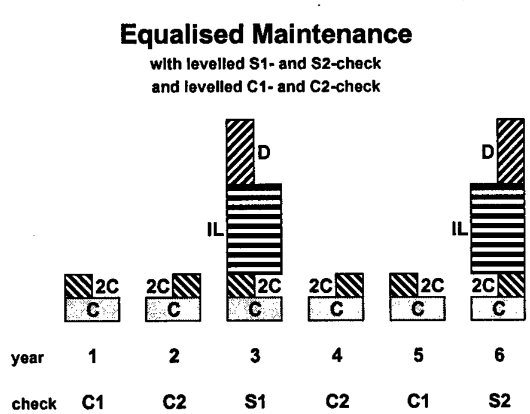


Figure 13

Heavier structural inspections with higher probability of damage findings are scheduled during higher checks with a longer turn around time, and more flexibility for unscheduled corrective action.
Major expenditures for structural modifications are only expected in the time frame 2000-2005. The modifications are mandatory per FAA AD and are to be considered as life extension. A decision not to incur these expenditures is de facto a decision not to extend operation of the fleet.

5.3 Fatigue Index

Differences in flown GAG cycles and more damaging TCA mission profiles may accumulate more fatigue damage than taken into account in the new TCA MS. The combination of both is expressed in the fatigue index (FI).
The fatigue Index is calculated yearly (Fig. 14). A FI lower than or equal to 100 % is covered by the new TCA MS.

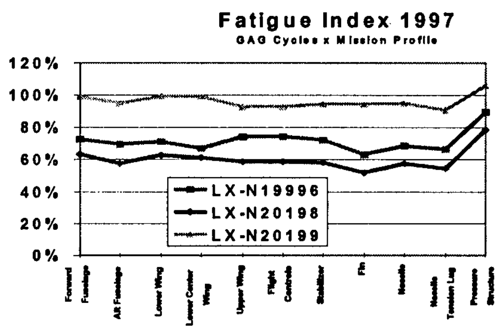


Figure 14

Cases where the FI exceeds 100 % require further assessment to define supplemental inspections to maintain the damage tolerance concept of the maintenance program.

5.4. Sustain TCA

With continued operation till 2020 the highest time TCA will have

52 years	(2.6 x DSO)
44,000 flight cycles	(2.2 x DSO)

Figure 15 shows the increase (shaded) of the fatigue DSO from the start of TCA operations for the different major structural areas, adjusted with the correction factors from the cycle counting study.

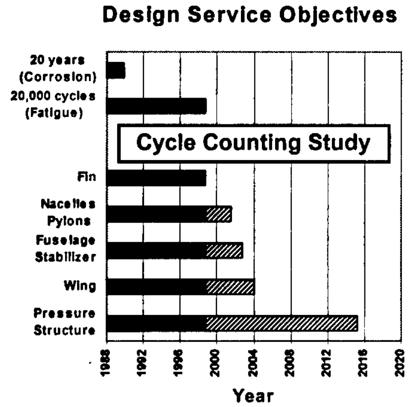


Figure 15

6. CONCLUSION

The cycle counting study provided a scientific basis to increase structural inspection intervals, postpone major structural modifications and extend the design service objectives, reducing significantly the maintenance costs, while the structural airworthiness of the TCA fleet is maintained.
A similar approach can be applied to any airframe to adjust its maintenance requirements when the current utilisation differs from the utilisation defined at the design stage.

REFERENCES

1. "Supplemental Structural Inspection Document for High Time Model 707/720 Aircraft", Boeing, D6-44860, Revision N.
2. "Aircraft Structural Integrity Plan for EC-18B Airplanes, Structural Audit and Durability and Damage Tolerance Analysis (DADTA)", Boeing, D500-11117-1.
3. "Aircraft Structural Integrity Plan for EC-18B Airplanes, Force Structures Maintenance Plan, Boeing, D500-11117-2.
4. "Structures Conferences", Boeing, 1991-1996.
5. "Cycle Counting Study", Sabena, Doc.No.321, Revision 2 dated 30 April 1995.
6. "NATO Sustain TCA Study, Structural Aspects", Sabena, Doc.No.346, Revision 1 dated 31 May 1996.

F-16 LOADS/USAGE MONITORING

D.J. Spiekhout
Structures and Materials Division
National Aerospace Laboratory NLR
P.O. Box 90502, 1006 BM Amsterdam
The Netherlands

1. SUMMARY

Load monitoring of the F-16 fleet of the RNLAF has been carried out by NLR as a routine program since 1990. At that time the old system was replaced by an electronic device capable of analysing in flight the signal of a strain gage bridge. In later years, updates of the hardware have been implemented in order to record also some flight and engine parameters. Furthermore, collecting of administrative data has been integrated in the routine RNLAF maintenance debriefing procedures.
In recent years development of a complete new load monitoring system took place. This system is fully integrated in the operational- and maintenance- procedures of the RNLAF.
Main characteristics are an increase of the number of strain gage bridges to five and a fleet wide implementation. Besides flight parameters, engine and avionics parameters are being measured. Ground stations for data handling are located at the squadrons and at NLR. By using up to date data base management programs, results are presented to the RNLAF on a weekly basis.

2. INTRODUCTION

Since 1990, load monitoring of the airframe of the F-16 fleet of the Royal Netherlands Air Force, RNLAF, is carried out by the Load and Usage Monitoring group of the National Aerospace Laboratory, NLR. The main sensor for the fatigue experience of the aircraft is a strain gage bridge at one of the main bulkheads in the fuselage The measured strain is a good indicator for the wing root bending moment, see reference 1.
In 1994, an upgrade of the system to a four channel version was carried out. Besides the strain at one location also vertical acceleration, pressure altitude and calibrated airspeed are recorded. In this way more information about the usage of the aircraft in terms of flight envelope and manoeuvring becomes available, see reference 2.
In both cases, load monitoring equipment was installed only in a sample of the fleet.
In addition to the recorded "Load data" administrative information is needed like squadron, date of flight, mission type etc.
Four times per year a report is presented to the RNLAF with detailed information about the damage experience per tail number. As a damage indicator NLR uses the so called Crack Severity Index , CSI, see reference 3.
In chapter 3, an overview of the development to the present load monitoring program will be presented in more detail.
In 1993 development of a new instrumentation system started and NLR has been involved in the definition of its specification. Further, in close co-operation with the manufacturer, NLR did take part in the development and in the implementation of the system. In the new instrumentation package several functions are combined. Besides load monitoring of the airframe also engine and avionics health monitoring is carried out. In chapter 4, this new load monitoring program will be described.
Two important changes have been made with respect to the to the old load monitoring program for the airframe, namely an extension of the number of recorded strain gage bridges to five and fleet wide application. Special attention has been given to a more automated storage, analysis and reporting of the recorded and the collected administrative data.
In chapter 5 the new hardware which will be used in the F-16 fleet, namely the FACE system manufactured by RADA electronic industries inc. in Israel, will be described in more detail.
The paper ends with some concluding remarks.

3. OVERVIEW DEVELOPMENT LOAD MONITORING F-16 IN RNLAF

In references 2 and 3 it is described how the load monitoring of the F-16 fleet of the RNLAF has been taken care of since the introduction of the aircraft in 1979. It may be mentioned here that for the RNLAF this was the first weapon system which had a load monitoring system available from the very start. Till that time, load monitoring has always been retrofitted to the aircraft after so many years. So, in the case of the F-16 this was really a step forward.

3.1 Load data

The chosen instrumentation for the F-16, namely a Flight Loads Recorder, FLR, in a sample of the fleet for Load Environment Spectrum Survey, LESS, recordings and a

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

Mechanical Strain Recorder, MSR, from Leigh for individual aeroplane tracking was not very reliable. In combination with the long turn around times, the cassettes of both systems had to be read out by the USAF, this load monitoring program was not a success.
After an update of the Fleet Structural Maintenance Plan, FSMP, which was based on data recorded with the FLR, the FLR had been build out. Further, the RNLAF was looking for a replacement of the MSR. It was decided that the MSR was to be replaced, on a sample of the fleet, by a full strain gage bridge at the same location as the MSR. Also a choice has been made for an electronic device capable of searching peaks and troughs in a strain gage signal. Selected was the Spectrapot system produced by Spectralab in Switzerland. In 1994, the one channel version of the Spectrapot system was replaced by a 4 channel version in order to have a limited LESS capability. In the case of the RNLAF, this was recording of vertical acceleration or engine RPM, speed and altitude. Per squadron there are 3 units available and 10 aircraft have "provisions for". By replacing the equipment to another aircraft in case of long time maintenance it is possible to increase the number of recorded flights to an acceptable level.
In 1995 the Spectrapot system has been made a "tracked item" in the Core Automated Maintenance System, CAMS, of the RNLAF. From CAMS it is known in which aircraft the instrumentation package actually has been installed. Furthermore the CAMS system generates automatically work orders to replace the memory after 25 flight hours.

3.2 Administrative data

In 1995 the special debriefing form, which has been in use till that moment for only the recorded flights, has been replaced by extracting the data from the RNLAF CAMS computer. This was an important change, because the special debriefing form always did request a lot of attention from NLR in order to get the forms filled in properly. Without the data on the form the recorded load data can not be used.
It has been assured that all important <u>administrative data</u>, that used to be on the debriefing form, will also be available in CAMS. This lead to some changes in the CAMS input by the debriefer of the squadron. For example mission type and external store configuration have been added. Further, the decision has been taken by the RNLAF to collect this data for <u>all F-16 flights</u>, independent of the fact if there is a strain recording available or not. Each week this information is downloaded by NLR directly from the main CAMS computer.
In figure 1 the data flow in the present program is shown.

3.3 Individual tracking

By combining the load data from the sample measuring program and the CAMS data from all F-16 flights it is possible to present to the RNLAF information about the experienced fatigue severity per tail number. From the sample measuring program the <u>severity per mission type, per squadron and per time period</u> is available. By combining this information with the mission mix per individual tail number for the same period, an individual damage indication is calculated.

In the present program the damage indication used is the <u>Crack Severity Index</u>, CSI, which for some years has been developed by NLR. This CSI is used for quantification of the fatigue damage of recorded stress spectra in terms of crack growth potential, see reference 3. The CSI is a relative figure: in the application for the F-16 a value of 1.0 means a fatigue damage according to the usage and loading environment of the RNLAF in 1985.
On of the important features of the CSI method is, that interaction effects between large and small load cycles (or between severe and mild flights) are taken into account. As a result the fatigue damage of a flight is therefore dependent on the severity of the flights which has been flown before that flight.

3.4 Reporting

Besides the quarterly standard reports which are being made on a routine bases, in recent years also ad hoc reports have been made on request of the RNLAF. Partly these questions were raised by the standard reports, but also questions were asked in relation to the future usage and loading experience of the F-16 fleet after MLU modification. The large data base offers possibilities to investigate these questions. As an example actual and estimated take off weight distributions are presented, see figure 2.

4. NEW LOAD MONITORING PROGRAM

In 1994 an updated version of the Spectrapot system has been introduced in the fleet. At that time it was already clear that this was only a temporarily solution. Since 1990 discussions took place between RNLAF and NLR about the most favourable load monitoring program for the second half of the life of the F-16 fleet.
Two main modification projects were foreseen, namely the Falcon Up project, which is mainly a structural modification of the four main carry trough bulkheads and the Mid Life Update, MLU, which is basically an avionics upgrade of the aircraft. As a result of this structural upgrade it was expected that the mid fuselage section will give far less fatigue problems in the future. Consequently, other parts of the structure will become fatigue critical.
However, the main indicator for the loading experience of the aircraft is the strain gage bridge at one of the main bulkheads. This is not the best location for monitoring the fatigue experience of other parts of the aircraft, like outer wing, fuselage and stabilisers. Monitoring of these parts was even more important, because the RNLAF did find already fatigue cracks in these other parts of the aircraft. The conclusion was: <u>an increase of the number of strain gage bridges is needed</u>.
Furthermore, one of the basic assumptions for the old "sample" program was that aircraft in the same squadron

were flying more or less the same mission mixture. During the first years of operation of the F-16 in the RNLAF this was true. However, in recent years aircraft have been switched more often from squadron to squadron. Also more "out of area operations" in Canada, Goose Bay, and Italy, Villa Franca, took place with a different usage and loading experience for the aircraft as in the Netherlands, see figure 3. For these reasons a sample program will not be sufficient any more. The decision has been taken for a fleet wide implementation.

4.1 Development of new instrumentation

In the same period, the operational directorate of the RNLAF was looking for a pilot debriefing instrumentation independent of the so far used ACMI ranges. RADA electronic industries in Israel had their Air Combat Evaluation, ACE, system. A selection of the data on the Mux Bus in the aircraft is written on the video tape. In a ground station the video tapes of different aircraft are synchronised and the data on the tapes is used for creating an artificial picture in which the fight can be followed.
In 1993, test flights with this ACE system has been carried out in the Netherlands and the idea came up to combine the new system for the pilots with the need for a new load monitoring instrumentation.
The RNLAF did contract NLR for writing down the specifications for the Fatigue part of the combined system and to codevelop with the manufacturer the new "FACE" system.
During a test flight with the new system in 1994 it became clear that it was possible indeed to combine the pilot debriefing- and the load monitoring- function in one system. In 1995 the contract was signed by RNLAF to buy FACE systems for all aircraft and in 1996 the first test flights took place with the new pre production system. During the development phase close contact has been held with Lockheed Martin Tactical Aircraft Systems, LMTAS, for implementation of the system in the F-16, especially for a correct connection of the Mux Bus, hard- and software, and the wiring for the MLU configuration.
At this moment about fifty percent of the F-16 fleet of the RNLAF has been modified.

4.2 Main features of the new load monitoring program

In chapter 3 the present load monitoring program has been described. On first sight the new load monitoring program can be seen as an upscaling of the existing program: an increase of the number of straingage bridges to five and a fleet wide implementation.
However, as a result of the increased capability of the recording system, which is fully integrated with other aircraft systems, far more (flight)parameters for airframe-, for engine- and for avionics- monitoring are available. The software is very flexible in changing the parameters which will be measured. In the next chapter this recording system will be described in more detail.
The load monitoring program, which is the total of instrumentation, measuring, analysing and reporting, has to cope with all these new possibilities. Furthermore the RNLAF wants the results far more faster than in the past.

The main features of the new load monitoring program are as follow:
* Increase of the number of strain gage bridges to five. Besides the strain gage bridge on one of the main carry through bulkheads at FS 325, strain gage bridges are located at the lower skin of the outer wing at WS 120, at the keelson in the centre fuselage at FS 374, at one of the aft fuselage bulkheads at FS 479 and at one of the attach fittings of the vertical tail at FS 462, see figure 4.
For the selection of these locations a special measuring program has been carried out with two instrumented aircraft. During 330 operational flights the signal of 10 strain gage bridges have been recorded. The location of these 10 bridges have been chosen in close co-operation with LMTAS.
In our opinion, the chosen set of five strain gage bridges is a good compromise between a limited system and sufficient information about the loading environment of different locations in the airframe, see figure 5.
* World wide experience has shown that 100 % data capture is an illusion. For example sensors break down, memories get lost and wiring problems occur. This means that there must be a "backup procedure" for replacing the lost data of the flights concerned.
In discussions with the RNLAF it was stated that the results should be made available far more faster than in the old situation in which quarterly results were presented to the squadrons a long time after the end of that quarter. Conflicting with this wish was the decision taken that a memory cassette stays in the aircraft for 25 flying hours, so it may take weeks before the actual recorded data becomes available.
To find a solution it was realised, that each week administrative usage data, like mission mixture and flight hours per tail number from the CAMS system is stored in the data base at NLR. By using this data and what is called "Statistical CSI data " it is possible to calculate each week fatigue damage results per tail number. "Statistical CSI data" means the CSI values per mission type, per squadron and per time period. This is the same procedure as has been used in the old "sample" program to calculate CSI results for not instrumented aircraft.
With the weekly update of the 5 CSI values per tail number, the RNLAF can fine tune their maintenance and show the pilots the results with respect to fatigue damage of the flights of the previous week.
After reception of the actual recorded data from an aircraft, two actions will take place: first: the so far used statistical CSI data used for that tail number will be replaced by the actual recorded data and second: the statistical CSI data for that squadron will be updated. In this way there is a continuous updating of the statistical CSI data. And of course, if recorded data is really lost, the data for replacing the actual recorded data is as accurate as possible for the average squadron experience.
* In order to make it possible to present weekly results, a large change has to be made in collecting, storing and

analysing the data in comparison with the methods used so far. One has to bear in mind that there will be a large increase in the amount of data by changing from a sample program on a limited number of aircraft with one strain gage to a fleet wide program with five strain gage bridges per aircraft. To cope with this, the whole process of data handling has to be automated to a large extend. As an example, data transfer from the squadron ground station to NLR will be done automatically during the night. Special care has been taken to ensure "secure" data communication.

For storing and analysing the recorded and CAMS data and for reporting of the results the relational data base package from ORACLE has been selected. Since last year the programming of the application for the F-16 load monitoring program is under way at NLR. For storing the data, a large number of tests are included in this application program to check on the quality of the recorded data. Cross checks with other signals are carried out. Under investigation is the use of neural networks for these checks.

Further, the different structural configurations of the airframe are taken into account. For example, for the 325 bulkhead there are three different versions, namely the "early" production, the "final" production and the "Falcon Up" modified bulkhead. The program corrects the strain gage signal for not being on a "final" production bulkhead. By the way, the status of all structural technical orders for each tail number will also be included in the weekly update of the CAMS data.

The ORACLE application program is also written in such a way that recorded data from the old Spectrapot instrumentation program can be used as well. A last remark can be made with respect to the different avionics versions of the F-16 aircraft. The ORACLE application program covers the Operational Conversion Upgrade, OCU, version of the aircraft with analogue and digital engine control, EEC and DEEC, and the new Mid Life Update, MLU, version of the aircraft. This effectively means that there are three complete different sets of Mux Bus and engine data available.

With respect to analysing the data, at this time most attention is given to the calculation of the CSI values for the five strain gage locations. Especially the determination of the reference sequences for these locations for the CSI calculation are a subject of discussion with LMTAS.

The final result will be that weekly reports will be generated automatically. These reports are than on line available for the RNLAF. In the start up phase of the program this will only be a limited version. It is expected that, in the future, more results will be reported. It takes some time for RNLAF and NLR to fully realise all the new possibilities of the new integrated recording equipment. For the engine monitoring already discussions have been started with RNLAF and Pratt and Whitney about future more detailed recordings. Furthermore, information about the systems in the aircraft is available. At this time it is foreseen that more often than in the past ad hoc measuring programs will be made in assistance of avionics fault detection.

5. DESCRIPTION OF INSTRUMENTATION PACKAGE

The complete FACE system combines two major functions, pilot debriefing and load monitoring for airframe and engine. In the aircraft there is only one set of hardware. The hardware in the aircraft consists of two electronic boxes, namely the Flight Monitoring Unit, FMU, and the Data Recording Unit, DRU, see figure 6. Amplifiers for the strain gage bridges are located close to these bridges.

On the ground both functions are completely separated. At each squadron there are two ground stations: an Operational Debriefing Station, ODS, for the pilots and a Logistic Debriefing Station, LDS, for maintenance. The use of the system is quite different for both groups. The pilots use the system locally as a tool for operational debriefing for specific mission types directly after a flight. For load monitoring all flights have to be recorded and a Data Recording Cassette, DRC, will be read out only after 25 flight hours. The LDS is connected to the Central Logistic Debriefing Station at NLR.

As mentioned before, the FACE system is completely integrated with other aircraft systems. In combination with the software installed in the FMU, this makes the system a very powerful tool.

In figure 7 the general structure of the FACE system is given. From different sources information is available and there are a number of storage devices. In the FMU the selection is made which signals are to be recorded with what way of data reduction and on what device the results are to be stored.

* Input: all digital information from the data busses is available. For the MLU configuration this means a total of four data busses. For example, information like flight parameters, attitude, accelerations and store configuration. For the engine the system is connected to the Digital Electronic Engine Controller, DEEC. This makes all the digital data of the engine available, such as Rpm's, pressures and temperatures. Both, Mux Busses and DEEC handle a few hundreds of signals. Further the FMU can adapt 15 analogue input signals. Examples are the strain gage bridge signals. Also some discrete values are monitored. A number of analogue channels are spare for future recordings. The last input signals into the FMU are video signals in the aircraft. These signals are being used for the pilot debriefing function of the system.

* Output: there are three output devices. For maintenance the most important one is the DRC on which the data for load monitoring of airframe and engine is stored. Normally, it stays in the aircraft for 25 flight hours.

The video tapes are used for the pilot debriefing function of the system. The FACE FMU writes additional digital data from the Mux Busses on these tapes. This is used on the ODS for generating an artificial replay of the fight. Normally, the tapes are removed after each flight.

As a third storage device the Voice and Data Recorder, VADR, will be present. This device is only used for mishap investigation. Normally, it stays in the aircraft and the data is overwritten all the time.

* In the FMU a choice has to be made which signals are to be used and what the data reduction has to be for those signals. For this, an input file has to be made on the LDS, which is called "Set Up Configuration file". With user friendly software on the LDS this file can be easily generated. Next step is to upload this file to the FMU. If no new SCF file is uploaded the FMU uses the resident one.

In the FMU different "processes" can run at the same time. In total up to 15 processes can be specified. A process is defined by choosing a master signal and than adding a number of slaved signals. Per master signal a maximum of 50 slaved signals can be selected. In principle each signal, which is known at the input side of the FMU, can be chosen for master or slave. Remark: in total 200 combinations of master/slave are possible.

For each process a data reduction algorithm has to be selected. At the moment this can be a "peak and trough search", PAT, a "time at level", TAL, or a "SAMPLE" data reduction, see figure 8. The PAT algorithm searches the master signal for peaks and troughs. A specified range filter is used to filter out small cycles. In this way only cycles which are important for fatigue are stored in their actual sequence. This algorithm is used for all the strain gage bridges.

In the TAL algorithm the time of crossing of specified levels are recorded in their actual sequence. Up to 100 levels can be specified. As a result the time spent in each interval between two levels can be calculated. For example, the use of the after burner of the engine during a flight. No slaved signals are possible with this algorithm.

The SAMPLE algorithm gives the possibility of skipping a number of samples in the master signal before the next recording. Again, at the moment of recording a sample of the master signal, also the values of the slaved signals are stored.

One has to bear in mind that the available sample rate for the different signals in the aircraft is not the same. For the Mux Busses the highest sample rate is 50 Hz. For example this is the case for accelerations and roll- pitch- and yaw- rates. For other Mux Bus signals the sample rates are less. The DEEC signals are at the most sampled with 5 Hz.

For the analogue signals the highest sample rate possible is 1000 Hz and this sample rate has been chosen for the strain gage signals. It is possible to specify smaller sample rates for the analogue signals in the FMU. Also for analogue signals a choice has to be made for the filter cut off frequency and the gain from a number of predefined values.

Data reduction is further possible by selecting the flight mode: ALL, AIR or GROUND during which the recording should take place. Also a time slot during a flight can be specified or a combination of two signals with a specified range for both signals. For example: only recording if Mach is between 0.8 and 0.9 and the altitude between 500 and 1000 feet.

After making all these selections a SCF will be available for uploading into the FMU in the aircraft.

It may be clear from the above, that every F-16 aircraft in the fleet can more or less be used as a fully instrumented test aircraft. Besides for the airframe and the engine a lot of data can be made available for monitoring the health of the avionics systems.

For the routine airframe and engine monitoring the SCF as presented in figure 9 is in use at the moment. As mentioned before it is expected that there will be an increase of the signals to be recorded as a result of discussions with P and W.

After 25 flight hours the memory cassette is taken from the aircraft and down loaded on the hard disc of the LDS. Once per week this data is than sent to the CLDS at NLR for further data storage and analysis as has been described in the previous chapter.

In figure 10 an example is given of a recent ad hoc recording program. A recording like this has been very easily added to the routine SCF.

6. CONCLUDING REMARKS

In this paper an overview has been presented of the development of the load monitoring program for the F-16 fleet of the RNLAF from a simple recording device to a fully integrated system. Fully integrated into the avionics of the aircraft and fully integrated into the operational- and maintenance- procedures within the RNLAF.

The FACE system is a very flexible instrumentation tool. To a large extend, each aircraft of the fleet can be used as a test aircraft for special ad hoc recording programs.

Communication lines between RNLAF, squadrons and air staff, and NLR are very short. This has been very favourable during the development phase of the FACE system. It is expected that this will be also the case during the implementation phase and later on.

Important is the increase of the number of strain gage bridges to five and the fleet wide installation in the nearby future.

7. REFERENCES

1. Spiekhout, D.J., Load monitoring of F-16 aircraft of the RNLAF with a smart electronic device. AGARD conference proceedings CP-506, Fatigue management, Bath, United Kingdom, May 1991.

2. Spiekhout, D.J., Reduction of fatigue load experience as part of the fatigue management program for F-16 aircraft of the RNLAF. AGARD conference report 797, An assessment of fatigue and crack growth prediction techniques, Bordeaux, France, September 1993.

3. Jonge, De, J.B., The crack severity index of monitored load spectra, AGARD conference report 797, An assessment. Of fatigue and crack growth prediction techniques, Bordeaux, France, September 1993.

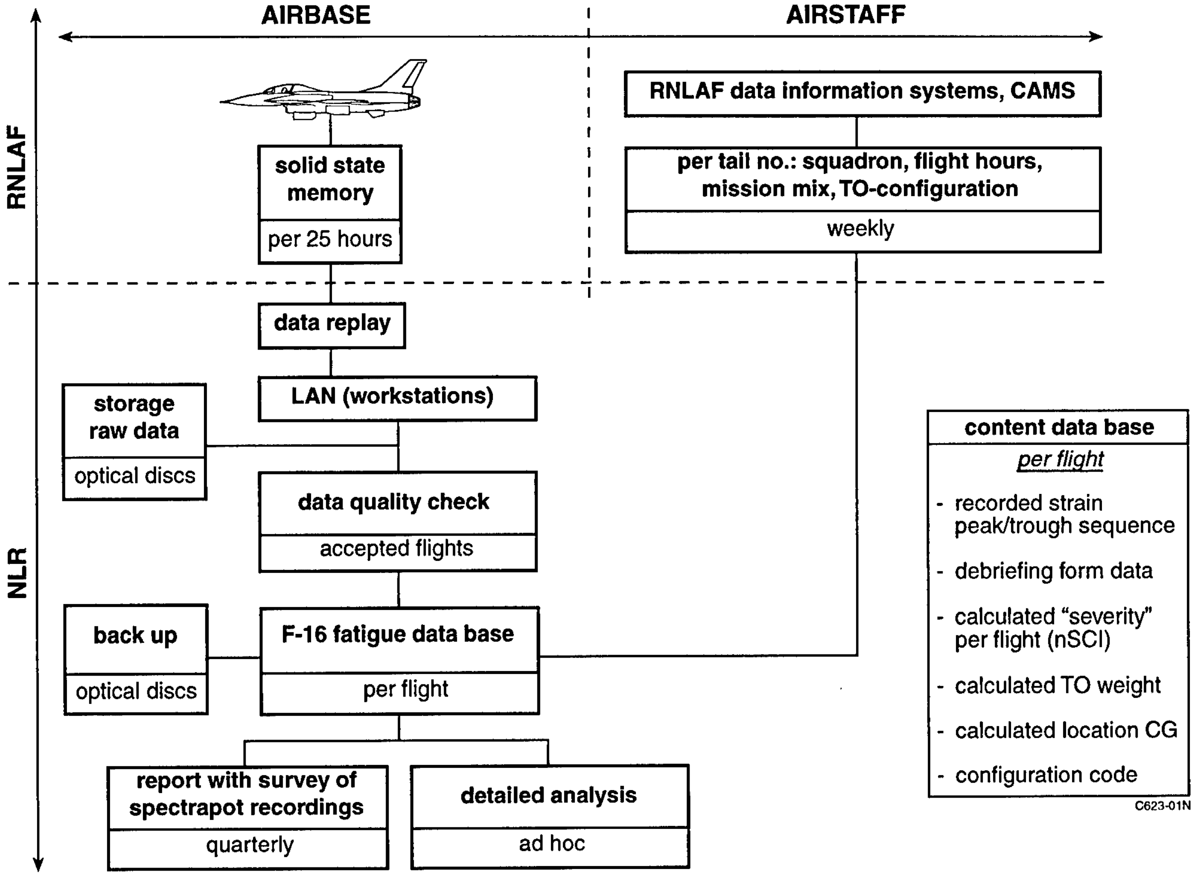


Fig. 1 Data flow in F-16 load monitoring program

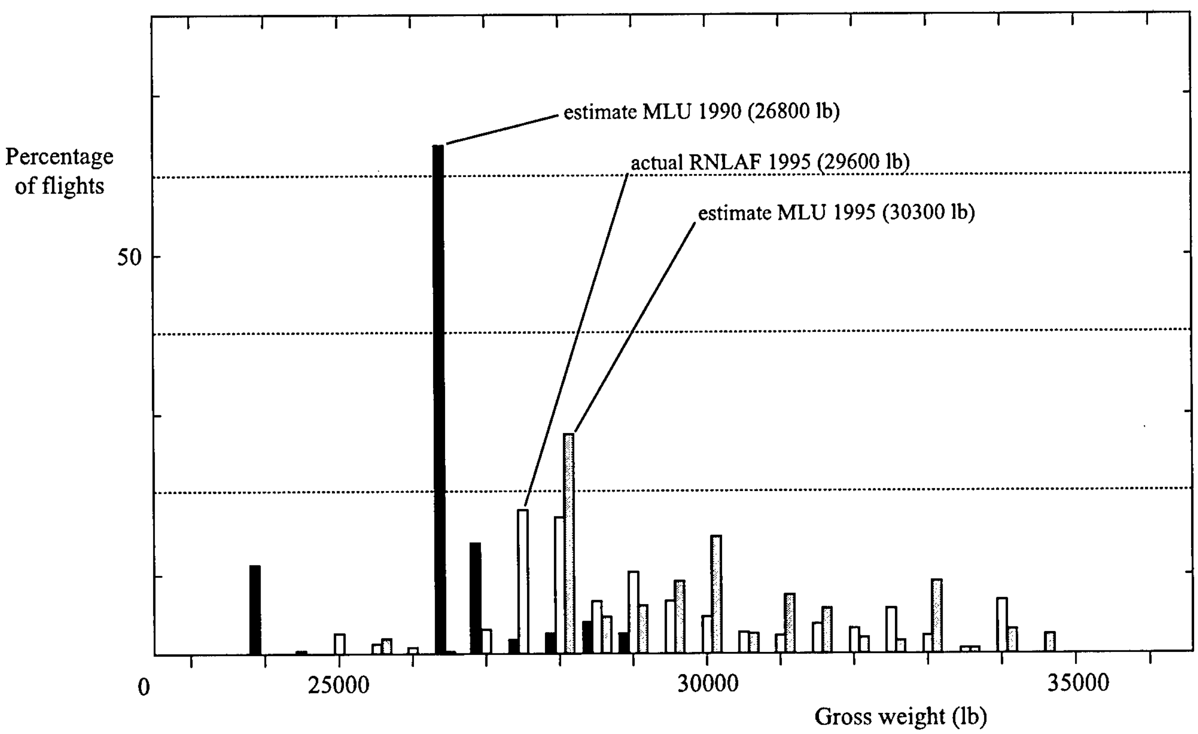
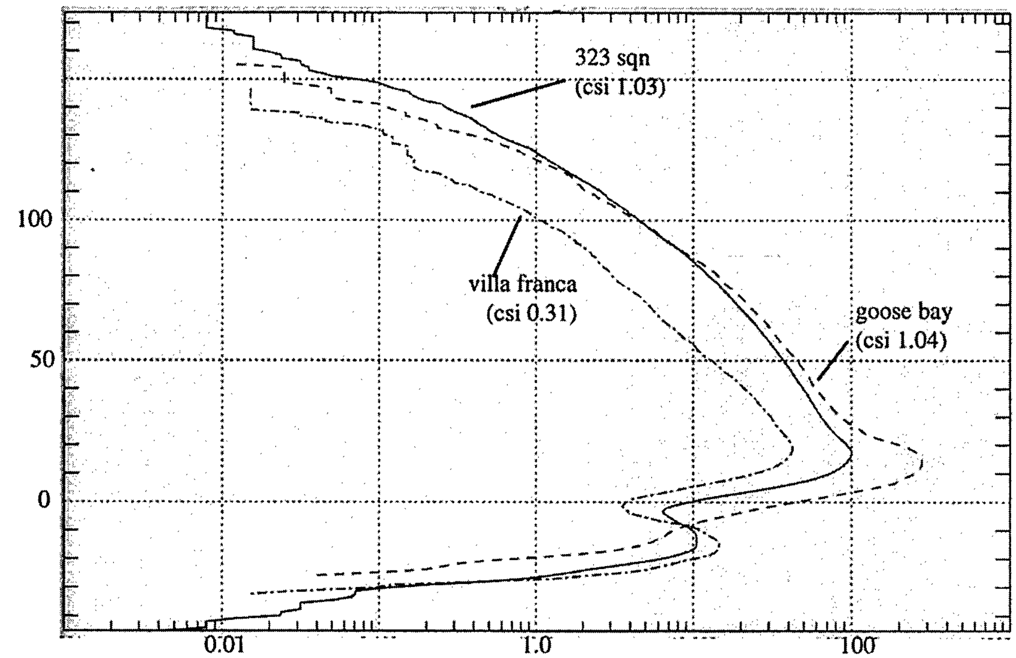
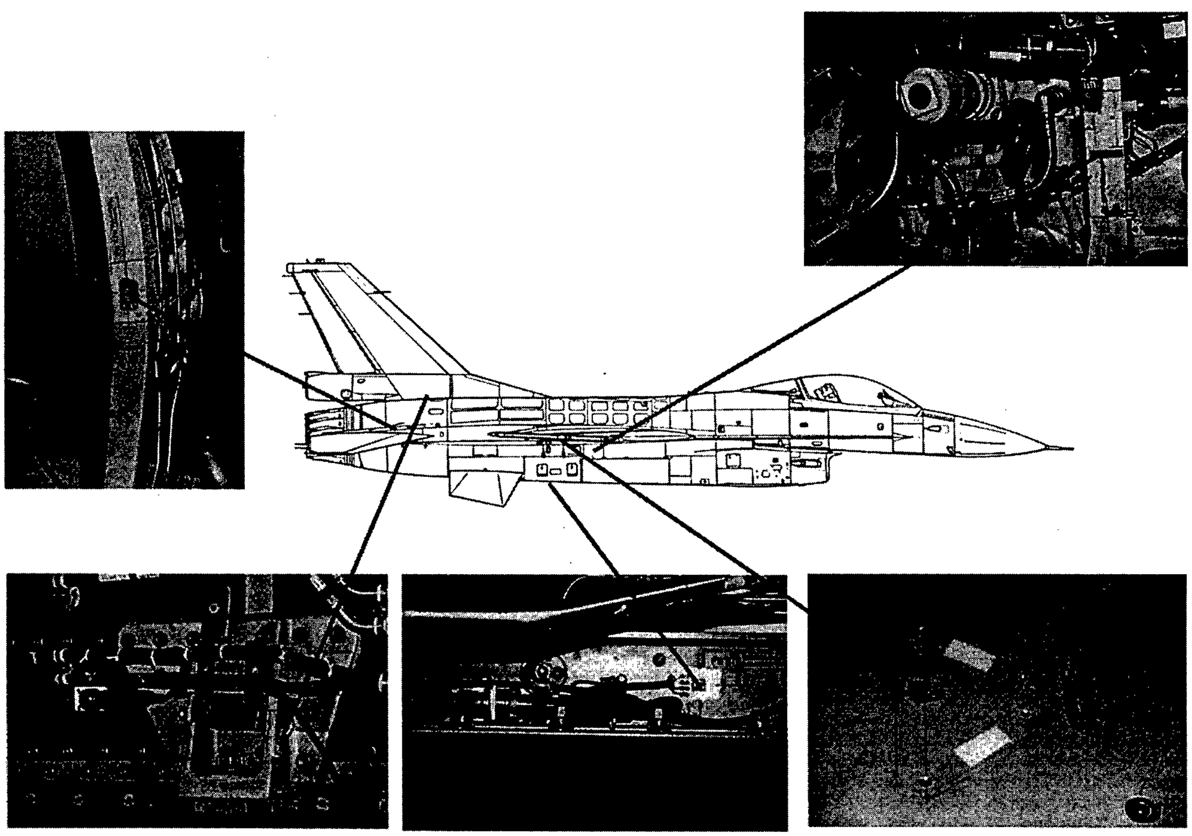
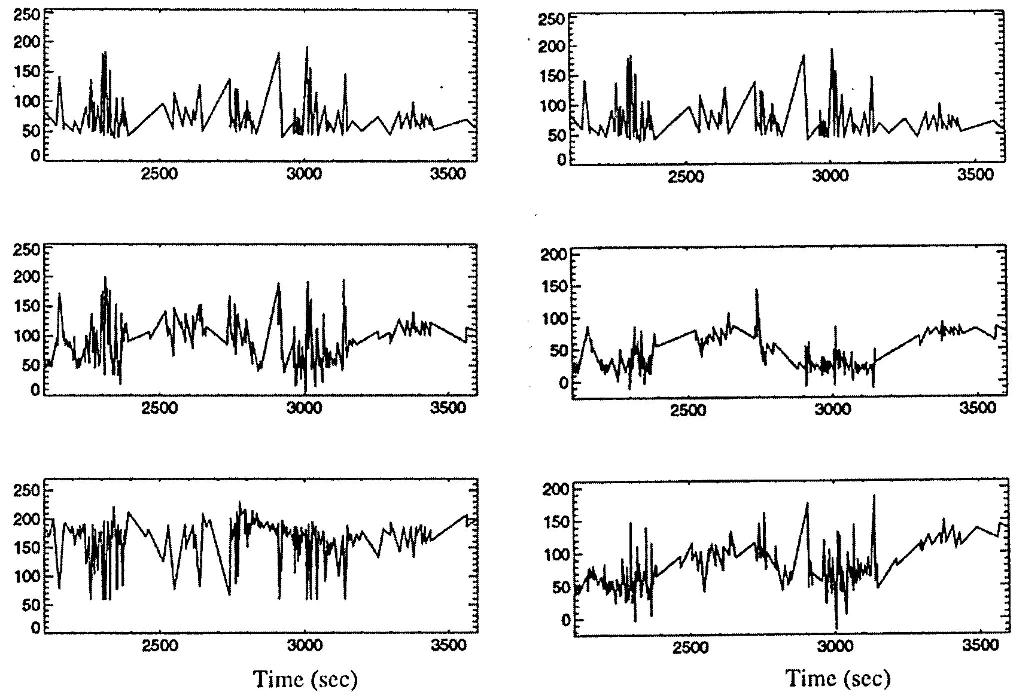
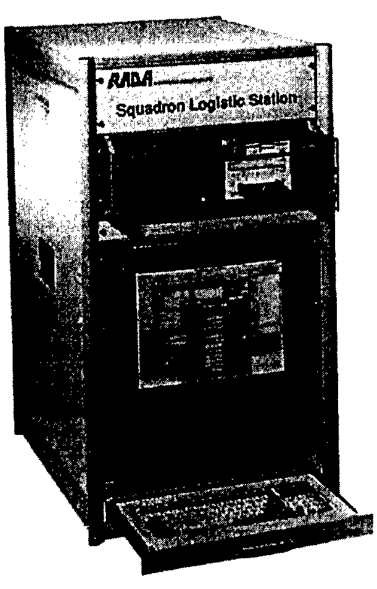


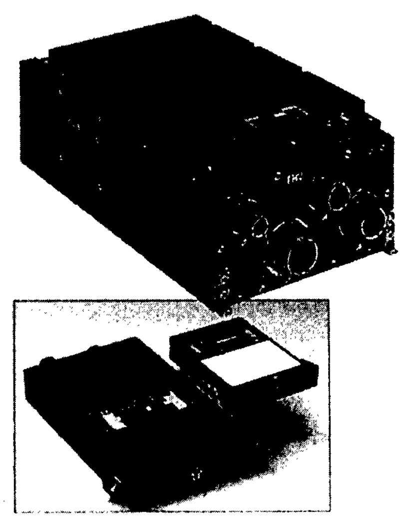
Fig. 2 Example Take Off weight distributions


323 sqn
(csi 1.03)
villa franca
(csi 0.31)
goose bay
(csi 1.04)
100
50
0
0.01
1.0
100




250
200
150
100
50
0
2500
3000
3500
Time (sec)
Time (sec)


RADA
Squadron Logistic Station



DRU/DRC

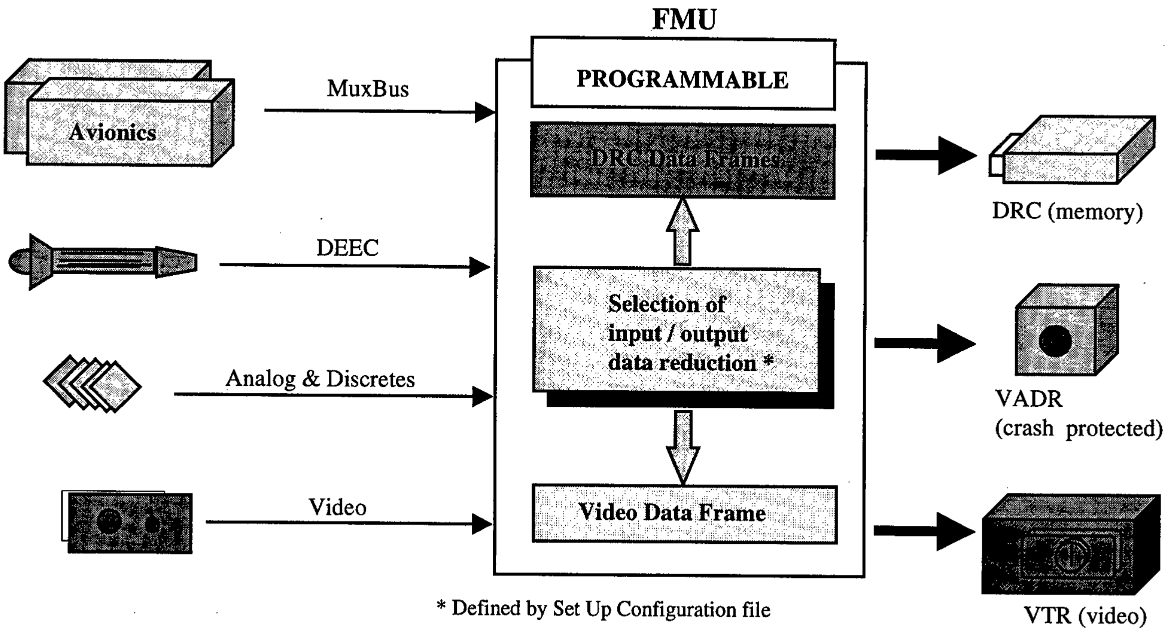


Fig. 7 Overview FACE system structure

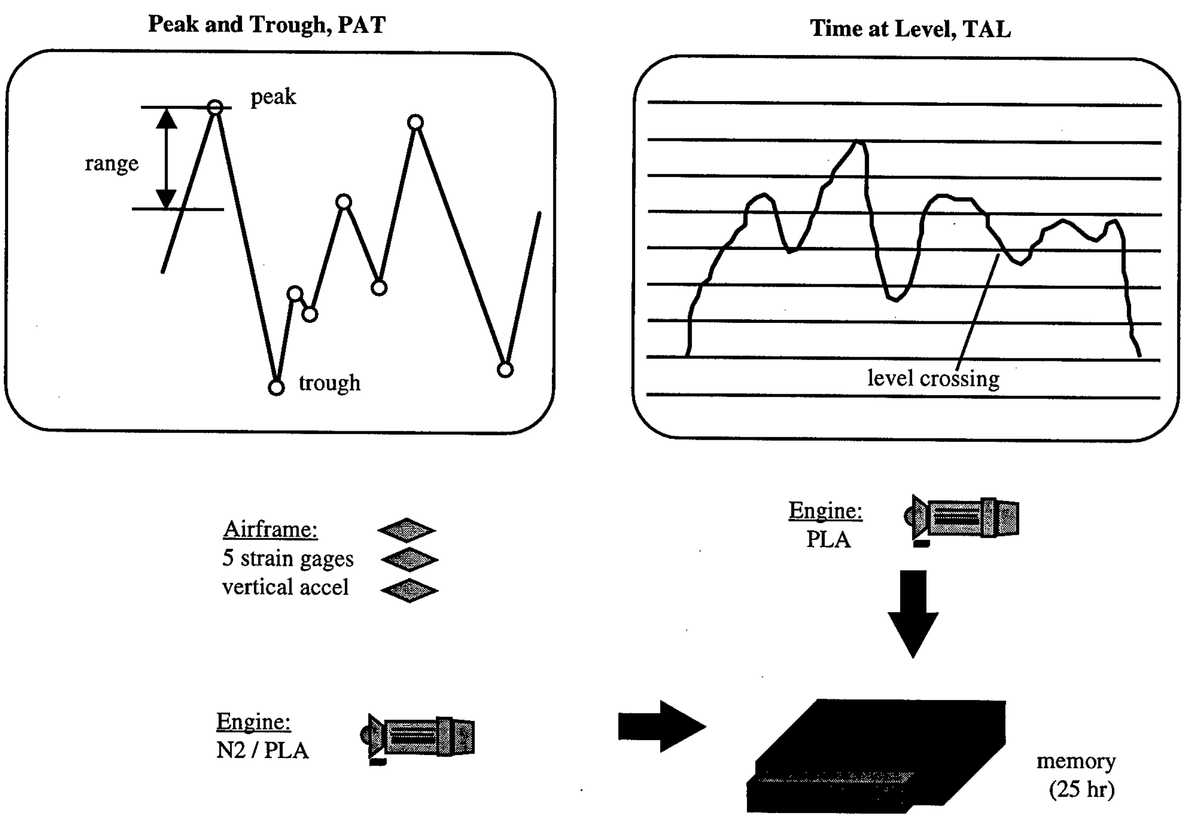


Fig. 8 Data reduction algorithms

Airframe

Process no	Master	Algorithm	Slave
1	Nn	PAT	AOA
2	FS325	PAT	-
3	FS374	PAT	-
4	FS462	PAT	-
5	FS469	PAT	-
6	BL120	PAT	-

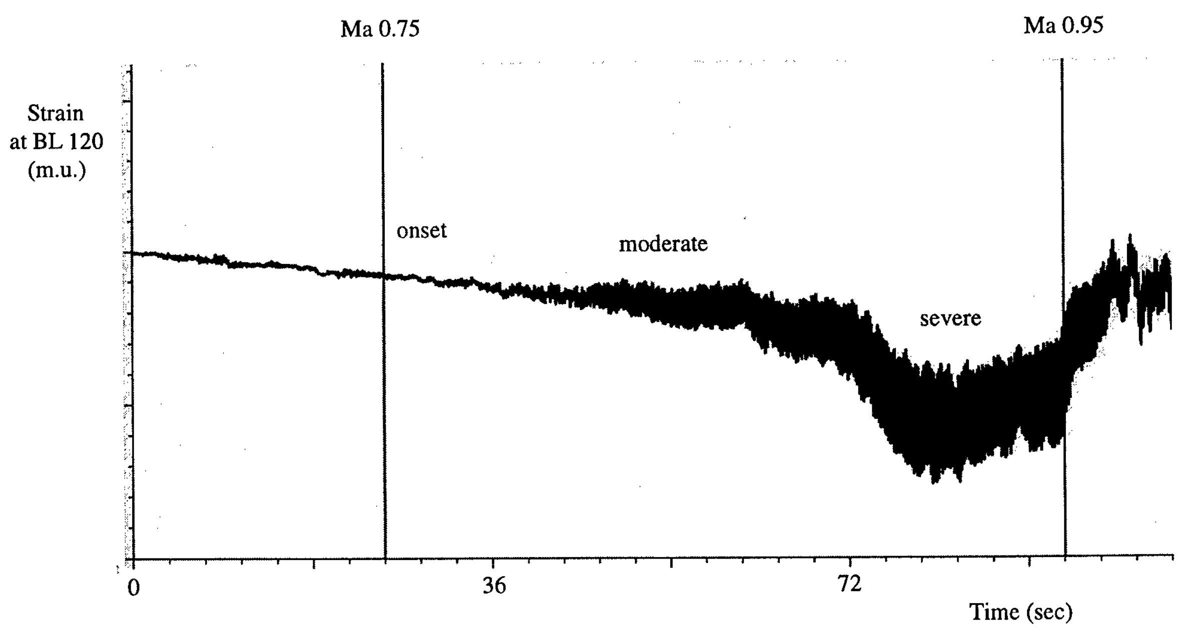
General

Process no	Master	Algorithm	Slave
1	Nn	Sample 0.2 Hz.	Nz,Ny,Nx,FS325,FS374,FS462,FS479,BL120, Roll,Pitch,Yaw,Roll rate,Pitch rate, Yaw rate, AOA, Mach,CAS, Fq,Ph,N2,PLA, platform azimuth

Engine

Process no	Master	Algorithm	Slave
1	N2	PAT	-
2	PLA	PAT	-
3	PLA	TAL	-

Fig. 9 Default "Set up Configuration file" for OCU version



Aircraft configuration:
2* Aim9, 2*370 gallon, 2* MK84

Fig. 10 Example of Limit Cycle Oscillation recording

CC130 DATA ANALYSIS SYSTEM FOR OLM/IAT

Captain A.M. van den Hoeven
Directorate of Technical Airworthiness
National Defence Headquarters, Bldg 155, Rockliffe
Ottawa, Ontario K1A 0K4, CANADA

Abstract

The Canadian CC130 fleet was fitted with a Operational Loads Monitoring and Individual Aircraft Tracking (OLM/IAT) system to assess usage severity. This article details the project that assembled the system to analyze the data and interpret the trends of aircraft structural fatigue. The CC130 Data Analysis System (DAS) was developed to process and analyze both parametric and strain data and assess usage severity using a fracture mechanics basis.

The DAS development included innovative ideas on processing complex aircraft loads data by using an "engineer in the loop" process. Quick look analysis was also enabled by pre-processing validation data and plotting all parameters. The transfer functions were developed using a finite element model and flight test calibration data. The fracture mechanics models were validated through a comprehensive coupon testing program. Fleet management tools were developed to allow component based tracking of inspection requirements, usage severity trending and prediction of future aircraft role assignments.

The DAS system has provided a sound engineering solution to the problem of assessing operational loads data for the CC130. This paper concludes with some of the lessons learned in achieving a solution to integrating a new manner of processing OLM data.

Introduction

The Canadian Forces CC130 are currently the highest time military fleet of Lockheed C-130 aircraft in operation. As such, they have been carefully managed through an active Aircraft Structural Integrity Program (ASIP) to ensure that they continue to provide reliable, airworthy service. A key element of this program has been the establishment of a fleetwide OLM/IAT program. The OLM/IAT system has been in operation since 1996 and has enabled the close scrutiny of current operational use of all airframes to ensure continued airworthiness. It has also established the possibility of optimized maintenance by tailoring inspection requirements of individual aircraft at the component level.

To take full advantage of this capability, a robust analysis solution was required that would allow a fundamental understanding of the structural loading severity of current operations, throughout the various components of the aircraft. The CC130 Data Analysis System (DAS) was an initiative undertaken by the Canadian Forces, beginning in 1996, to develop a fracture mechanics based methodology to assess the severity of individual aircraft usage from data measured by the OLM/IAT system.

The Canadian Forces manage the CC130 fleet under a Damage Tolerance philosophy, with a Durability and Damage Tolerance Analysis (DADTA) having been carried out on all of the major components during in the nineteen eighties. The DAS was designed to use the parametric data captured by the OLM/IAT system to assess the severity of usage by providing analytical stress-time histories at key tracking locations through the use of transfer functions. These transfer functions would be continuously validated by measured stresses derived from the strain sensors on the OLM/IAT sub-fleet. In this manner, the CC130 DAS straddles the methodologies of a parameter-based approach and a direct measurement

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

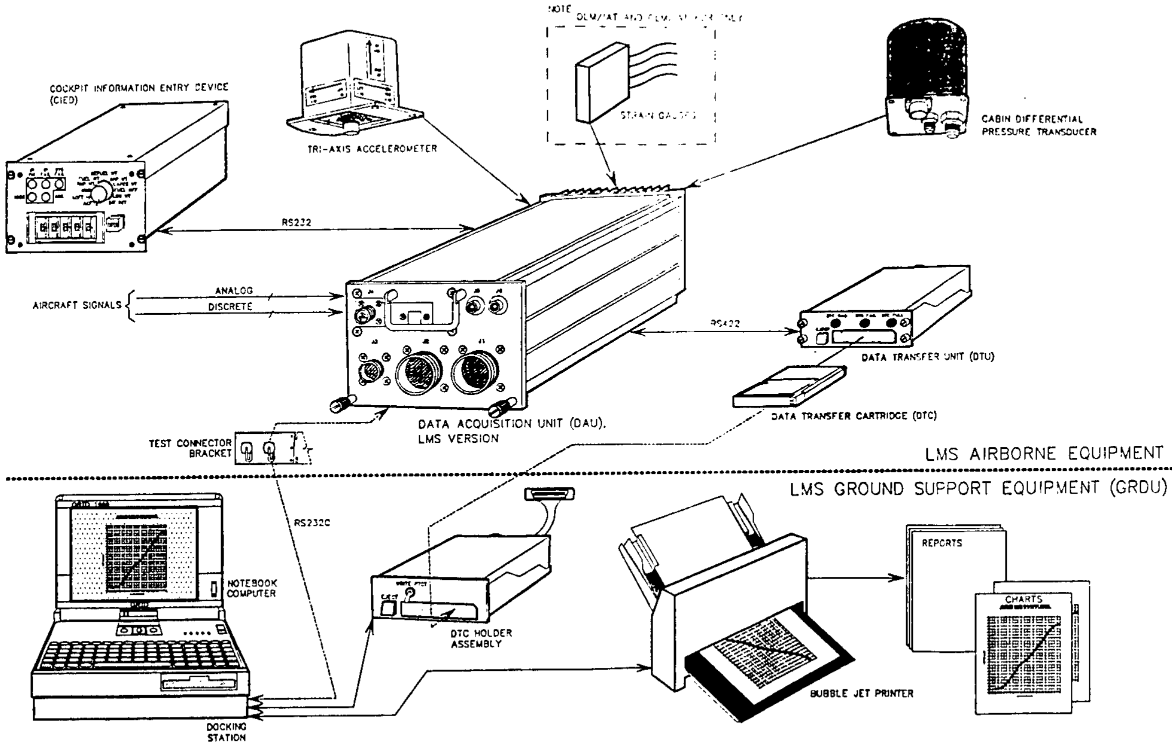


Figure 14.1 OLM/IAT System Components

approach, with one third of the fleet equipped with strain sensors.

OLM/IAT System

The CC130 OLM/IAT systems were designed and manufactured by Canadian Marconi Company in 1994, and subsequently installed in all 30 aircraft of the fleet. The systems consist of a Data Acquisition Unit, aircraft and dedicated sensors and a Data Transfer Cartridge (DTC), as depicted in figure 14.1. The same hardware is used throughout the fleet, however there are two variants; namely the OLM/IAT and the IAT. The OLM/IAT aircraft have five strain sensors located in the wing, upper fuselage, and empennage, in addition to the eight data channels present on the IAT variants, schematically represented in figure 14.2. The DTC are replaced when an 80% full indication is displayed to the aircrew or ground maintenance personnel, roughly equating to one month (or 100 hours) of flying time. Once removed from the aircraft, the cartridges are downloaded and pre-processed at each of the three operating bases and subsequently sent to a central Data Analysis Center.

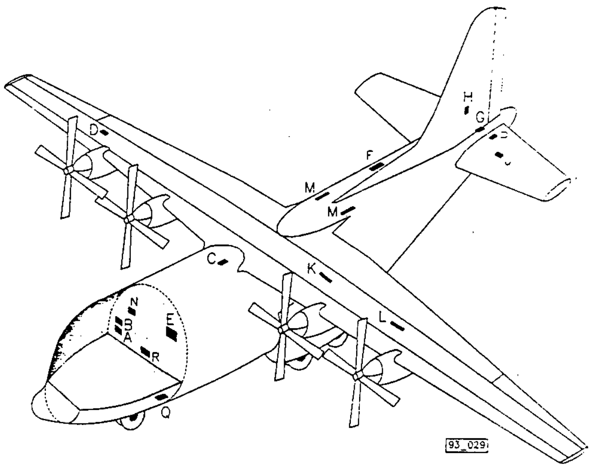
Data Analysis System

The Data Analysis System is the engineering solution to the processing of CC130 OLM/IAT data. It represents the culmination of efforts in a number of areas to fully comprehend and interpret the usage data and provide a severity factor to be applied to each individual mission carried out. It is manifested by a software program, which processes and analyzes each of the CC130 flights and allows for fleet management interpretation. The program and analysis are divided into six main modules, namely the:

a. File Processing Module,

b. Transfer Function (Loads) Module,

c. Fracture Mechanics Module,



LMS Airborne Equipment	CMC Part Number	Location On Illustration	Location On Aircraft
DAU	100-602269-000 100-602273-000	E	STN 245
DTU	100-602286-000	A	Navigator's Console
DTC	100-602326-000	A	In DTU at Navigator's Console
CIED	100-602280-000	B	Navigator's Console
CSMU	244-990140-384	F	FS 927.0
Strain Gauge (x2), Horizontal Stabilizer, LH Side	918-990139-950	J	HSS 80.0
Strain Gauge (x2), Vertical Stabilizer, LH Side	918-990139-950	H	FS 1041.5
Strain Gauge (X2, 2 sets), Upper Centre Fuselage	918-990139-950	M	FS 670.0
Strain Gauge (X2), Rear Beam at LH Wing	918-990139-950	K	WS 95.0
Strain Gauge (x2), LH Wing	918-990139-950	L	WS 357.0
Tri-Axis Accelerometer	320-990139-035	C	WS 0.00 to 10.00 R
Cabin Differential Pressure Transducer	320-990139-213	R	FS 182.0
Rudder Position Sensor	320-990139-779	G	TBD
Aileron Position Sensor	320-990139-778	D	OWS 125.982
Elevator Position Sensor	320-990139-778	P	FS 1115.13
Transformer	322-990116-244	Q	Flight Director Junction Box
GRDU Test Connector Bracket	–	N	STN 245

Figure 14.2 Aircraft Equipment Installation

d. Usage Severity Module,

e. Individual Aircraft Tracking Module, and

f. Fleet Management Module.

File Processing Module
The efficient processing of OLM data is one of the key components to a successful program to ensure that timely data analysis and interpretation can be achieved. While desirable, excessive automation of this processing can result in erroneous data interpretation, especially in the initial development of a new analysis solution. In the DAS, a semi-automated approach was used with an "engineer in the loop" strategy that allowed for full automation but also enabled detailed interpretation by an experienced engineer. This capability was considered necessary at the start of the program to build confidence in the new data systems and was later retained as a permanent feature to allow for continuous validation and ad-hoc analysis of individual flights.

The file processing module carries out a number of mission characterizations based on flight duration, cruise altitudes, touch and go's, as well as other events and provides a method of classifying mission sorties. It also carries out validation and error checking and provides visual cues to the maintenance technicians and engineers to allow for semi-automated processing of flight files with a high confidence level. The data collected on each flight is arranged on a main screen and validation results are displayed by changing the colour of those parameters which are outside of the prescribed ranges, or which do not conform to the logic rules built into the system. Figure 14.3 shows the layout of the file processing screen that displays all of the flight information, including indications of Built-in-Test (BIT) errors and overlimits outside of the aircraft operating instructions. These features allow the operator of the system to quickly scan a large number of

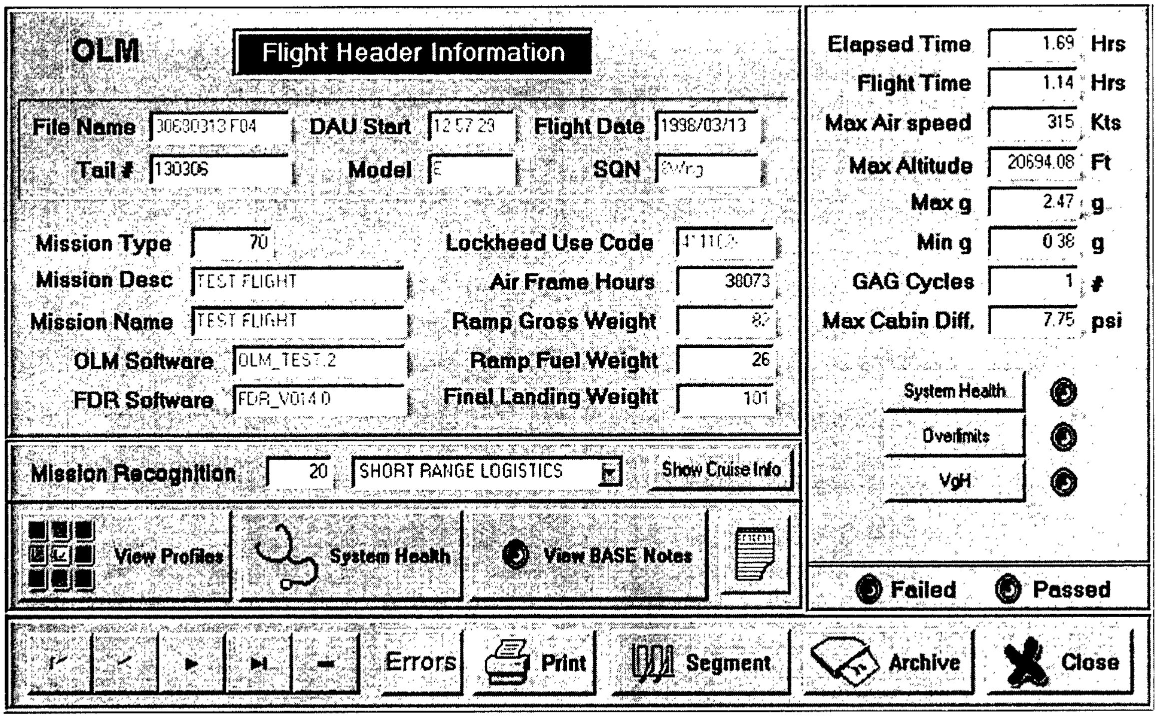


Figure 14.3 File Processing Screen

files and have any non-compliant values brought to his attention. These same criteria could be used to allow for a fully automated validation of the data, however an experienced OLM/IAT engineer brings a significantly higher level of interpretation to the process.

The processing module automatically prepares a plot of all parameters on a common time axis to enable a quick look check to be carried out on the expected response of the various sensors throughout the flight. This graphical presentation, shown in figure 14.4, allows for efficient sanity checks, which would be extremely difficult to fully automate. It also aids in identifying intermittent sensor errors and can provide a method of identifying non-standard response of a measured result by using multiple sensors to interpret the total aircraft load environment.

Transfer Function (Loads) Module

The DAS uses transfer functions to derive the local stress time histories at various tracking locations throughout the aircraft. These tracking locations have been picked as critical, representative components that are used to assess the usage severity. The transfer functions have been derived through the use of a finite element modeling effort, validated through a strain survey. With a validated FE model, including aerodynamic loads, the transfer function equations were derived to relate the parameters measured on the IAT aircraft with the local stress state in a given component (reference 1). The loads are collected into a sequential peak valley spectrum, which is subsequently prepared

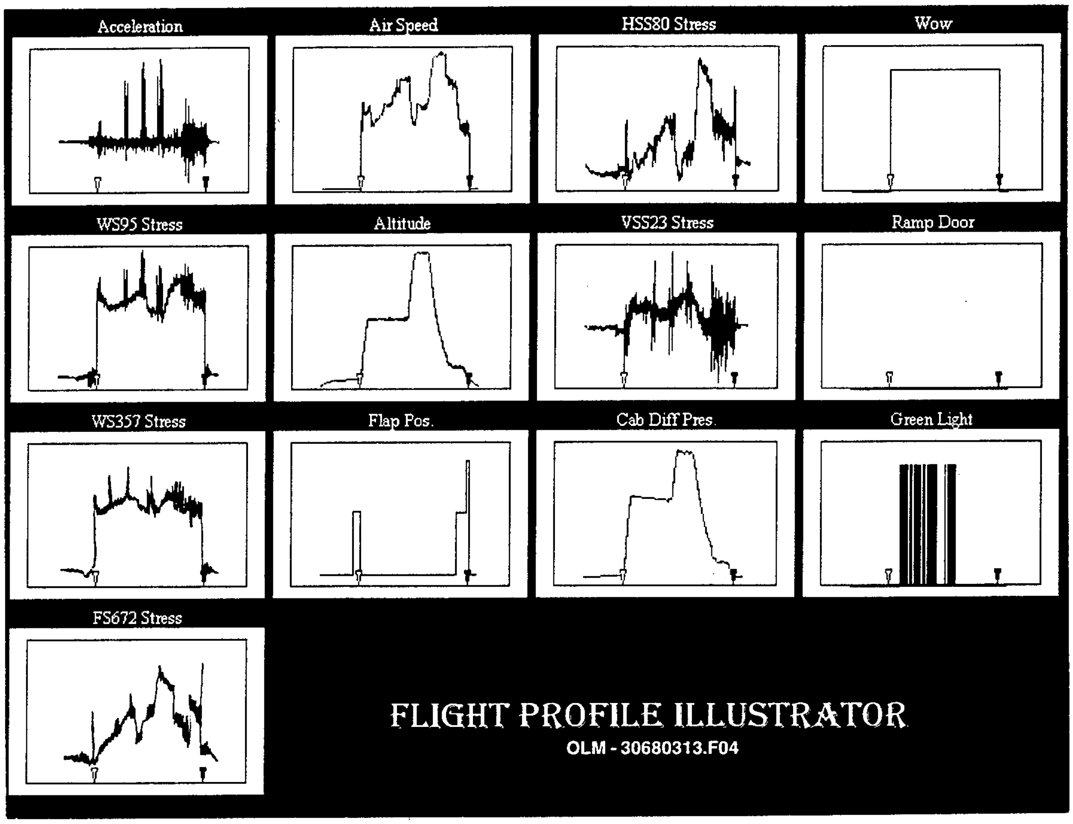


Figure 14.4 Flight Profile Illustrator Screen

for the Fracture Mechanics Module interpretation. Currently, dynamic aeroelastic effects are being investigated to validate the basic correction factors used in the wing transfer function equations.

Fracture Mechanics Module

The CC130 fracture mechanics basis was derived from the Durability and Damage Tolerance Analysis carried out in the early eighties. This module takes the spectra developed for an individual mission, group of missions or individual aircraft history and calculates a crack growth interval, which is compared to the baseline interval in order to assess severity. The crack growth models were developed and tested using representative coupons, which were used to calibrate the fracture mechanics parameters. The Quality Engineering Test Establishment and the Royal Military College of Canada have carried out much of the validation work, references 2 through 5. A validation of the methodology has progressed from baseline testing through to a variety of spectra, which represent specific aircraft histories and individual mission types. The resulting fracture mechanics models have been found to be very robust and are able to predict CC130 crack propagation with reasonable precision. Since the results are used to make relative severity assessments, the overall confidence level in this approach is very high.

Usage Severity Module

The measured spectra from the OLM/IAT aircraft were used to derive the equivalent baseline spectrum, comprised of a mission mix representative of the spectrum utilized for the DADTA. The IAT measured spectra were subsequently compared to the baseline to derive a severity factor that was truly reflective of the spectrum effect since the fracture mechanics basis was identical. This severity is currently assembled to reflect an individual aircraft's exact mission mix, however it was also organized into standardized mission types. The standardized mission types are used as fill-in data where OLM/IAT data has been lost or corrupted. It also allows for forward projections of various mission mixes of the current fleet. A key difference in the current DAS process centers on the definition of standard missions. Past attempts have labeled specific missions by rigid definitions of sortie profile, duration and typical manoeuvers. The DAS uses a mission recognition strategy that identifies distinguishing features of each mission and scores them in a neural network fashion to automatically deduce the mission type. This also allows the mission type to transform itself over the course of time, for example the number of touch and go's carried out during proficiency training may change from 5 to 8, resulting in a change of definition of the training mission itself. This continuous updating allows for the fill-in strategies to remain valid and ensures that the most current understanding of the missions is used. It also allows for more complex classification of missions beyond the seven standard mission types used for the CC130. The flight is broken down into segments defined by the altitude, airspeed and other relevant events measured by the IAT system, in order to be evaluated.

The usage severity is used directly to assess the equivalent baseline hours of the individual aircraft histories. It is also used for predictions of future fleet trends in order to determine upcoming inspection intervals for individual aircraft. This module also gathers and assigns a severity factor to the various types of missions to enable fleet management decisions, including mission mixes and aircraft role assignment.

Individual Aircraft Tracking Module

One of the main objectives of the DAS is to allow for optimized maintenance of individual aircraft. To achieve this goal, the tracking of individual aircraft usage must also allow for the incorporation of component level tracking and even individual part level tracking. The DAS was designed to provide usage severity analysis for the various components in the CC130. In order to achieve a workable solution, the aircraft was divided into five main areas, namely;

a. the Center Wing,

b. the Outer Wing,

c. the Fuselage,

d. the Vertical Stabilizer, and

e. the Horizontal Stabilizer.

An OLM strain gauge has been mounted in each of the five components to provide ongoing validation of the parametric calculations of strain. Within each of these zones, there are additional critical locations of interest that have been established. The relationship between these additional locations and the tracking location is currently being investigated in the ongoing development. A recent center wing strain survey has been completed and the data is being used to establish the best methodology of grouping the structural components and determining their correlation with measured strain at the tracking location.

To truly take optimal advantage of the usage severity measurements, one of the prime requirements is a maintenance tracking software that enables task and configuration management tracking. This interaction must be achieved within the larger weapon system management solutions, and cannot be carried out in isolation. The final implementation of this aspect of the CC130 program has not been completed however significant progress to this end has been made. There are emerging "Commercial Off the Shelf" (COTS) software applications which are being considered for solving the CC130 and other fleet requirements.

Fleet Management Module

The fleet management module of the system allows the weapon system manager to bring together all the elements of the OLM/IAT program and present useful output regarding the program performance and, most importantly, the severity of flying operations in the fleet. The data collection statistics and other aspects of the OLM/IAT system performance were important to include in the fleet management module, especially in the early development of the program. They allow for confidence building in assessing the level of data capture and the transition from the previous Nz counting accelerometers.

The current outputs from the DAS include g exceedance data, which can be plotted in any combination of aircraft, squadron or fleet over any period of time chosen by the user. This method still provides some good indications of overall severity of operations. The output also includes the current results of severity assessment based on fracture mechanics, by individual aircraft for each of its critical components (i.e. outer wing lower surface). These severity factors are applied to the actual flying hours to calculate the equivalent baseline component hours consumed during the period of operations analyzed. The fleet management module produces standardized reporting on fleet usage severity for trend analysis by using the key critical tracking location in the wing.

Conclusions

The CC130 Data Analysis System for OLM/IAT has been created to address the critical function of assessing the usage severity of the world's oldest operating fleet of military C-130 aircraft. Their continued airworthiness depends on a reliable measure of the operational loading environment of an extremely variable mission mix. The Canadian Forces utilize their C-130 fleet in broad range of missions, which are distributed in a non-homogeneous manner throughout the aircraft assets.

The DAS has used innovative methods of processing the OLM/IAT data to allow for in-depth analysis of each flight in order to gain confidence in the data during the early stages of the program. The "engineer in the loop" process allows for automation to occur while allowing the experienced operator to rapidly assess the flight parameters and gain insight into the physical attributes of the flight. The analytical modules have been developed with the support of transfer function equations and fracture mechanics testing, resulting in a sound technical approach to relative severity measurement. Recent developments in aircraft maintenance tracking software will likely enable the CF to carry out fully optimized maintenance in the near future.

One of the key achievements of the DAS effort has been to establish an in-house engineering capability and understanding of the operational loads environment. This capability will allow the CF to manage the damage tolerant C-130 fleet effectively into the "aging aircraft" stage of its life.

References

1. Desrochers, C., Gvozdic, R.,"Hercules Interim Structural Tracking", Martec Report TR-97-02, Martec Halifax, NS, Canada, January 1997.
2. Jahani, M., Duquesnay, D.L.,"Fatigue Crack Growth Measurements of Al 7075-T6511 and 7075-T73511 Extrusion Specimens", Royal Military College, Kingston, Ont, Canada, September 1997
3. Douchant, A., "Validation of an Analysis Software Program for Fatigue Crack Growth Predictions for use with the CC130 Hercules Data Analysis System", QETE, Hull, Que, Canada, June 1997.

4. Douchant, A., "Phase II(A): Establishing inter-laboratory Correlation using Al 7075-T6 Plate Material for the CC130 Hercules Data Analysis System ", QETE, Hull, Que, Canada, June 1997.
5. Douchant, A., "Phase II: Generation of a Fatigue Crack Growth Life Predictions using Al 7075-T6511 and 7075-T73511 Extrusion for the CC130 Hercules Data Analysis System ", QETE, Hull, Que, Canada, January 1998.

Service Life Monitoring of the B-1B and the Impact on Flight Operations and Structural Maintenance

Anthony G. Denyer
Associate Technical Fellow
Boeing North American
MC-011-SL15, P.O. Box 3644
Seal Beach, CA 90740-7644, USA

ABSTRACT

The paper reviews the B-1B Bomber fleet monitoring program that records the service environment and tracks individual aircraft for potential damage. The entire 100 aircraft fleet was equipped with a solid state flight data recorder that monitors both flight parameters and loads. The paper will describe the onboard data collection system and outline the analytic tools used to translate the recorded data into inspection and maintenance requirements as well as assessing the economic life of the structure. The paper will compare service usage with the design specification usage showing that usage to date is significantly more severe than that for which the aircraft was designed. Due to the severity of the recorded usage data the entire structure was re-analyzed during the 1993 to 1995 time period with the results showing a reduction in the economic life and a potential need for a reduction in inspection intervals. Implemented and proposed actions to offset the impact of the current severity of flight operations are described. These include a review of the analytic methodology, continued individual aircraft monitoring, a test program to provide data in support of inspection intervals, proposed operational changes, analytical condition inspections and a revised fuel management schedule. Finally the paper will discuss the lessons learned in efficiently handling large quantities of recorded data and the importance of monitoring service usage for early detection of usage severity that will impact the structural integrity.

INTRODUCTION

B-1B Background

The B-1B Bomber is a variable geometry (swing-wing) aircraft capable of high subsonic speed while terrain following at low altitude (below 500 feet). The aircraft design mission includes low altitude penetration of enemy airspace and weapon delivery combined with long range cruise and in-flight refueling. The design limit load factors for the aircraft are 3.0g with the wings aft of 65 degrees and 2.0g with the wings forward of 65 degrees. The first B-1B aircraft entered service with the USAF in 1985 and the 100th production aircraft was completed in 1988.

In 1970 the released specifications and design requirements for the pre-production B-1A Bomber included the Aircraft Structural Integrity Program (ASIP) requirements of MIL Std 1530A (1) and the Damage Tolerance Requirements of MIL-A-83444 (2) which in turn implemented fracture mechanics technology as an integral part of the design approach. The damage tolerance approach as defined in (2) requires single load path structure to be qualified as slow crack growth while multi-load path structure with crack arrest features may be qualified as slow crack growth or as fail-safe structure.

Rockwell elected to qualify all B-1 safety of flight structure as slow crack growth assuming the presence of flaws in the most unfavorable position and computing the damage tolerance life using the principles of linear elastic fracture mechanics. The first structural inspection, as defined in MIL A-83444, is at one half the damage tolerance life. The B-1B design specification included an economic life requirement of 30 years (9,681 flight hours) and a damage tolerance life of 19,362 flight hours to negate the need for structural inspections during the design life of the aircraft. Analysis, supported by an extensive component and full-scale fatigue test program, demonstrated that the design requirements were met.

Maintaining Structural Integrity throughout the Service Life

Maintenance of structural integrity throughout the service life is accomplished by a procedure known as Force Management defined in Figure 1. All service flight operations are monitored to provide the records of flight parameters and loads. The recorded flight loads data are analyzed with a series of 4 contractor developed programs, highlighted in Figure 1, and known collectively as the Force Management Data Package.

The output of these programs is a series of recommendations that include:

1. Structural inspections to ensure safety of flight
2. Structural maintenance actions to ensure the aircraft meets the desired life or to extend that life.
3. Changes to flight operations to reduce the structural loading severity and extend the life.
4. Rotate aircraft to various operational requirements to maximize the structural life of the fleet.

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

The flow of force management process, shown in Figure 1, is continued throughout the life of the aircraft so that inspection requirements can reflect the impact of changing aircraft usage with time and any maintenance actions such as structural repairs and modifications.

This remainder of this paper is confined to describing the procedures developed to monitor and analyze the flight operations of the B-1B and to show the impact on the structural integrity and the resulting maintenance and flight operation recommendations.

DATA ACQUISITION PROVISIONS

Structural Data Collector

Collecting operational data for the B-1B is achieved using a microprocessor based solid state data collection and storage device known as the Structural Data Collector (SDC). One such instrument is installed on each of the 100 B-1B aircraft. The SDC samples, processes and records data from six, ASIP dedicated strain gages located at strategic points on the airframe, three ASIP dedicated accelerometers and 30 mission parameters extracted from non ASIP information available in the Central Integrated Test System (CITS).

The Central Integrated Test System provides on-aircraft capability for in-flight passive monitoring, data entry, fault isolation and active testing of most aircraft systems and subsystems. In support of its primary functions CITS monitors many of the parameters to be processed by the SDC recorder. This parameter monitoring capability is used to supply the SDC with parameters of interest via a dedicated high-speed serial-digital CITS/SDC link.

The CITS data entry function is used to supply the SDC with the necessary flight documentation of aircraft serial number, mission type and date and weight data, via the CITS Control and Display Panel. TABLE 1 provides the list of the significant airframe structure parameters that are monitored and recorded by the SDC and the documentary items input to the SDC by the crew.

Other Data Acquisition Requirements

The B-1B ASIP monitoring system requires data inputs from sources other than the SDC.

1) Possession and Utilization Record Summary
The USAF maintained REMIS/EIMSURS (3) system provides a summary of aircraft ownership, possession and utilization records, a sample of which is shown in TABLE 2. This includes the current flight hour status of each aircraft.
2) Aircraft Mission Records.
The USAF maintained Core Automated Maintenance System (CAMS) (4) provides a record of each mission in terms of date of mission, mission type, flight length and landings. TABLE 3 reflects a sample of this data, which provides back up information in the situation that the SDC data is unavailable.

The USAF maintained data collection systems are the responsibility of the USAF Air Logistics Command who provide a monthly record by means of computer tapes.

DATA COLLECTION AND PROCESSING PROCEDURES

Onboard Aircraft SDC Data Collection and Processing

The main function of the SDC is parameter monitoring. TABLE 1 defines the list of parameters, their source, monitored sample rate, and data compression method. The SDC software performs the five sub-tasks as follows:

1. Each parameter is sampled at the time interval defined by its sample rate shown in TABLE 1.
2. Each sampled parameter value is validated so as to protect the archival memory from erroneous data. The tests include maximum and minimum range tests, maximum rate of change tests and excessive recording of a parameter in the archival memory.
3. Each parameter is processed through one of two data compression algorithms. These algorithms significantly increase the number of flight hours of data that can be stored in the archival memory by systematically eliminating insignificant or redundant information. Parameters that are cyclic in nature, such as strain records, are compressed using a peak-valley search routine. The routine locates and saves only significant local maximas and minimas in the parameter time history. All intervening points are discarded. Smoothly varying parameters, such as altitude, are compressed by a moving window technique called time history compression. This algorithm saves a parameter value whenever its current value has changed by more than a prescribed amount from the previously saved value.
4. Each validated and compressed parameter value is stored in highly compacted variable format

storage records. Each record contains a parameter identifier, the relative time tag to its previous record, the parameter data value and the values of associated time hack parameters. The basic assumption is only to include those data that changed from the previous record.

5. The final task is to record the data in the SDC archival memory. Records are archived sequentially into Electronically Erasable Programmable Read Only Memory (EEPROM). Two techniques are used to minimize the impact of EEPROM hardware failure. First a read after write ensures that the information is stored properly. Secondly the memory is logically divided into 1K blocks. Data within blocks are formatted and stored allowing retrieval from any one block independently of all other blocks.

Data Extraction Procedures

At scheduled intervals, or when maintenance indicates that the SDC memory is filled to capacity, the SDC is removed from the aircraft and the stored flight information is extracted from the SDC memory and transferred to floppy disks by the CITS ground processor.

Ground Data Accumulation and Processing Programs

A software package, shown in Figure 2 performs the task of accumulating the flight loads data received from the field and generating computer tapes of flight loads history suitable for batch processing through the ASIP analysis system.

The software package comprises two separate computer programs. One program is the Transcription Microcomputer Program, which provides the microcomputer to main frame interface. This program was developed by the USAF at the Aircraft Structural Integrity Management Information Systems (ASIMIS) facility and was specifically tailored to the USAF hardware/software environment. The other program is the Raw Data Reduction (RADAR) program which is a main frame computer program for accumulating, validating and editing data and generating analysis data tapes. The functions and inter-relationships of these programs are described below.

Transcription Micro-computer Program

A microcomputer program copies the SDC 8-inch flexible disks as received from the field onto mainframe compatible storage media (disk files or magnetic tape). The data contained therein is copied byte by byte without reformatting onto a mainframe accessible storage device. The output file provides the input to the accumulation and validation mainframe software.

Raw Data Reduction (RADAR) Program

The RADAR program converts the as recorded SDC information into sequenced time histories of each recorded parameter in engineering units. The output of the program is the SDC USAGE TAPES suitable for batch processing through the Loads/Environment Spectra Survey (L/ESS) and Individual Aircraft Tracking (IAT) analysis programs. The program is equipped with sort routines to separate the data by aircraft and sort in date sequence, based on the dates provided in the SDC documentary data.

A 'VALIDATION' module evaluates the SDC records for validity and suitability for further processing. If key aspects are missing, clearly invalid, or inconsistent with other data, an entire flight may be declared invalid. Flights are declared invalid, for example, if the aircraft serial number identification has been omitted from all documentary records on a particular data extraction from the SDC. Another cause of invalid data are those flights for which the data is incomplete (flights appear to end in the air) due to saturation of the SDC memory or loss of communication between the SDC and CITS. Individual parameters are also evaluated and may be declared invalid. Validity checks include monitoring coincident values of various parameters such as Mach number and altitude for combinations outside the aircraft envelope. A not infrequent occurrence is 'drop out' where a parameter records an extreme value and returns to normal. These can be detected and removed. Extensive printed diagnostics allow the analyst to monitor automated validation decisions made by the program.

An 'EDIT' module is provided wherein individual parameters, flights or an entire SDC record may have validity flags reset.

Sample parameter time histories (altitude, load factor and wing stress) for a typical mission are shown in simplified plot form in Figure 3

ASIP ANALYSIS SYSTEM (FORCE STRUCTURAL MAINTENANCE PACKAGE)

The ASIP analysis system, known as the Force Structural Maintenance Package has four major components which are high-lighted in Figure 1 which also shows the inter-relationship of the package with the overall aircraft structural integrity program

1. A Loads, Environment Spectra Survey (L/ESS) program to accumulate fleet usage data as recorded by the loads data collection system
2. A Durability and Damage Tolerance Assessment (DADTA) program to establish the structural life of the force under operational conditions as defined by the L/ESS program
3. An Individual Aircraft Tracking (IAT) program to monitor each individual aircraft for damage accumulation based on the loads data collection capability
4. A Force Structural Maintenance Plan (FSMP) which defines the inspections and maintenance actions necessary to maintain structural integrity. Both the DADTA and the IAT program support the computations leading to FSMP recommended actions.

Collectively, these programs are the tools that convert the recorded flight information into structural life assessments and inspection and maintenance requirements.

Loads / Environment Spectra Survey (L/ESS)

L/ESS accumulates and validates the SDC recorded field data into a series of databases from which fleet average operational usage can be reconstructed in sufficient detail to make a structural life assessment of the fleet. The approach is to block the mission data into discrete periods or mission segments characteristic of a particular type of flying or ground taxi operation and to categorize the information into five relational databases. Figure 4 shows the flow of the L/ESS program, compiling the databases from the SDC records such that the data can be reported at 6 month intervals or used to support analysis. The time history records of the aircraft weight, wing sweep, altitude and Mach number together with documentary data are used to classify each flight profile using a pattern recognition procedure. Once the mission profile has been classified into one of 34 current types, the mission data is broken down into discrete mission segments for which the average values of major flight and geometric parameters are computed. The other databases contain load factor and the strain gage data which are separated by mission type and flight segment such as terrain following within a given weight, Mach number, altitude and wing angle band. Load factor data is further divided as being due to a gust, pilot induced maneuver or a ground (taxi) event. The L/ESS database provides detailed statistics of each of the SDC collected parameters for each squadron as well as for the fleet as a whole. From the statistics in the usage database representative flight segment by flight segment sequenced mission profiles can be reconstituted with the service flight profiles program. The operational airframe spectrum derived from the L/ESS database has been compared with design spectrum and has provided the basis for an updated durability and damage tolerance assessment.

Durability and Damage Tolerance Assessment based on Operational Records

In 1993, after collecting 10,000 flight hours of operational data with the L/ESS program, Rockwell re-analyzed the entire airframe to provide an updated assessment of the durability and damage tolerance capability of the B-1B. Continuous monitoring of the L/ESS program showed that the operational usage had stabilized after initial fluctuations as the aircraft first entered service. The mix and type of operational mission profiles are shown in TABLE 4. As shown in Figure 5 the aircraft is currently operating, throughout the mission, with a gross weight 50,000 lb. higher than in the design spectrum. This increase is due to increased fuel reserves. The other major difference between the design and the service conditions is the higher than anticipated load factors due to pilot induced maneuvers. Figures 6 and 7 show, in exceedance curve format, load factor data for both wings forward and wings aft situations. The pattern flying once per flight loads, for example averages 1.85g in service as compared to 1.5g in the design spectrum. Figure 8 reflects the effect of both increased load factors and gross weight on the stress in lower cover of the wing carry through structure.

Both the durability and damage tolerance analyses used the fracture mechanics approach, the only difference being the selection of initial and final flaw sizes and the criteria of requiring one lifetime of durability and two lifetimes of damage tolerance to preclude any major inspections during the service life. This approach was selected although many components such as the wing pivot lug plates and the wing carry through lower wing skin are multi-load path. The basis for the analysis was the recorded usage shown in Table 4 repeated 30 times to represent one lifetime of operational usage. The stress spectrum at each analysis location was compiled using an automated spectrum generation program which combines the mission profiles representing each mission and the recorded load factors for each mission segment with the output of a finite element model of the structure. Typical initial flaw size for durability was 0.01 inches defined as the equivalent initial flaw size (EIFS) such that the crack growth analysis commencing at this size represents the life of a nominally unflawed part. The final allowable crack size for durability was the minimum of: the crack length that would cause functional impairment (such as fuel leaks), a crack that would be uneconomic to repair; or a crack of a size that would exhibit rapid unstable crack growth if the part be subjected to design limit load. As defined in MIL A-83444, 0.05 inches was the damage tolerance starting crack size for areas of local high stress concentrations, such as bolt holes, while 0.125 inches depth was used for other locations. The final flaw size for damage tolerance analysis was that which would exhibit rapid unstable crack growth if the part were subjected to design limit load. When compared to the design analysis the updated DADT assessment reflected, as expected, a significant reduction in the economic life, although it continues to be beyond the design life requirement, and increased inspection requirements if the current usage is maintained.

Individual Aircraft Tracking (IAT) Program

The IAT program has four (4) purposes: -

a) To provide, for each aircraft, the rate at which the fatigue capability is being used and an estimate of the remaining life.
b) To provide usage statistics for each aircraft, for each squadron, and force wide to aid USAF using command in force use planning.
c) To provide for each aircraft, the intervals at which specified critical structural locations must be inspected in order to maintain structural integrity.
d) To project the economic life of each aircraft and the force

The IAT analysis and reporting is updated every six months using a package of computer programs, shown in figure 9, developed specifically for these tasks. The input to the tracking system is the flight and maintenance records of each aircraft. Its output, supplied in periodic reports, consists of usage statistics, accumulated structural damage estimates, remaining life estimates and structural inspection intervals.

A realistic assessment of the accumulated damage on any aircraft requires that all missions be included. To ensure this goal the program first establishes the accumulated flight hours at the period end date from the AFR 65-110/REMIS/EIMSURS data, a sample of which is shown in TABLE 2. The primary source of flight data is the SDC USAGE TAPES produced by the RADAR program. Inevitably the SDC does not provide records for all flights due to the SDC being inoperative, not installed in the aircraft, a saturated SDC memory or recorded data being declared invalid. The IAT program also receives records of B-1B missions from a USAF maintained database known as the Core Automated Maintenance System (CAMS) which provides, for each flight of each aircraft, the flight length, number of landings, type of mission and the base of operation. Finally if the merging of the SDC and CAMS records results in less flight hours than the "official" (REMIS) aircraft accumulated flight hours, the "fill in" flights are defined by IAT statistical records for the last two years of appropriate squadron operation.

The IAT Program monitors each aircraft throughout the lifetime and computes damage assessments at twenty-two selected locations in the airframe and at three locations in each main landing gear structure. The locations were selected after considering, critical structural locations, the need to track each major component, representation of as much airframe primary structure as possible and computing time and cost. The stress history for some tracked locations is compiled directly from SDC recorded strain gage records while, for others a predetermined equation relates the stress to the aircraft SDC recorded load factors and the mission parameters such as gross weight, altitude, Mach number and aircraft geometry (wing angle and flap angle). The accumulated structural damage is estimated with the same approach as that used for the design analysis. For the airframe structure, linear elastic fracture mechanics provides both the durability and damage tolerance regions of the crack growth curve and for the landing gear structure, Miner's rule fatigue analysis is used. For those missions without detailed SDC data, the mission type, flight length and number of landings together with statistical key mission parameters such as take off gross weight and terrain following parameters are used to compile the damage from a pre-defined library of 120 crack growth curves representing the range of missions historically flown by the B-1B. The accumulated damage is presented in terms of ever increasing crack length for the airframe components and in terms of fatigue damage fraction for landing gear structure.

The remaining life for each airframe part is determined by commencing the crack growth analysis at the crack size defined by the service history to date and projecting crack growth for the future usage. The projected usage for any aircraft is defined as the average usage for the applicable squadron as compiled from the most recent two years records. From the IAT historic records a projected usage model is defined in terms of a mission mix and key statistics such as flight length, take off gross weight, terrain following parameters and number of touch and go landings for each mission. The crack growth is computed using an integration procedure commencing with the current crack size for both durability and damage tolerance and repeatedly applying the composite usage file until the respective limiting crack sizes are reached. Based on the key parameters of each mission a change in crack size is determined from the library of crack growth curves. The landing gear uses a parallel procedure but is based on the cumulative fatigue damage model.

The economic life an individual aircraft is defined as the sum of the accumulated flight hours to date for that aircraft and the remaining durability life while the damage tolerance life is defined by adding the remaining damage tolerance life to the aircraft accumulated flight hours.

The usage statistics accumulated by the IAT program are indexed and maintained for individual aircraft defined by serial number, by USAF squadron number and as a force wide accumulation. Damage records are indexed and maintained by structural location on appropriate individual components that are defined by title and component serial number. Structural components are defined as those items that are considered interchangeable between aircraft or from spares. Serialized components include wings panels, stabilizers, nacelles and wing sweep actuators. The non-interchangeable components, wing center box and fuselage are identified by the aircraft serial number. All reporting of usage data and damage values is by aircraft serial number. The IAT program maintains a configuration file to correlate specific component serial numbers to appropriate aircraft.

Output from the IAT Program

Inspection intervals to preserve safety are defined for each aircraft and each structural location accounting for usage severity differences in past usage and projected future usage and the rate of flight hours accumulation. The IAT program provides the damage tolerance life for selected critical structural locations. For those locations not covered by the IAT program the following procedure is followed: based on the most recent damage tolerance analysis a ratio is obtained between the damage tolerance life of the desired inspection location and the damage tolerance life of the closest tracking control point. The IAT program monitors at least one location on each major component. The control point selection is based on similar primary loading such as wing bending. This ratio is then applied to the tracking program control point damage tolerance lives for each aircraft.

Inspection intervals are defined as one half of the analytically computed damage tolerance life (the time for the inspectable flaw to propagate to critical crack length). The inspectable flaw is a crack size that can be reliably found with the appropriate Non Destructive Inspection (NDI) technique.

The IAT program also produces the remaining useful life for each component based on the durability analysis. This data is used for planning purposes, leading to recommendations as to changes to aircraft usage, rotation of aircraft within fleet operations to maximize the life of the fleet and planned life extension programs.

Force Structural Maintenance Plan

The inspection and maintenance requirements to ensure structural integrity are issued in the Force Structural Maintenance Plan. The first issue of the plan reflected the inspections and economic life defined as a result of the design analysis and the design verification test program.
Subsequent annual issues reflect the revised inspection requirements resulting from the ever changing service environment as defined by the IAT program and the updated DADT assessments conducted in 1993 and 1995. Due to the severity of the service environment the recent analysis indicates that a significant reduction may be expected in economic life of the wing and wing carry through structure and that inspections will be necessary on both the wing carry through and wing outer panel to meet the damage tolerance criteria.

The economic life of the B-1B, or the time at which a structural life extension program should be implemented, although reduced, continues to be in excess of the design goal of 30 years and under the current operations is projected to be beyond the year 2040. The critical component from durability considerations is the outboard wing at the attachments of the ribs to the lower cover. Other components have a projected economic life beyond the year 2060.

From a damage tolerance considerations the critical areas are the wing carry through at the attachment of the rear bulkhead to the lower wing planks, the wing pivot lug plates and the fuselage crown forward of the wing. The most difficult and costly inspection is the wing carry through lower cover that has Taper Lok fasteners in a stack of four titanium components. The analysis in the critical area is, however, conservative, as it does not account for the inherent high fatigue capability of the Taper-Lok fasteners. Cracks did not initiate in this area during the DVT test but analysis indicates that the test spectrum, while being more severe than the B-1B design spectrum, was significantly less severe than the current spectrum. A test program utilizing small element specimens assessed the capability of the Taper Lok joint and resulted in a postponement of inspections. Additional analytical work is underway to assess the crack growth behavior in the multi-load path lower skin of the wing carry through structure and a possible re-assessment of the inspection requirements.

Operational Changes and Other Recommendations

In addition to the FSMP inspection requirements, the following recommendations have been made to the Air Force Using Commands to reduce the rate at which the available B-1B life is being consumed:

1. Reduce the fuel reserves from the current 50,000 lb. to 30,000 lb.
2. Reduce the maneuver load factors, particularly during the pattern flying segments and during touch and go landing practice.
3. Revised fuel management to reduce fuselage fuel earlier in the mission and maintain fuel in the outboard wing longer. This will reduce wing and fuselage bending stresses for the later segments of the mission particularly the pattern flying.
4. An implementation of an Analytical Condition Inspection Program (ACI). The purpose of the ACI program is to provide a sample inspection of critical structural areas from randomly selected aircraft to detect damage, wear and corrosion not anticipated by the formal structural analysis or detected by the design development and verification tests.

The Fuel Management Center of Gravity System (FCGMS) software is currently being modified to change the fuel usage sequence.

The Using Commands have implemented procedures to reduce structural damage and these are being monitored by means of a software program utilizing the Structural Data Collector records and monthly reports issued to the Using Commands. Some reduction in usage severity has been achieved in the 12 months of operating this program.
An ACI program has been implemented.

REVIEW OF B-1B LOADS MONITORING SYSTEM

Utilizing the Central Integrated Test System to provide many of the parameters needed by the ASIP program has the advantage of reduced hardware costs and ensures timely maintenance as the CITS is a safety of flight system. The CITS data input and control panel in the aircraft crew compartment allows documentary data; aircraft tail number, flight date, mission type and aircraft accumulated flight hours to be loaded to the SDC. This is very valuable for bookkeeping purposes to ensure the data is associated with the correct aircraft and to allow correlation with the CAMS flight records when accounting for missing SDC flight records. The disadvantages include the difficulty of incorporating ASIP requested changes into CITS and the long calendar time before such changes enter service. In addition changes in CITS for non ASIP reasons may affect the ASIP program.

The large quantity of data (averaging 50,000 bytes per flight) from a 100 aircraft fleet averaging 80 flights a year requires highly automated ground analysis programs. Every value of each parameter is checked with an automated validation procedure; declared valid and so flagged, declared valid as changed based on evidence of other parameters, declared invalid and so flagged or flagged as not provided. The validation procedure is approximately ½ of the software code and consumes ½ the software running time.

Early data collection samples indicted a very large variation is recorded parameter values, many of which were patently invalid. This necessitated a long maturation time to find problems and build the validation procedures.
The primary causes of data problems requiring extensive automatic review and correction are:

1 Data dropout where a parameter records an extreme value and returns to normal. These are infrequent instantaneous parameter recordings within an otherwise normal time history and can

be removed or corrected without detriment to the overall mission records.

2 Data outside acceptable range for a given parameter or outside the range of acceptable standard deviation or having an unacceptable rate of change. An extensive acceptable range and rate of change was incorporated for all parameters based on the aircraft flight envelope plus a reasonable tolerance. Further acceptable ranges were developed for combinations of selected mission parameters such as weight, Mach number and Altitude, and for selected geometry parameters such as flap angle and wing sweep and Mach number. Some parameters are corrected and others are declared invalid.

3 The strain gages are not calibrated necessitating a flight by flight calibration based on the pre and post flight ground recordings compared to analytical assessment of the stresses at the strain gage locations for the aircraft on the ground condition. The ground stress for the wing and forward fuselage strain gage locations are a function of the aircraft gross weight.

4 The recordings of discrete (true/false) events such as landing gear position (up or down) weight on wheels (yes or no) are subject to bit jitter. With careful comparison with concurrent recordings of other parameters these records can be corrected.

5 Documentary data inconsistent with recorded data. The mission type supplied by the aircraft crew as documentary data via the CITS control panel prior to the flight is ignored if the SDC records prove a mission type input error.

6 The SDC documentary data and records are inconsistent with the CAMS records in terms of the date of the flight, mission type, flight length, number of landings and accumulated flight hours. Careful selection of tolerances (plus and minus one day on dates) and correction of obvious date input errors can solve some problems. Flight length as defined from the SDC records is from 40 knots taxi speed during the take off run to engine shut down. The CAMS records are based on a pilot report and obviously vary from this. The flight length of each mission is adjusted as necessary to maintain both the accumulated flight hours and number of missions defined by the REMIS/EIMSURS system. The sequence of missions on one extraction from an SDC is sacrosanct as is the "official" record of accumulated flight hours on each aircraft supplied by the USAF REMIS/EIMSURS system.

The usable data return from the SDC has historically been between 90% and 95% of the data recorded. The capture rate is currently at 60% of the flight hours flown down from a high of 75%. The difference between capture rate and usable data return is primarily due to flight operations without an installed SDC. Another cause of reduced capture rate is the lack of maintenance of the onboard ASIP dedicated sensors (accelerometers and strain gages).

The data turn around time from flight completion to analysis completion in support of the monthly operational severity report is 45 days with a goal of 30 days. The data turn around time has significantly reduced since the program inception due to electronic data transfer between the USAF and the contractor replacing mailed magnetic tapes.

CONCLUSIONS

The severity of the service usage as compared to the design usage clearly demonstrates the importance of fleet usage monitoring and the development of inspection and maintenance requirements to meet the demands of current and future usage. Continual monitoring is particularly important on the B-1B as the proportion of flying at various wing sweep angles, the proportion of flight time at low altitude and the wide capability in gross take off weight can not only significantly affect the life of the structure but also determine the critical component.

The importance of early warning of the usage severity cannot be over emphasized. Changes to operational procedures can be implemented to alleviate potentially serious problems before they became critical and engineering has time to perform the necessary testing and analysis to develop inspection procedures and structural modifications if necessary.

Maintenance schedules based on an IAT program allow for immediate inspections to be performed on those aircraft subjected to severe usage thus maintaining the safety of flight. The less severely used aircraft can be scheduled for maintenance over an appropriate period to minimize the impact of out of service aircraft and the work load at the maintenance facilities. In addition, the correlation of usage statistics with damage rates allows the USAF to rotate aircraft through the squadrons in order to maximize the lifetime capability of the fleet. Finally the utilization plans can be tailored to some degree to minimize the structural damage.

The B-1B is the first USAF aircraft fleet wherein each aircraft is equipped with a loads monitoring device recording a significant number of flight parameters. Individual aircraft tracking is thus achieved without the use of pilot reporting forms. A very large quantity of data is collected necessitating a highly automated program with minimal manual interaction. The SDC records, however, collected from 30 remote sensors and flight crew keyed in data is not 100% perfect. It was necessary therefore to write a sophisticated program capable of verifying, making intelligent assessments and correcting, if possible, the 'real world' data. After initial teething problems with the SDC and the SDC/CITS interface, the ground software program is operating with a high return of actual flight data. The net result is to support the B-1B Force Structural Maintenance Plan with significantly enhanced data leading to reduced maintenance costs and improved structural integrity.

Design criteria such as not accounting for the enhanced fatigue capability of Taper Lok fasteners or cold worked holes is prudent during the design phase to ensure an inherently durable and damage tolerant structure. Before embarking on an expensive inspection and modification program requiring significant aircraft down time, however, the impact of this criteria should be taken into account and the conservatism reduced if engineering analysis and structural tests so indicate.

REFERENCES

1. MIL STD 1530A Military Standard - Aircraft Structural Integrity Program Requirements, December 1979.

2. AFR 65-110 Airforce Regulation - Aerospace Vehicle and Equipment Inventory, Status and Utilization Reporting System, April 1980.

3. HAF-683-604 Core Automated Maintenance System Functional Description.

4. NA-78-491-5 Improved Methods for Predicting Spectrum Loading Effect Fifth Quarterly Report, 30 September 1979.

5. Denyer, A.G. Fracture Mechanics Technology Applied to Individual Aircraft Tracking – American Society of Testing and Materials, Proceedings of the Thirteenth National Symposium of Fracture Mechanics.

Table 1 – Mission Parameters Monitored and Recorded by SDC

Parameter	Data Comp Method	Sample Rate /sec	Data Source
Strain - Stab (Left Hand)	pv	40	ASIP
Strain - Stab Right Hand)	pv	40	ASIP
Strain - Stab Fitting Side Plate	pv	40	ASIP
Strain - Wing Sweep Actuator	pv	40	ASIP
Strain - Wing Lower Cover	pv	40	ASIP
Strain - Fwd Fusel Longeron	pv	40	ASIP
Vertical Acceleration (Nz)	pv	40	ASIP
Lateral Acceleration (Ny)	pv	20	ASIP
Long Acceleration (Nx)	pv	20	ASIP
Pitch Rate	pv	8	CITS
Yaw Rate	pv	8	CITS
Roll Rate	pv	8	CITS
Wing Sweep Position	th	1	CITS
Flap Position	th	1	CITS
Left Stabilizer Position	hack 3	20	CITS
Right Stabilizer Position	hack 3	20	CITS
Left Inboard Spoiler Position	hack 3	20	CITS
Right Inboard Spoiler Posit	hack 3	20	CITS
Lower Rudder Position	hack 3	20	CITS
Gross Weight	th	1	CITS
Fuel Weight	hack 1	1	CITS
Center of Gravity Position	hack 1	1	CITS
Mach Number	th	1	CITS
Airspeed	Th	1	CITS
Altitude (Pressure)	th	1	CITS
Altitude (Radar)	hack 2	1	CITS
Weight on Wheels Indicator	desc	1	CITS
Main Gear Down	desc	1	CITS
Refuel Nozzle Contact Indicator	desc	1	CITS
Structural Mode Control	desc	1	CITS
Terrain Following Radar	desc	1	CITS
T F Radar Mode	desc	1	CITS
Terrain Following Ride	desc	1	CITS
Aircraft Serial Number	None	None	CITS
Mission Date	None	None	CITS
Take Off Gross Weight	None	None	CITS
Stores Weight	None	None	CITS
Mission Type Code	None	None	CITS
Base Code	None	None	CITS

NOTES:
1. ASIP - ASIP Dedicated Sensor
2. CITS -Data from Central Integrated Test System Monitoring
3. CITS Control - Data manually input to CITS Control and Display Panel
4. Desc -Discrete Parameter
5. pv - Peak – Valley compression algorithm
6. th - Time History compression algorithm
7. hack1 - Recorded with gross weight
8. hack2 - Recorded with pressure altitude
9. hack3 - Recorded with vertical acceleration

Table 2 – Typical Possession and Utilization Records from REMIS/EIMSURS for 1 Aircraft for 1 Month

Item	Typical Data
Aircraft Serial Number	B1Bxxxx
Date of Record	97031
Owning Organization	TAC
Possessing Organization	Squadron Number
Possession Hours	744
Accumulated Flight Hours	3562
Hours for Month	26.5
Missions for Month	6

Table 3 – Typical Data from CAMS Database for 1 Aircraft (B1B850074) for 1 Month

Mission	Date	Mission Code	Flight Length	No. of Landings
1	97005	T2	4.5	3
2	97012	T3	2.1	2
3	97022	T2	6.1	1
4	97023	T2	4.6	4
5	97028	T3	3.8	3
6	97030	T2	5.4	2

Table 4 – Mission Mix as Defined by the Service Loads Recording System

Code	Occ per year	Mission Definition	Flight Lgth
		Training Missions	
1a	21	2 low altitude high speed segments with TFR on	5.77
1b	2	1 low altitude high speed segment with TFR on	3.91
2a	8	1 low altitude/high speed segment with TFR off, with refuel	5.0
2b	7	1 low altitude segment. with TFR off, without refuel	5.0
3a	10	High altitude with air refuel	3.65
3b	10	High Altitude without air refuel	3.65
11	12	Pilot proficiency flight	0.99
1H	2	Heavy weight mission 1a	6.14
2H	3	Heavy weight mission 1b	6.89
		Other Missions	
4	6	Ferry flight	4.02
5	1	Functional check flight	2.25

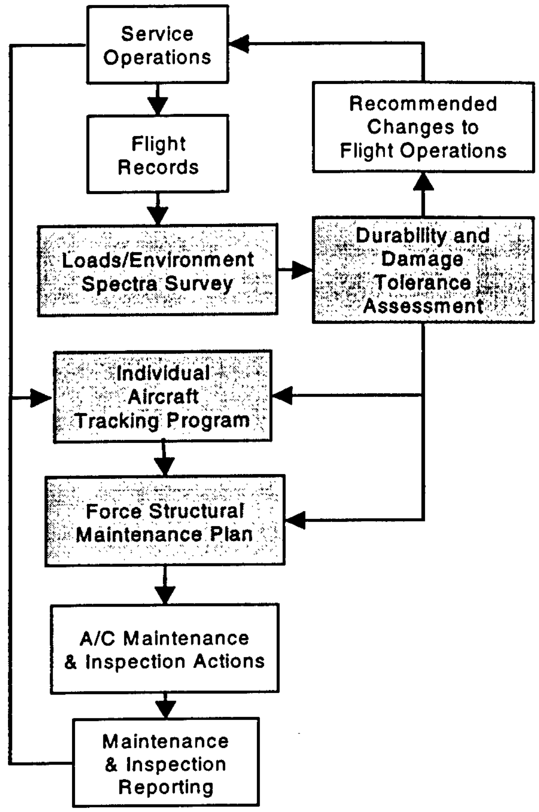


Figure 1 Maintaining Structural Integrity through Force Management

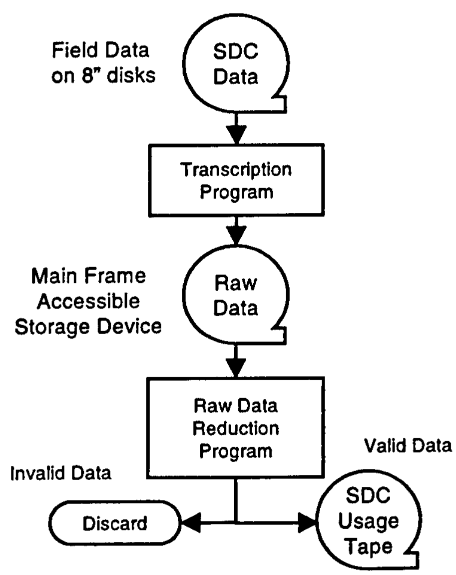


Figure 2 Ground Data Accumulation and Processing

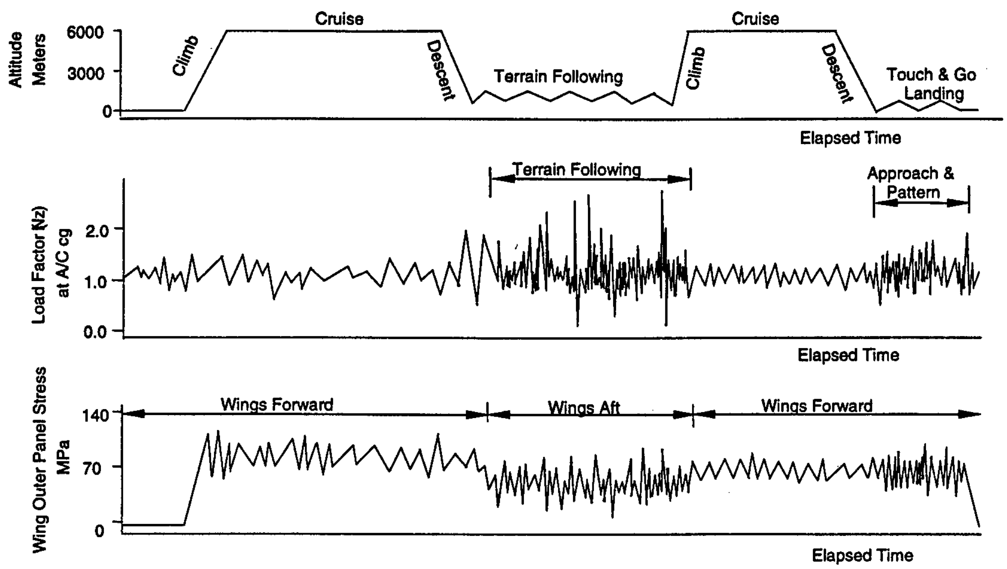


Figure 3 - Typical Mission Time History

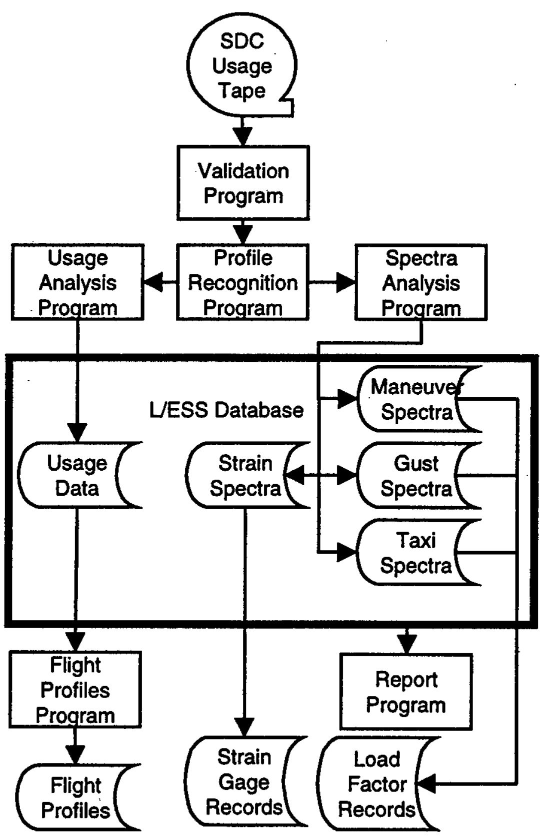


Figure 4 - L/ESS Program Flow

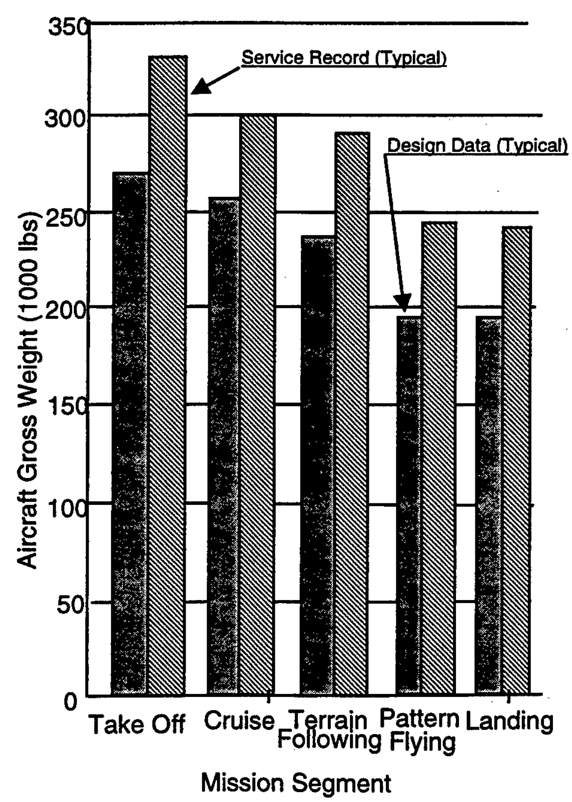


Figure 5 - Design Versus Service Gross Weights

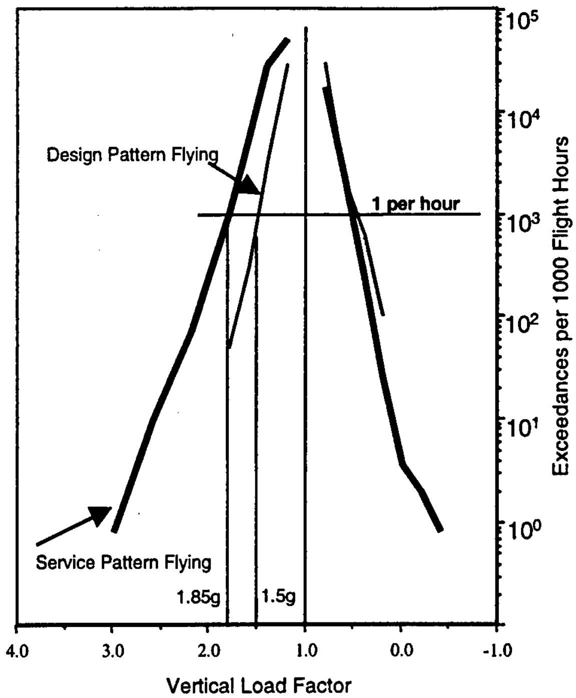


Figure 6 - Wings Forward Load Factors

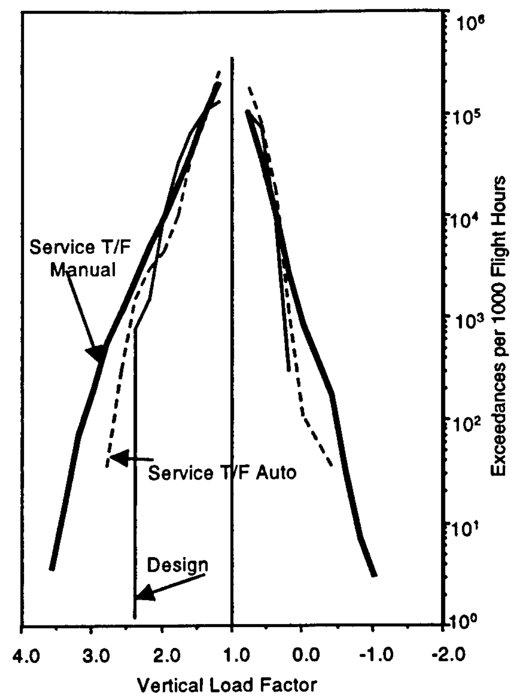


Figure 7- Wings Aft Load Factor Data

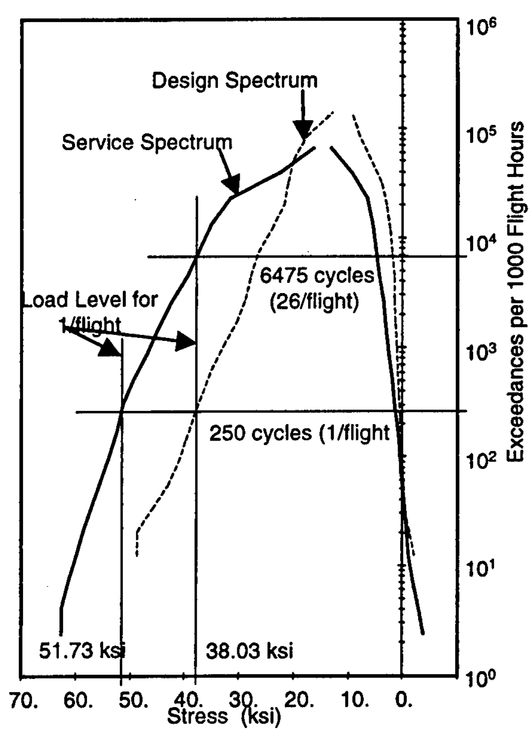


Figure 8 - Wing Carry Through Lower Cover Stress

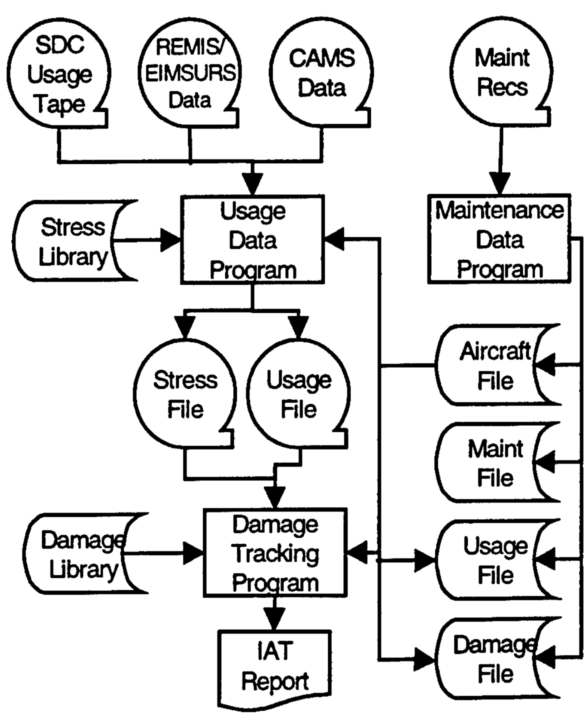


Figure 9 - Individual Aircraft Tracking Program

EUROFIGHTER 2000: AN INTEGRATED APPROACH TO STRUCTURAL HEALTH AND USAGE MONITORING

S.R.Hunt, I.G.Hebden
British Aerospace Defence,
Military Aircraft and Aerostructures,
Warton Aerodrome, Preston, Lancashire,
PR4 1AX, United Kingdom

1. ABSTRACT
This paper outlines the Structural Health Monitoring system being developed for Eurofighter 2000. The system is designed as an integral part of the avionics system, both on and off-aircraft, enabling the customer to perform fleet-wide monitoring of fatigue life and significant structural loading events.

2. INTRODUCTION
Eurofighter 2000 (EF2000) is a four-nation collaboration between Germany, Italy, Spain and the United Kingdom. The Structural Health Monitoring (SHM) system being developed for EF2000 is resident on each aircraft and will enable the operators to monitor accurately fatigue consumption and significant structural events, thereby safeguarding the structural integrity of the aircraft. The information obtained can be used to plan maintenance actions effectively, and to manage the fleet fatigue life pro-actively by the rotation of aircraft.

British Aerospace have system design responsibility for the EF2000 SHM system. The SHM is integrated with both the on-aircraft avionics systems and the ground support system. The on-aircraft software is part of the Integrated Monitoring, test and Recording Sub-system (IMRS). The IMRS is also responsible for crash data recording, video voice recording, management of equipment maintenance messages and provides the engine health monitoring interface with the ground.

The SHM ground analysis is performed by the Ground Support System (GSS), which is responsible for both mission and engineering support activities for EF2000. An overview of the SHM system is shown in Figure 1.

Different customer requirements for the SHM system has lead to the development of a flexible system which can be configured, without recourse to software change, as either a parametric based or strain gauge based fatigue monitoring system.

3. ON-AIRCRAFT SYSTEM
The SHM algorithms form part of the IMRS software fitted to each EF2000 and as such have access to the majority of the data within the avionics systems e.g. flight control system. The IMRS software is written such that on entry into service a growth potential of 100% is present (both processing and memory).

The SHM system performs real-time fatigue calculations and determines the life consumed by the airframe. Significant structural events and flight performance parameters are also monitored. A facility exists to record parameter and strain gauge time histories, if requested by the operator, for ad hoc studies (Figure 2).

The SHM data are available from the aircraft via two sources:

Maintenance Data Panel (MDP). The MDP is an on-aircraft equipment which displays information to the support personnel allowing them to query on-aircraft systems data. The SHM data available on the MDP are details of the total life consumed by each SHM monitored location and information on SHM event messages which may have occurred on the previous sortie.

Portable Maintenance Data Store (PMDS). The PMDS is a solid state memory device approximately the same size as an elongated cigarette packet (≈ 100*60*25 mm). The PMDS is used to transfer SHM, engine and maintenance data to and from the aircraft. The PMDS can store data for up to five individual flights.

In addition to the above equipment the aircraft has an optional recording device, the bulk storage device, which can be used to record a set of parameters requested by the support personnel.

3.1 Fatigue Calculation
Two SHM versions are being developed. These reflect the customer requirements for either a parametric based or strain gauge based fatigue monitoring system. Both versions will be validated by comparison with flight test data correlated to fatigue test results.

The SHM calculates fatigue life consumed at specified locations on the airframe. A single software is fitted to the aircraft which provides the operator with the facility to define, via data uploaded by the support personnel, whether each location is to be either parameter or strain gauge based.

For parametric based locations, real-time data are captured from the Flight Control System (FCS), Armament Control System (ACS) and the Fuel Gauging System (FUG). The FCS provides aircraft altitude, velocities and accelerations, the ACS provides

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

information on the aircraft weapons configuration and the FUG provides fuel mass information. These data are fed into the SHM which calculates the stress at each location by comparison with approximately 17500 "templates" held in internal memory. Each template, derived from finite element analysis and results of ground based airframe fatigue tests, corresponds to a particular aircraft configuration and set of flight parameters. The process iterates, generating a history of stress. The stress history is subjected to a real-time range-mean-pairs cycle counting analysis to calculate stress spectra and fatigue damage (Figure 3).

For strain gauge based locations, the stress is calculated directly from strain gauge measurement at an iteration rate defined by the user. A real-time range-mean-pairs cycle counting analysis, identical to the parametric locations, is performed to calculate stress spectra and fatigue damage.

The health of the data input to the SHM algorithms from external sources i.e. flight control system and strain gauges, are monitored. These, together with data indicating the health of the SHM data, are also available for off-aircraft analysis.

Typical SHM locations are given in Figure 4.

3.2 Event Monitoring

The SHM monitors in real-time the allowable parameter envelopes for structurally significant events and displays to the support personnel, via the MDP, any excursions outside these boundaries. Additional data are available on the PMDS for down-load to the GSS where further investigations using crash recorder data and bulk storage device data, if available, can be undertaken.

The definition and validation of the relevant parameter envelopes and the event corner points will be performed during aircraft development.

Up to seven envelope types can be monitored, each envelope being dependent on up to six configurations (e.g. clean, tanks). Up to five sub-factor envelopes and corner points for these envelopes/configurations can be defined to the aircraft software. An illustration of the functionality is shown in Figure 5.

3.3 Auxiliary Data

The SHM calculates auxiliary data. This consists of data which would be traditionally collated in the flying log, and additional design/performance parameter summaries (Figure 6).
In order to reduce the work load of the support personnel the following data are made available by the on-aircraft SHM system for each flight: flight date, take-off time, flight duration, stores configuration, mass of stores/fuel at each take-off and landing, number and type of landings (roller, touch and go, arrested etc.).

In addition, the following performance parameter summaries are available from each flight: number of airbrake operations, undercarriage cycles and refuelling probe deploys/refuels. Tabulation of "fatigue meter style" recordings of Nz in various aircraft mass and "points in the sky" bands are recorded together with Nz vs. roll rate tables. Maximum and minimum values of airspeed, altitude, Nz, and roll rate within one minute bands are also available.

3.4 Bulk Storage Device

A Bulk Storage Device can be fitted to each EF2000 for special studies purposes. The system is designed to compress and record data for a ninety minute flight, depending upon number of parameters and their sample rates and resolutions, and upon flight activity. The definition of which data time histories are recorded, and the rates and resolutions of those data, is controlled via the ground support system.

Any data entering the IMRS can be accessed, including the twenty analogue channels. Internal SHM data can also be accessed e.g. stress at each location. This gives the operator the ability to perform ad hoc studies e.g. operational load monitoring exercises, on any aircraft at any time, without the expense of special recording equipment.

4. GROUND SUPPORT SYSTEM (GSS)

The GSS provides the mission and engineering data processing for the aircraft and the management of the interface between EF2000 systems and the respective in-service ground based systems to support the aircraft operational staff at squadron level. In addition the GSS also supports engineering staff in the maintenance of the aircraft. The design of the SHM ground based analysis is integrated into the engineering support facility (Figure 7).

The quantity and complexity of the SHM data retrieved on a flight by flight basis from the aircraft is prohibitive to the support personnel. A full understanding of all the data would require specialist knowledge and training. As the support personnel are not expected to have detailed specialist knowledge, the SHM ground based analysis is designed to be used by an operator with a minimum of training.

Note: The volume of SHM data retrieved per flight from each EF2000 is approximately 350 times the volume of fatigue data currently retrieved and stored for a Tornado aircraft.

4.1 SHM Ground Analysis

The PMDS can store up to five flights' worth of data. On receipt of a PMDS the GSS de-multiplexes the SHM data into individual flight files and scales the data into engineering units. The GSS checks for completeness and data integrity. Identification of a failure, either in the data or in the transfer of data from the PMDS to the GSS will result in the GSS operator inserting a new PMDS into the aircraft and down-loading data stored in non-volatile memory on-board the aircraft. Post-flight the pilot is required to enter a profile code describing the sortie flown. This profile code is used in the verification of SHM data retrieved from the aircraft and will assist in fleet planning activities.

The SHM analysis assesses the validity of retrieved SHM data by comparison of individual aircraft results with that aircraft's history of fatigue data collected and by comparison with fleet trends. The GSS analyses the SHM health messages to determine the quality of the data. The system automatically calculates the values for any "unmonitored" flying using formally agreed rules.

The SHM validated flight records are stored in a database within the GSS. This database also stores both the aircraft component and role equipment serial numbers. This validated information can be transmitted to national information centres for fleet planning.

The database of SHM results held by the GSS at a base can be queried by authorised support personnel in order to manage their aircraft. A variety of standard query formats and chart types are provided, together with the facility to answer ad-hoc queries. Typical examples of data formats are given in Figures 8 to 11.

The system also allows the user to access the crash recorder data and BSD, if available, to display the significance of any event messages produced by the aircraft system.

4.2 SHM Up-load Data

The system controls the SHM definable data to be up-loaded onto the aircraft. This SHM data defines to the aircraft software the number and type (parametric or strain gauge) of fatigue locations to be monitored. The relevant structure and values of the stress templates for the parametric locations and the stress equations and sample rates for the strain gauge locations are also definable.

The SHM up-load data also defines to the aircraft software the number of parameter event envelopes to be monitored. The relevant parameters to be monitored and the event envelope corner points are also definable.

The ground system controls the preparation of the up-load file defining the data to be recorded on the bulk storage device. All data entering the IMRS system can be selected for recording (not just SHM specific) and their required recording rates and resolutions defined.

5. SYSTEM SUMMARY

The EF2000 SHM provides a complete structural health and usage monitoring facility incorporating:

1. Maximum automation on- and off-aircraft

2. Integration of the systems on- and off-aircraft for minimum maintenance

3. Minimum specialist training required for operators on the Base

4. 100% growth potential for on-aircraft system (processing and memory) at initial entry to service

5. Fatigue life consumption, stress spectra and parameter exceedance reporting available on a flight-by-flight basis for each aircraft.

6. Up to five flights of individual SHM data available for down-load to the ground support system

7. Flexibility to alter the SHM locations (up to twenty) to either parameter based or strain gauge based calculations or a combination of both, without recourse to aircraft or ground software change

8. Ad hoc facility for recording user selectable flight parameter time histories

6. CONCLUSIONS

The EF2000 SHM is the latest in a line of monitoring systems (1) and supplies information never previously available on a fleet-wide basis. The system is designed to maximise the long-term benefits to the aircraft operator, and marks the initial phase of a BAe initiative aimed at exploring "smart" structural health monitoring systems for military combat aircraft.

7. ABBREVIATIONS

ACS	Armaments Control System
BSD	Bulk Storage Device
EF2000	Eurofighter 2000
FCS	Flight Control System
FH	Flying Hours
FUG	Fuel Gauging System
GSS	Ground Support System
IMRS	Integrated Monitoring, test and Recording Sub-system
MDP	Maintenance Data Panel
Nz	Normal Acceleration
p	Roll Rate
PMDS	Portable Maintenance Data Store
SHM	Structural Health Monitoring

8. LIST OF REFERENCES

(1) Gill, L. and Wright, B.D., "Demonstration of the potential of Operational Loads Measurement", I. Mech. E. Aerospace Industries Division Seminar, "Operational Loads Measurement", University Of Bristol, 1993

Figure 1 Overview of Aircraft and Ground SHM Systems

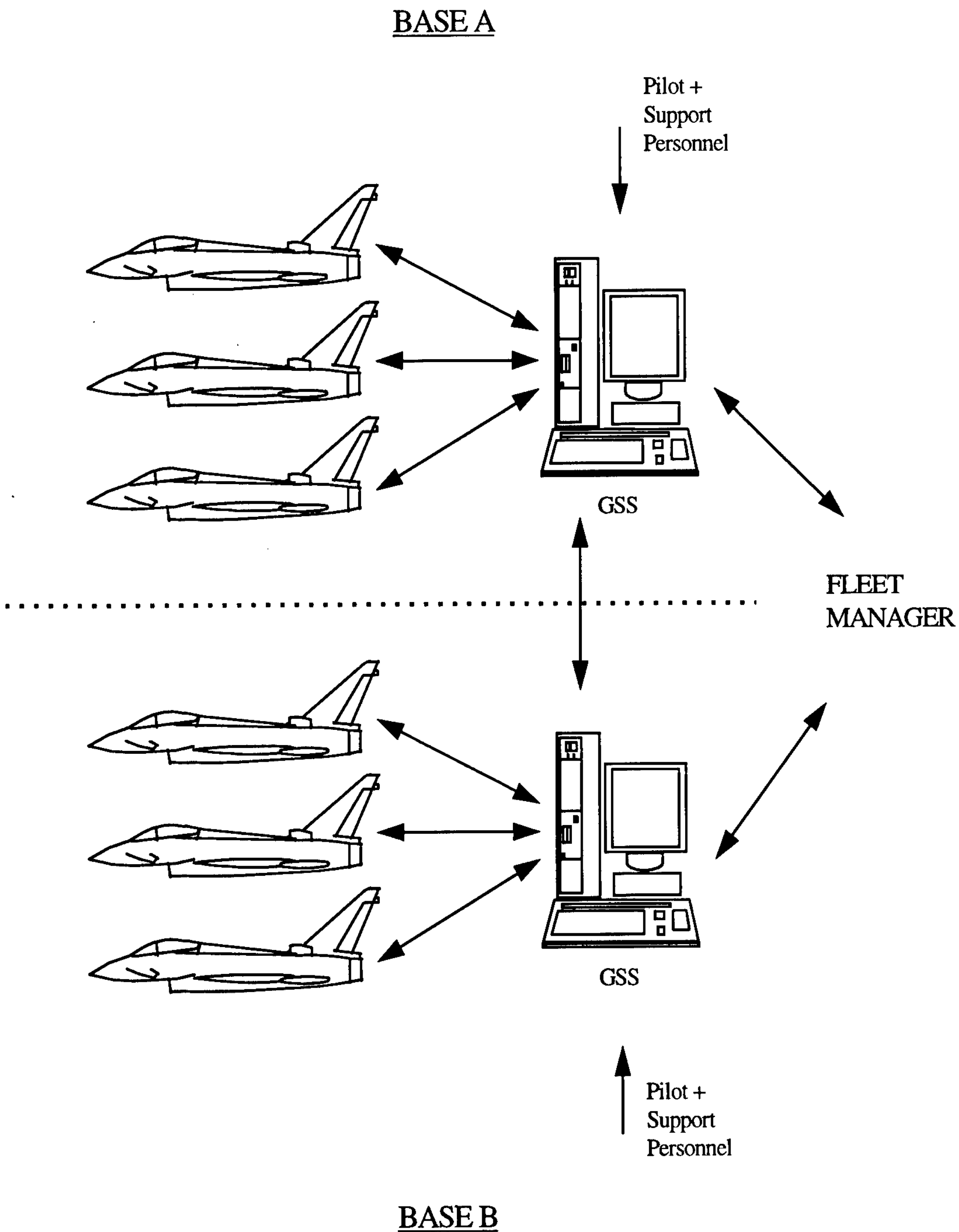


Figure 2 Schematic diagram of aircraft SHM system

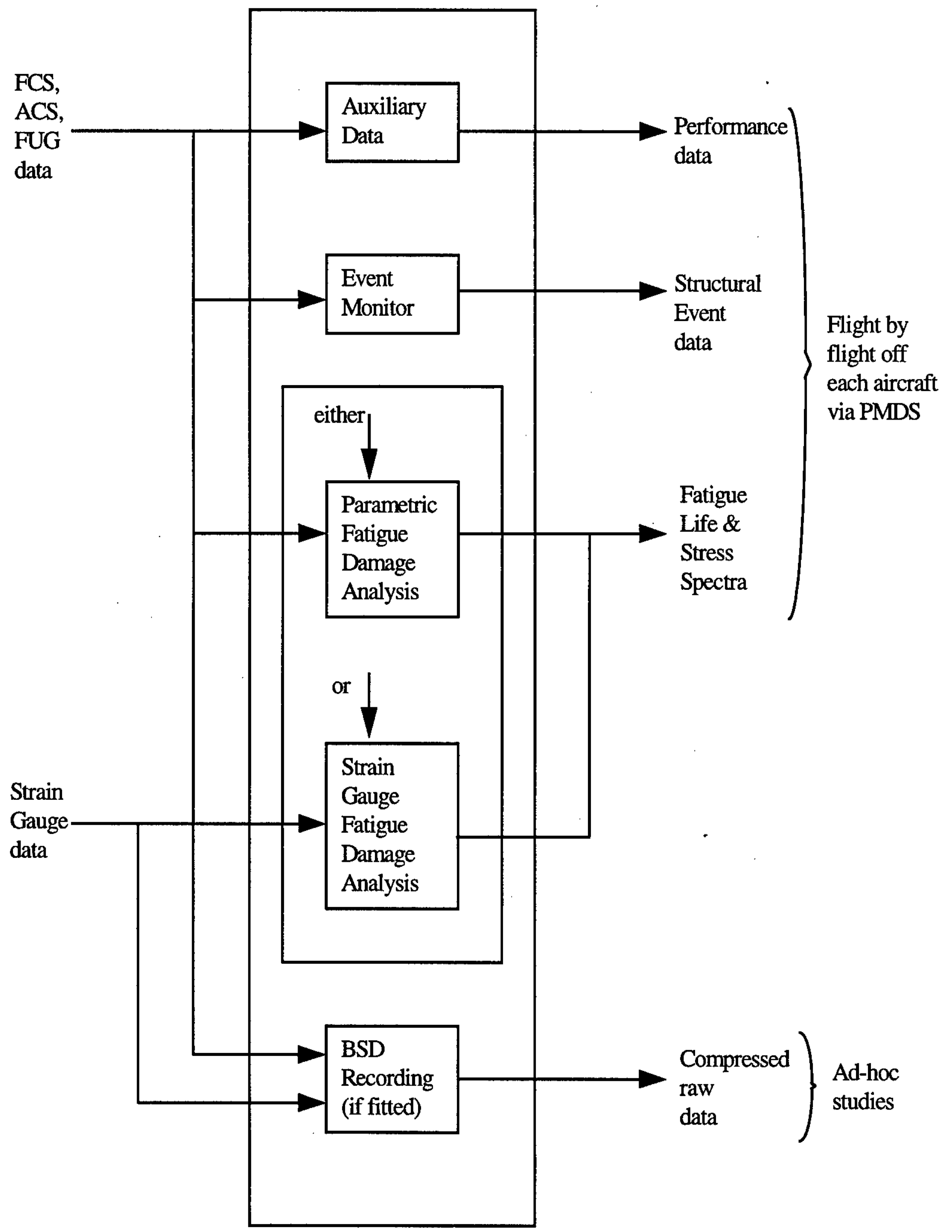


Figure 3 Overview of Fatigue Calculations

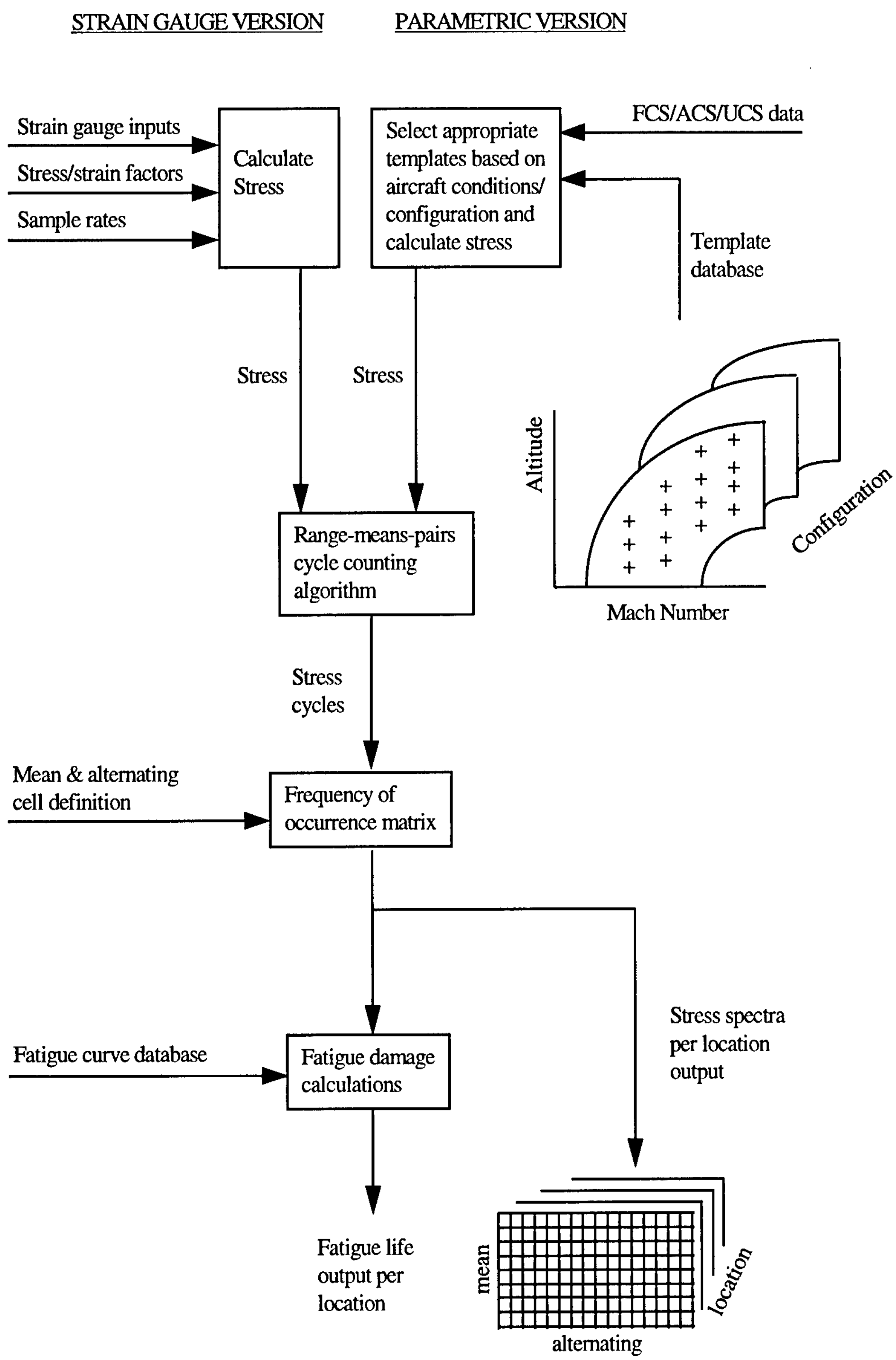


Figure 4 Position of SHM Locations

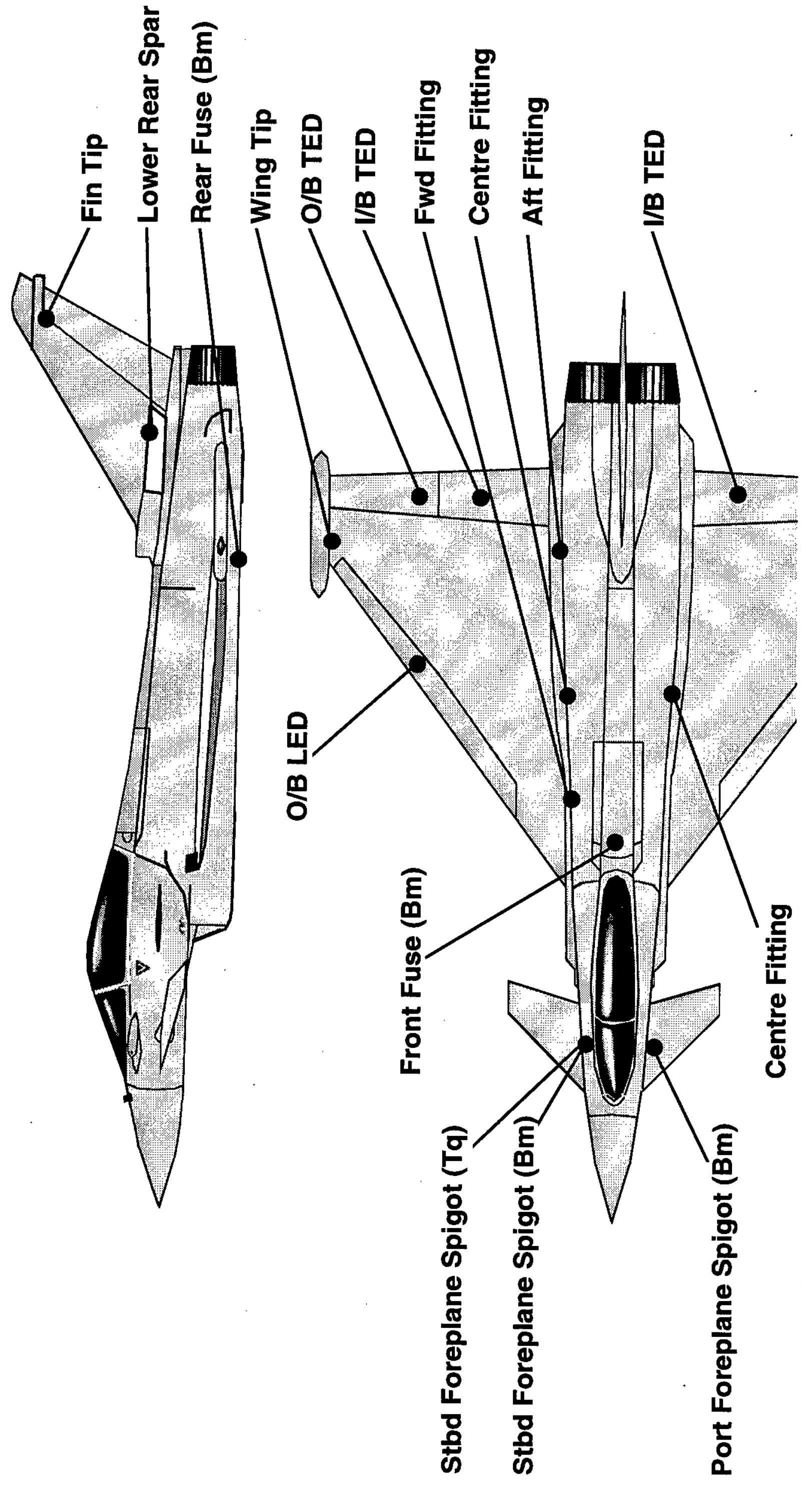
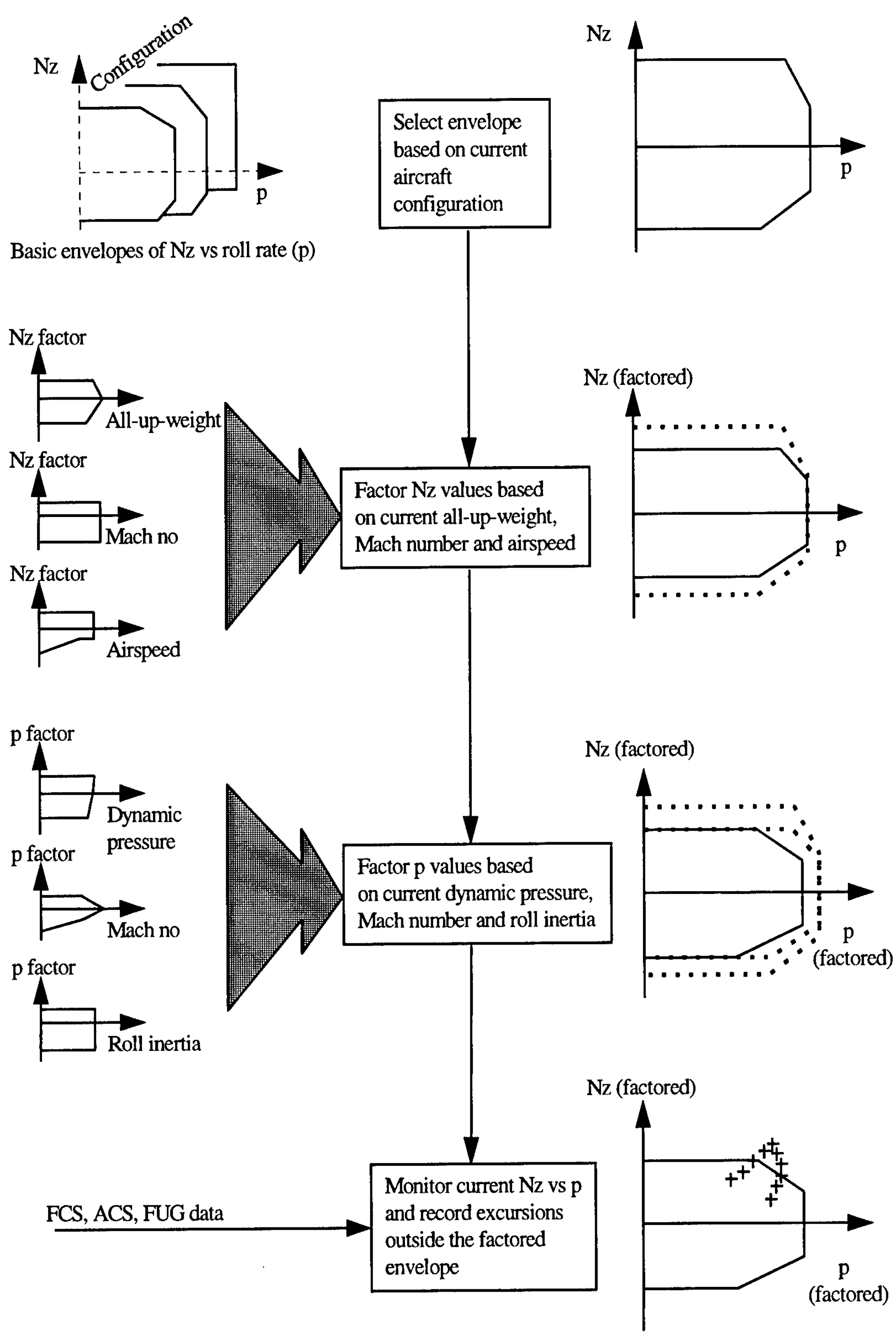
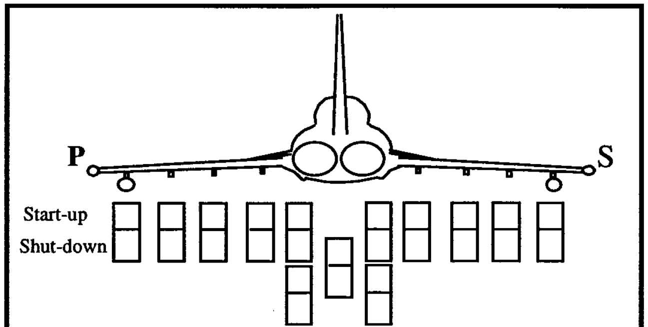


Figure 5 Example of Structural Event Monitoring



Tail No.	Flight No.	Unit Code	Station/Unit/Sqn

Flight Details	Date (Day/Month/Year)	
	Take-off Time	
	B/F Total Aircraft Hours	
	Duration	
	Total Aircraft Hours	
	% Supersonic / Sortie Code	
Mass in kg	Basic Aircraft Weight	
	AAR No. / Fuel Recvd.	
	Fuel at Take-off	
	Fuel at Landing	
	Ballast Mass	
	Ammo Mass / Take-off / Landing	



STORE CODES

RECORD OF LANDINGS	
Roller Landings This Flight	
Total Roller Landings	
Touch & Go Landings This Flight	
Total Touch & Go Landings	
Full Stop Landings This Flight	
Total Full Stop Landings	
Arrested Landings This Flight	
Total Arrested Landings	
Total Number U/C Cycles	

G COUNTS		
Level	Subsonic	Supersonic
1		
2		
3		
4		
5		
6		
7		
8		
9		
10		
11		
12		

ROLL RATE COUNTS		
Level	Subsonic	Supersonic
1		
2		
3		
4		
5		
6		
7		
8		
9		
10		
11		
12		

SERIAL NUMBERS	
Port Wing	
Starboard Wing	
Fin / Rudder	
Port Foreplane	
Starboard Foreplane	
Port Flaps	
Starboard Flaps	
Airbrake	
Undercarriage	

Captain's Name:	(Service No)	Tradesman's Name

Figure 6 Example of Auxiliary Data Output

Figure 7 Schematic diagram of Ground SHM System

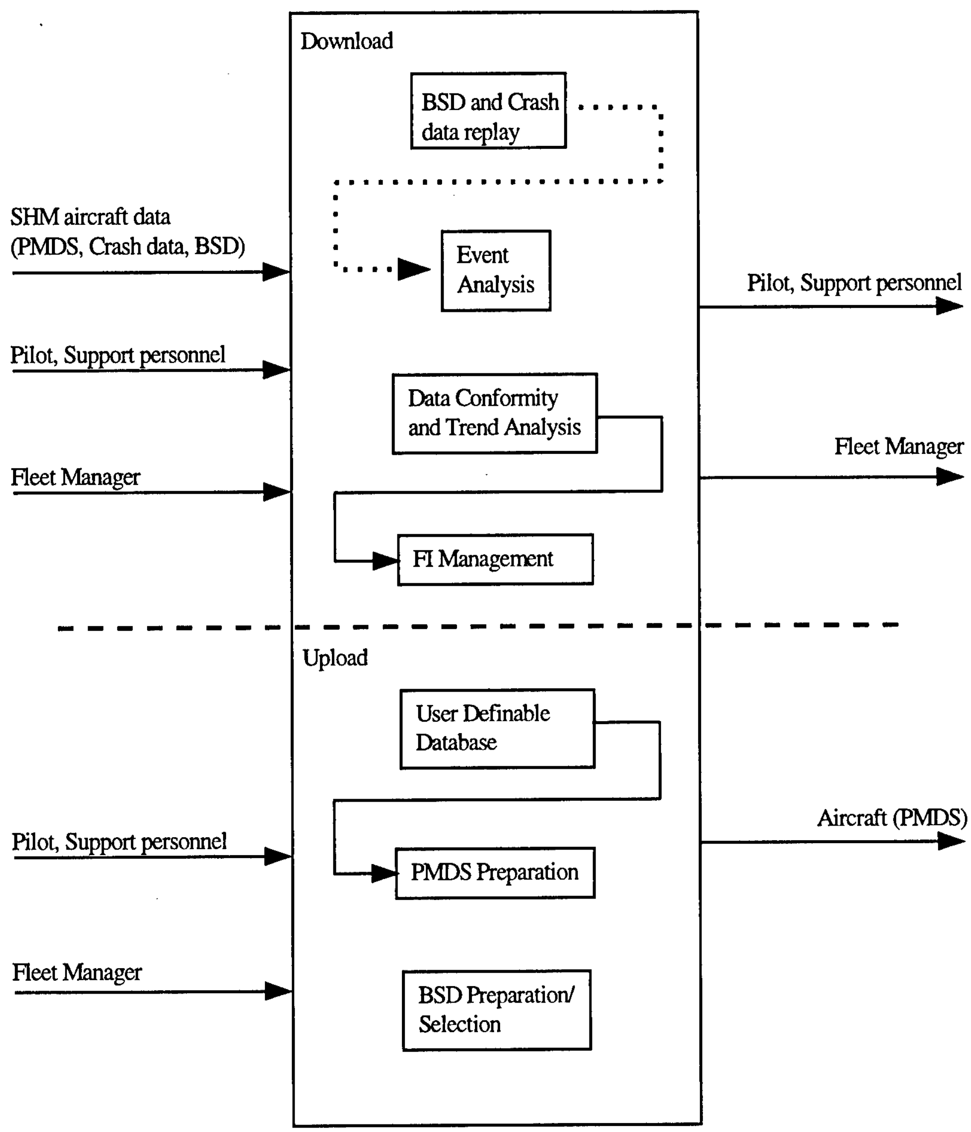


Figure 8 Typical Ground Analysis Queries - Tabular Output

SQUADRON xyz

Aircraft Number	Flying Hours	Lead location		Lead component		Landings				Sorties		Date	
		Name	Life	Name	Life	Stop	Roll	T&G	Arrest	Unmonitored	Total	From	To
Total													

Figure 9 Typical Ground Analysis Queries - Graphical Output

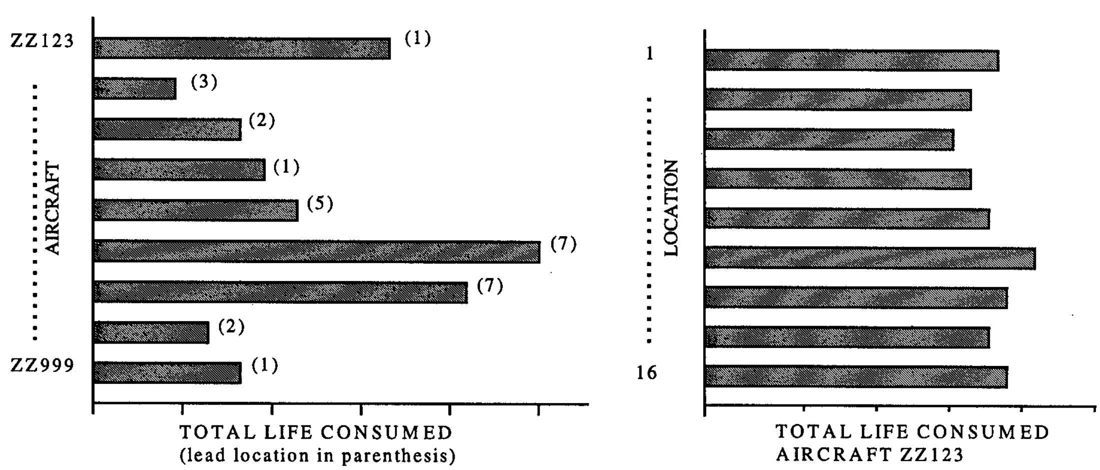


Figure 10 Typical Ground Analysis Queries - Graphical Output

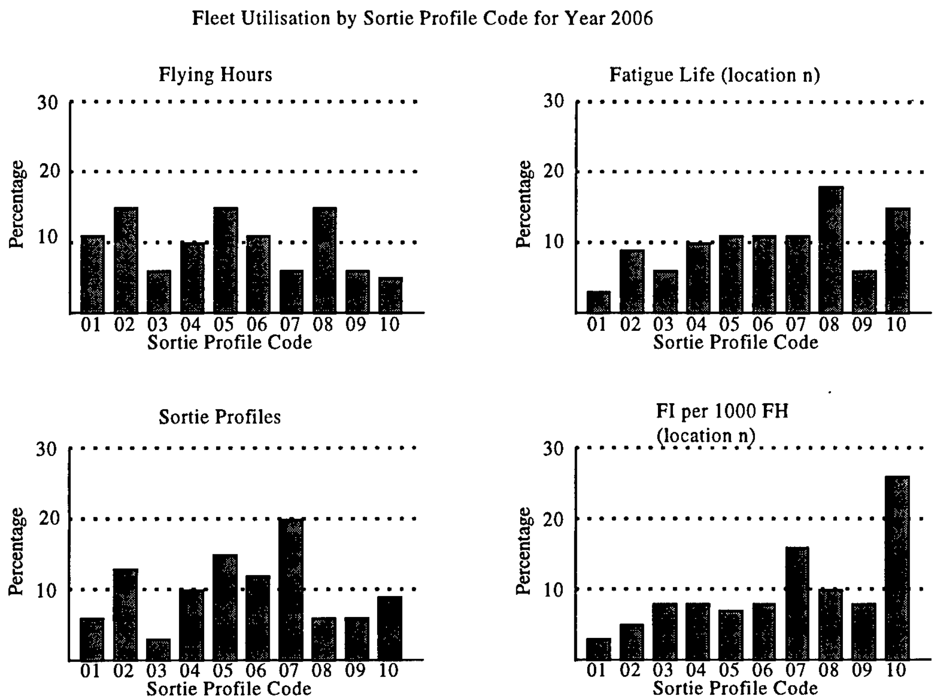
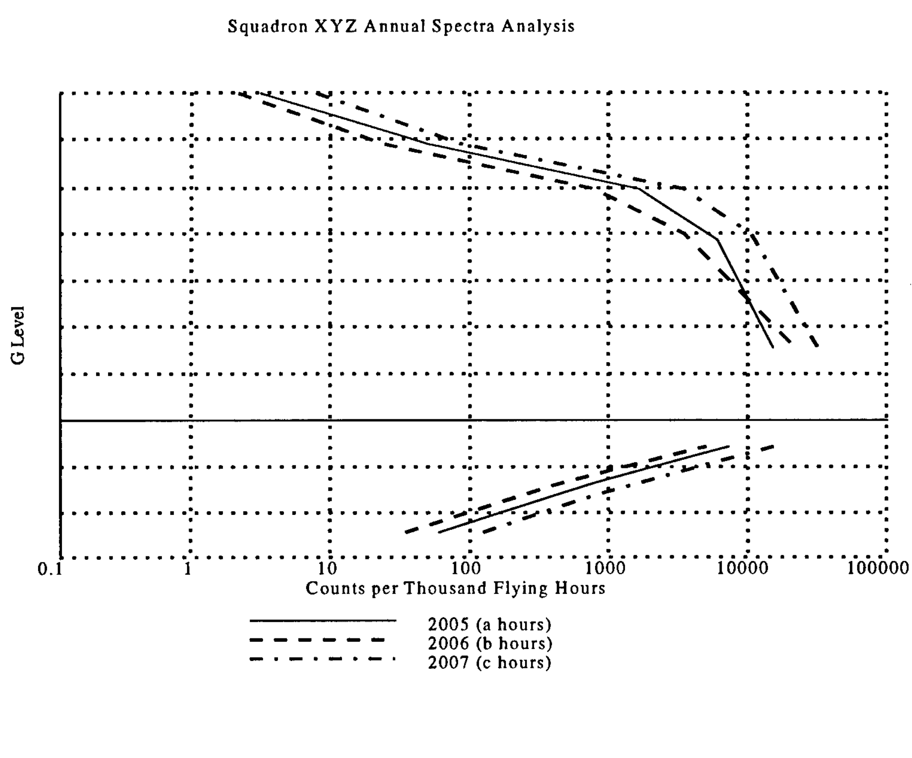


Figure 11 Typical Ground Analysis Queries - Graphical Output



OPTICAL FIBRE SENSING TECHNIQUES FOR HEALTH AND USAGE MONITORING.

P.D. FOOTE
A. BALL

BRITISH AEROSPACE, SOWERBY RESEARCH CENTRE
FPC 267, PO BOX 5, FILTON, BRISTOL, BS12 7QW

SUMMARY

The rationale for structural health monitoring is presented with an emphasis on the requirements of structural monitoring sensors. The advantages of optical fibre based sensing techniques are discussed and examples given of systems employing Bragg fibre gratings as strain sensors for load monitoring. The key issues surrounding the integration of optical fibres into structures, especially those fabricated from composite materials, are discussed. The problems of damage detection in composite material are discussed with reference to techniques for impact detection using Bragg grating sensors.

1. INTRODUCTION

In the early days of aviation history, aircraft structural health was based purely on personal experience or comprehensive inspection. With the awareness of fatigue and the implications on flight safety, there was a step change in the philosophy of aircraft structural health monitoring. An ability to predict the structural life, based on simple measured parameters, provided aircraft engineers with a valuable tool for both maintaining flight safety and reducing life cycle costs. Since the first introduction of recorded sortie data and flight parameters, development has been purely incremental. Even today the most effective systems are still a derivative of the early approaches. If a 'step-change' process improvement is to be created in Structural Health Monitoring (SHM), and hence a step-change reduction in associated life cycle costs, then a new vision of the benefits of SHM is required.

In the world of Smart Structures this vision is of an autonomous aircraft, capable of individual structural health assessment and a prediction of future condition. This is true for both civil and military aircraft. Realisation of this vision is still a long way off, but the technologies to support it are beginning to be demonstrated. It has been shown, in projects such as the X33 [1], that not only is there a desire to realise the Smart Structures vision, but a level of it may be achievable.

Combined with a deeper understanding of the fundamentals of damage initiation and growth, Artificial Intelligence (AI) and fibre optic strain sensing have shown potential for improved load monitoring and prediction of remaining useable life. Existing SHM systems, via a manual process, show an ability to update damage models to account for inaccuracies in original design assumptions. Advances in Systems developments for Testability, Maintainability & Reliability show how the existing SHM approaches could be automated either on or off-board. In addition, the *digital battlefield* raises the potential need, and has shown the technology elements for, real-time fleet management.

However, if a step-change is to be achieved then the developing technologies of automated damage detection will need to be integrated into the SHM system. Many potential detection technologies for both metal and composite structures are being investigated. In the authors' view, those showing most potential and demonstrable degree of maturity include detection of crack growth from acoustic emissions and impact detection by stress wave monitoring.

Acoustic emission, already used in the off-shore oil industry, has shown great potential to identify crack initiation in metal aircraft structures [2], and has the potential for further enhancement and automation using AI and advanced statistical approaches. If this is linked to the existing metallic fatigue assessment, the

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

potential for automated assessment and life prediction for each aircraft appears feasible.

Although still at the 'proof of concept' stage, stress wave detection technology has been demonstrated for impact damage detection in carbon fibre composite materials [for example 3]. Linked to a greater understanding of damage initiation and growth in composites, this technology could allow the vision of full structural damage assessment and life prediction leading to true condition based maintenance, to become reality.

This vision must be cost effective. Although it would undoubtedly bring significant improvements to operational availability, it is on the basis of reduced life cycle costs, with minimum impact on fly away costs, that such a system must be assessed. This will only be achieved by the minimal additional system fit, and the inclusion of the SHM system requirements in future air vehicle system design architecture at the initial design stages. As such the use of advanced sensor fusion will be paramount, as will low cost sensors. This paper will address sensor forms that are based primarily around the strain-monitoring role for Operational Load Monitoring, but have the potential for wider use as higher bandwidth acoustic or dynamic response sensors.

2. SENSORY STRUCTURES BASED ON OPTICAL FIBRE TECHNOLOGY.

Optical fibre sensors offer significant advantages over traditional electrical devices. Intrinsic fibre sensors (those in which the fibre itself acts as the sensor) are extremely compact and light weight (fibres are typically 1/8th of a millimeter in diameter and weigh less than 40 micro-grammes per metre, unclad). Light travelling in fibre is unaffected by radio or microwave interference so no screening is needed for fibre sensors or their down-leads and no impedance matching amplifiers are required to drive links. Multiple sensors may be strung along a single fibre thus economizing on the need for connections. These latter features add further to the system weight advantage and obviate the need for electrical power delivery to remote modules.

For an aerospace environment, both electro-magnetic compatibility and weight are major considerations, but the same environment also places severe constraints on the optical fibre technology. Interconnections are required between single mode fibres, with core diameters of around 9 microns. These require precision aligned, optically polished components which must remain in alignment even when subject to mechanical shock and vibration. Most fibre technology is optimized for the telecommunications industry and significant development of connectors is necessary to comply with, for example, a modern fighter airframe environment.

Opto-electronic hardware for converting optical signals from sensors into electrical data for storage and analysis, is currently developmental. Depending on the technique used, a complex decoding task exists to extract sensor data from optical fibres and as yet there exist no industry standards specific to these systems.

However, the prospects for overcoming these difficulties are good. Optical avionic harness programmes (e.g. DTI/CARAD Fibre Optic Harness Study Phase II RA6/31/07) are actively addressing the connector issues for multi-mode fibres. The higher bandwidth requirements anticipated in future avionics (e.g. for radar and even civil, in-flight entertainment), are also stimulating single-mode connector development in similar programmes. Opto-electronic hardware for sensor applications is riding on the back of developments for telecommunications applications and is set to benefit from the huge investment in that industry.

3. OPTICAL STRAIN SENSORS FOR LOAD MONITORING

Operational load monitoring of airframes allows a measure of aircraft usage. Load monitoring in aircraft can be achieved from flight parameter data coupled with airframe load case models. The fatigue behaviour of key structural components is then combined with these load histories to provide a measure of fatigue-life usage. Strain sensing at points across the airframe provides a more detailed input to the load calculations and can reveal out-of-envelope conditions. Electrical strain gauges are used in some existing systems for this purpose. The capabilities of these current systems can be expanded using optical fibre strain sensing especially in respect of carbon composite structures.

A technique under exploration at BAe uses in-fibre Bragg gratings to detect strain. Optical fibre is processed at discrete intervals, to create sensor elements. These elements are perturbations in the fibre's refractive index with a grating structure [4]. They have the property of reflecting any light, which is shone down the fibre, in a predetermined range of wavelengths centred about a peak wavelength value. This peak reflection wavelength will change as the fibre and the grating structure within it is strained. Tensile strain shifts the peak wavelength towards longer wavelengths: the reverse for compressive strain. Measurement of this wavelength shift therefore leads to a strain measurement.

The concept for a strain sensing system architecture comprises three main elements:

1. The sensing fibre itself which would be bonded to or embedded within structural elements such as wing skins;
2. The interconnections typically single mode optical fibre connectors which terminate optical cables;
3. The opto-electronic interface which supplies light to the fibres and receives optical signals for decoding into electronic strain data. This interface would be part of a modular avionic system architecture, perhaps a remote data concentrator that handles other aircraft system status data simultaneously.

The sensing technique relies on providing a source of light, the wavelength of which may be tuned to predetermined values. This light is passed to a fibre containing a sequence of Bragg grating strain sensors. Under nominal or zero strain conditions, each sensor is manufactured to reflect at a distinct, unique peak wavelength. A string of sensors along the same fibre can therefore be identified uniquely, by their particular reflection wavelengths. The tunable light source accesses sequentially, each sensor in turn. This is analogous to tuning a radio receiver to different transmissions within a radio band.

Figure 1 illustrates an architecture for such a sensor network currently under investigation at BAe. The tunable light source is synthesized from a broad band Light Emitting Diode (LED) and an electronically tunable, optical filter. The filter passes light from the LED in a narrow range of wavelengths centred on a peak value. This peak value is electronically variable. As the filter transmission wavelength matches any of the reflection wavelengths of a Bragg grating sensor, a strong reflection signal travels back along the fibre to a receiver photodiode. The electrical output of the system is a temporal pulse indexed to the tunable filter drive signal. The architecture shown in figure 1 is capable of addressing up to 40 sensors from a single system on 8 parallel, fibre channels.

The limitations on the total number of sensors per system arise from trades-off between bandwidth of the light source and/or tuning range of the optical filter and the range of strain measurements required from each sensor. LED sources typically supply up to 40 nm of useful light bandwidth. A strain range of $\pm$ 3500 $\mu\varepsilon$ and a temperature range (Bragg gratings are also to sensitive temperature) of 200 ° C would require a channel bandwidth of approximately 5 nm per sensor. A 40 nm LED would therefore accommodate a maximum of 8 such sensors.

The rate at which strain information can be acquired is constrained principally by the tunable filter technology in this design. This active opto-electronic device is the heart of the system and although they are commercial components, they are still aimed at developmental systems mainly within wavelength multiplexed telecommunications applications. BAe is currently evaluating tunable filters based on both piezo-electrically scanned Fabry-Perot interferometers and integrated acousto-optical filters. In the architecture described here, the Fabry –Perot devices allow a sensor address rate of up to 500Hz for each sensor. The integrated optical filter, in a slightly modified architecture with fewer sensors is capable of 2kHz address rates.

The wavelength multiplexing method described above can be supplemented with spatial multiplexing as in the architecture of figure 1, where sensor numbers are increased by splitting the light path between multiple fibres. The practical limitations here, are the available power budget from the LED. The reflected optical power from a single Bragg grating in the above architecture is of the order of 1nW. This pushes simple PIN silicon photodiode detectors to the limit for signal bandwidths greater than a few hundred Hz.

An optical system based on this architecture could therefore deliver between 40 and 70 single strain measurement locations. The structurally integrated components would comprise a mere eight optical fibre cables with single ended connectors. These would connect to a Line-Replaceable Module with opto-electronic front end, situated within a modular avionics architecture.

The inclusion of optical fibre sensors within an airframe environment raises a number of issues depending on the mode of integration. Surface or parasitic structural integration would follow the existing electrical strain gauge approach. The use of optical fibres, however, raises the possibility of structurally embedded sensors.

4. STRUCTURALLY EMBEDDED OPTICAL FIBRE SENSORS

Optical fibres, because of their low cross-sectional area, are suited to amalgamation within the host matrix of thermoset or thermoplastic structural composites. The fine optical fibres are reminiscent of the structural carbon fibres and the possibility of embedded optical sensors has been pursued by a number of groups [see for example 5]. The motivation behind this is that the material can be treated as intrinsically 'sensitive'. The sensor readings would truly register the material strain without risk of damage by exposure to external forces.

Immediately the question of compatibility is raised when considering embedded components within a host matrix. Strictly speaking, the inclusion of optical fibre would constitute a new material with potentially unknown properties. Many studies have now been conducted such as those in reference [5] in which the structural implications of fibre inclusion in composite material have been evaluated. The influence of fibre density and fibre type on material strength has been investigated.

From our findings and the findings of others, typical knockdown factors in static strength are typically less than design allowables. There is also a growing body of evidence gained from fatigue tests of conditioned composite samples containing optical fibre, that fibre inclusion does not compromise material properties.

Sensor reliability and calibration is a major concern when instrumenting components in structural installations. For a Smart sensory structure where the sensors are embedded, reliability issues are magnified since sensor repair would be impossible without damaging the structure. Redundancy using multiple optical fibres can overcome localized catastrophic failures resulting in fibre breakage but equally important is the long term fidelity of strain data from the embedded sensors.

Likely failure modes would be slow disbonding between the embedded fibre and the host material. This would result in degraded transmission of stress from the structure to the sensor.

Figure 2 illustrates some results from fatigue testing of embedded fibre Bragg grating strain sensors in coupons of T800/914 material in a quasi-isotropic lay-up. In this example, the sensor's responsivity, expressed in micro-strain per kNewton load, was monitored during constant amplitude loading between tensile and compressive limits. Even after 1 million cycles, there is no significant degradation of the sensor's performance. For these trials, the optical fibre was laid in a 0-degree direction between 0-degree oriented, unidirectional plies of T800 material with loading also in the 0-degree direction. These fibres were also coated with acrylate buffer jackets which is not optimal for the relatively high temperature and pressure cure conditions of the material but which, nevertheless, did not compromise sensor performance in this case.

The evidence from experimental trials such as those described above suggests that optical fibre sensors can be combined with high grade aerospace materials while retaining their respective functional characteristics over a wide range of conditions. Still to be addressed, however, are full-scale qualification tests for material with embedded optical fibre and also manufacturing issues. Qualification is expensive for new materials and differing configurations of optical fibre would require independent tests (for example curved versus straight fibre paths and fibre ingress/egress points). Manufacturing entails heavy investment in tooling. Modification of existing processes to accommodate optical fibre would undoubtedly prove costly. These drawbacks must be weighed against the potential enhancements to material functionality that accompany sensor embeddment. It follows from this, in order to be cost effective, commitment to these sensory structure concepts must occur at the aircraft design stage.

5. SURFACE MOUNTED SENSORS

Many of the advantages of optical sensors for structural monitoring can be gained without the qualification or manufacturing constraints mentioned above. Surface mounted, or parasitic sensors such as strain gauges, have no impact on mechanical properties of the underlying structure provided the ratios of moduli or physical dimensions of sensor to structure are small.

A design for an optical strain gauge rosette is shown in figure 3. The device is modeled on electrical strain gauge designs forming the equivalent of a 'delta' rosette. To uniquely define planar strain, three independent axes of strain must be measured to determine the two principal coefficients plus the off-diagonal shear elements of the two-dimensional strain tensor. They are found most accurately by defining three strain measurements axes with the largest possible angle between them, hence the equilateral triangle arrangement of the delta rosette.

The gauge features four Bragg grating sensors arranged in sequence along a single fibre loop. The loop is sandwiched in a polyimide film. Three of the gratings form the sensor elements aligned at mutual angles of 120 degrees. The fourth is situated at the tip end of the fibre and is encased within a thin walled capillary of silica. The capillary and fibre are bonded so as to leave the contained grating unrestrained. As such, it is de-coupled from stress. This strain-isolated sensor will, however respond to changes in ambient temperature. As with most strain sensors, whether electrical or optical, Bragg gratings are susceptible to temperature variations. Normally, without an independent measure of one or the other, it is impossible to untangle the combined effects of temperature and strain from the data obtained using these devices. The configuration of this device overcomes this problem allowing such an independent measurement to be made.

Figures 4a to 4c to show measurements taken from experimental rosettes bonded to aluminium test coupons. Figure 4a shows the response to load of the temperature sensitive, strain isolated sensor. While the reference, electrical strain gauge bonded alongside registers increasing strain, the output from the isolated grating is unchanged. Figure 4b shows the effect on the same sensor of temperature change. Finally, Figure 4c shows the combined measurements from all four sensors when the coupon is loaded in a direction parallel with one of the three delta axes at constant temperature.

This prototype device has a number of notable features. The rosette is fabricated by laminating optical fibre between films of polyimide of the same type used for strain gauge manufacture. It can be bonded to test items using the same qualified adhesives and application techniques already painstakingly developed for electrical strain gauges.

The optical rosette also contains an integral temperature sensor from which data can be obtained to accurately compensate strain measurements against temperature variations. Electrical gauges rely on matching the thermal properties of the gauge to the material of the test structure. Different gauges are selected to suit, for example, titanium and steel. The optical device is suitable for any substrate (provided a good bond can be achieved).

Being all-optical, the rosette requires no electrical power, saving problems with local power delivery and also displaying no ohmic heating (the optical powers within sensors of this type are typically micro-watts with little dissipation).

Perhaps most significantly, the optical rosette needs only a single optical fibre connection to access all three strain sensors plus the temperature sensor. A resistive foil, electrical strain rosette performing the same measurements requires typically nine wire connections that must be screened from the effects of electro-magnetic interference. To fully instrument a structure the size of a fighter aircraft may take hundreds of strain sensors. The line driving requirements (local amplifier modules) of electrical strain gauges would add significantly to the weight of the structure when used in this number. Equivalent coverage using optical sensors could be achieved using at least an order of magnitude less weight.

6. IMPACT DETECTION USING OPTICAL FIBRE SENSORS

A major cause of damage in carbon fibre composite structures is delamination cased by accidental impacts. For aircraft, low velocity impacts from bird strike and runway stones are unpredictable and their effects are currently picked up by ground based inspection. Sensor systems capable of locating and quantifying impact events, in terms of total energy and duration, would form a major part of an automated inspection process for composite structures.

Necessary features of the system would be the ability to monitor continuously throughout the lifetime of the structure and to locate damaging events at least to the region of a replaceable sub-component. The lifetime monitoring constraint is crucial in a purely passive system. Damage can occur to the structure even when not in service from such accidents as dropped tools or collision with ground vehicles.

Impacts will generate flexural and extensional modes in a structure with differing propagation characteristics but at levels of far greater amplitude than, for example, the acoustic emissions generated by matrix cracking. As a method of damage monitoring, impact detection is indirect but poses less of a challenge to the sensor transducer technology with respect to sensitivity and noise rejection than the more direct methods of acoustic emission detection. The real challenge with impact monitoring is the signal processing required to make a reliable, autonomous decision that structural damage has occurred. The optical Bragg grating strain sensors described in the previous sections are capable of converting transient strain effects generated by stress wave propagation, into a wavelength modulation. The aftermath of an impact is therefore detectable using the same basic sensor technology. The origin of stress wave events can be located by simultaneous detection from multiple sensors distributed within the structure. Triangulation based on the relative timing of transient signal at the sensors allows the source to be pinpointed. This leads to signal bandwidth constraints on the sensor system. Acoustic events in composite material propagate in typically in the range 1 to 10 km/s. For a 1cm resolution, signal detection bandwidths in excess of 100 kHz are therefore required. The opto-electronic strain sensing system described earlier is restricted to bandwidths of a few kHz. This restriction results principally from the tuneable filter. The sensors themselves are capable of much greater bandwidths, limited only by the transit time of light through the Bragg gratings. These are typically 2 to 3 mm in length allowing potential signal bandwidths of greater than 100 GHz. Optical transceivers of multi-Gbit capacity are already developed for telecommunications applications.

Alternative methods of converting the wavelength modulations produced in the sensors to electrical signals can be used based on passive wavelength dependent filters. The amplitude of light transmitted by such filters depends on its wavelength content. Wavelength modulated light, such as reflected by an excited Bragg grating, will be converted therefore, to a corresponding amplitude modulated signal after passing through such a filter. This amplitude modulation is then converted to an electrical signal in a photodiode detector.

The transient record of an impact event is displayed in figure 5 . This was recorded from a fibre Bragg grating embedded in a composite panel. The signal detection used a wavelength filter fabricated from a second Bragg grating and employed the principle just described.

Clearly, the potential for extremely wide bandwidth optical sensors exists for the detection of damage in structures by monitoring acoustic events. Acoustic emissions from crack propagation in metals requires detection bandwidths in the 100 kHz to 1MHz range. Although not as well understood, the detection of matrix cracking in composites will probably require similar bandwidths. Active non-destructive ultra-sound probes for direct imaging of damage operate at multi-MHz frequencies.

7. CONCLUSIONS

Optical fibre sensors systems for structural usage monitoring applications are moving beyond the proof-of-concept phase.

The issues of structural integration of fibres in aerospace composites have been explored and indications are that a route to viable, qualified materials containing embedded fibres exists.

Parasitic optical sensors with fewer structural compatibility issues, could be utilised nearer term. They offer significant advantages over electrical strain gauge installations. Weight savings due to fewer connections without screening or line driving electronics can allow greater sensor coverage for more accurately resolved load monitoring.

Load measurement and damage detection based on acoustic event monitoring is possible using generic, intrinsic fibre sensors such as Bragg gratings. This multi-functional sensor approach fits well with the concept of integrated structural health and usage monitoring.

Associated optical system components are advancing in maturity mainly as a result of developments in the telecommunications industries aimed at large capacity data transmission. These advances can benefit the aerospace structural monitoring application.

The bandwidth, weight and EMC advantages of optical fibre over copper have already made an impact in the local networking and comms links in aircraft. Optical sensors would be readily compatible with this existing opto-electronic infrastructure.

Significant developments are still required in single mode fibre connection technology especially with respect to embedded fibre sensor concepts. Manufacturing practices encompassing fibre integration also need to be embraced at the structural component design phase if these techniques are to be cost effective.

8. ACKNOWLEDGEMENTS.

The authors acknowledge the support of the DTI/EPSRC LINK programme 'MAST' (YAG/8/2/1054) and the EEC BRITE EURAM programme 'MONITOR' (BE 95-1524).

9. REFERENCES.

1. I. Melvin *et al*, 'Integrated Vehicle Health Monitoring (IVHM) for Aerospace Vehicles' Proceedings of the International Workshop on Structural Health Monitoring', Stanford University, Stanford CA, Sept. 1997. Ed. Fu-Kuo Chang.
2. I.Searle, S. Ziola, S. May, 'Damage Detection Experiments and Analysis for the F-16' *ibid.*
3. W.J. Staszewski, K. Worden, G.R. Tomlinson, 'Optimal Sensor Locations for Impact Detection on a Composite Panel' Proc. DAMAS'97, University of Sheffield, 1997. Eds. J.M. Dulieu-Smith, W.J. Staszewski, K. Worden.
4. I. Bennion, J.A.R. Williams, L. Zhang, K. Sugden, 'UV-Written In-fibre Bragg Gratings', Opt. And Quant. Electron. Vol 28, pp93-135 1996.
5. D.W. Jensen, J. Pascual, J.A. August, 'Performance of Graphite/Bismaleimide Laminates With Embedded Optical Fibres', Smart Mat. And Struct. , Vol 1, pp 24-30 1992.

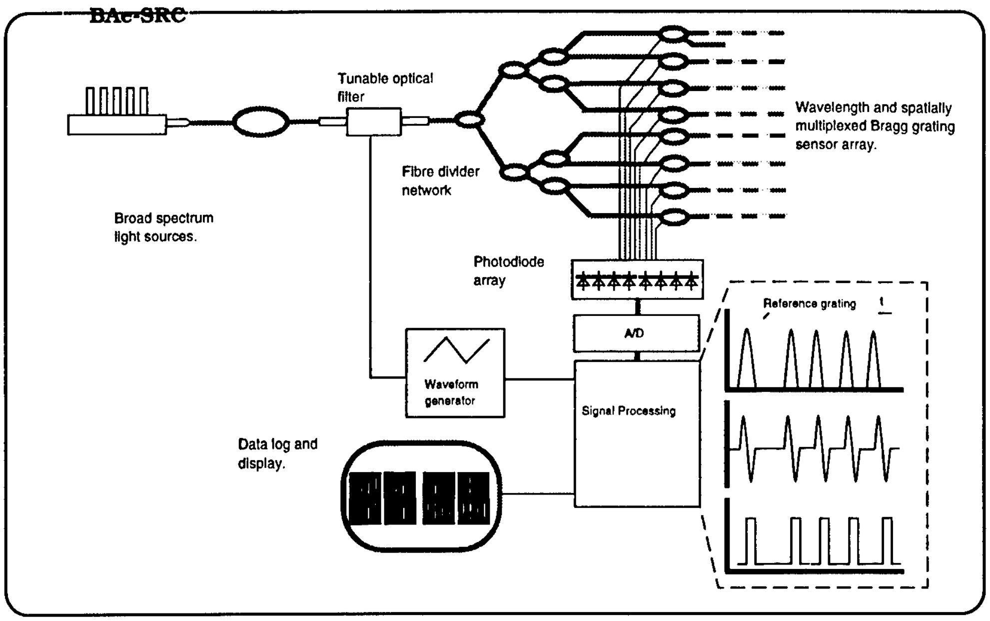


Figure 1. Optical strain sensor architecture based on in-fibre Bragg grating sensors.

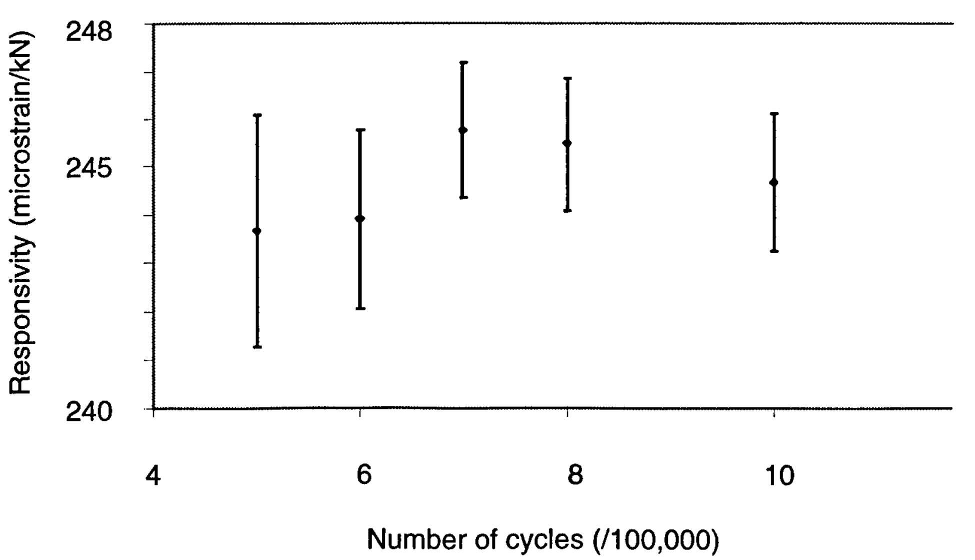


Figure 2. Responsivity of a Bragg grating strain sensor embedded in T800/924 as a function of cyclic fatigue loading

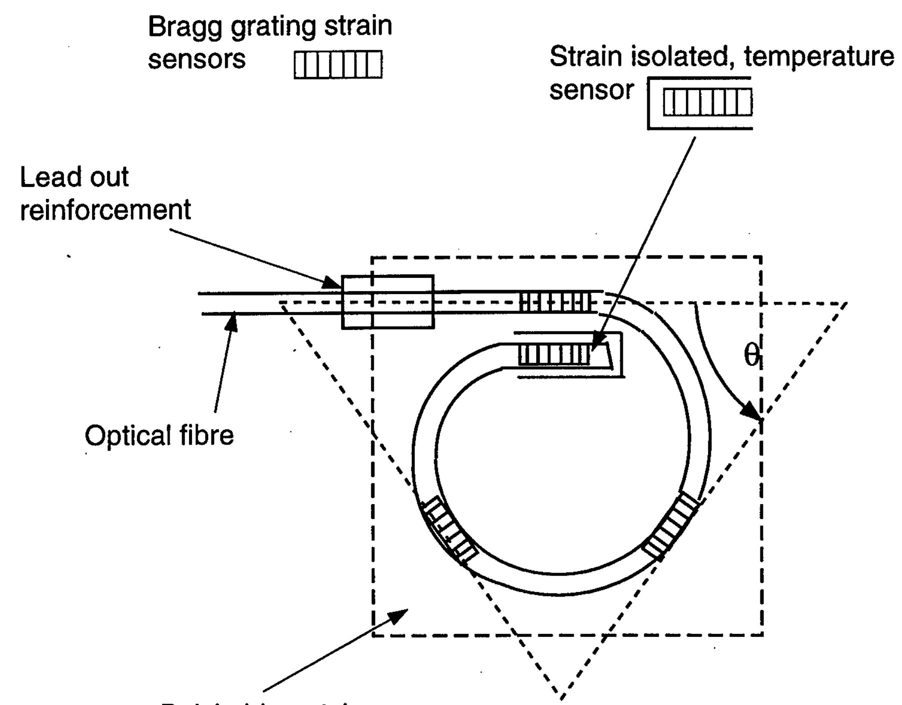


Figure 3. Optical fibre 'rosette' strain gauge using Bragg gratings.

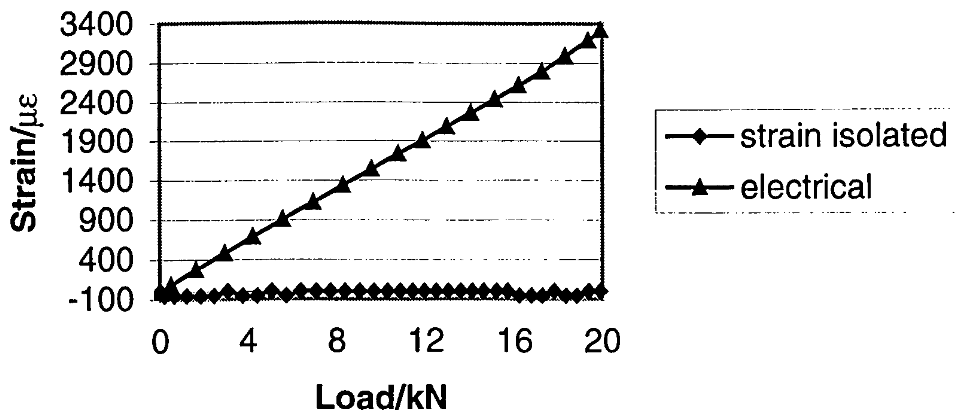


Figure 4a. Response of the strain isolated temperature reference sensor (see figure 3) to load.

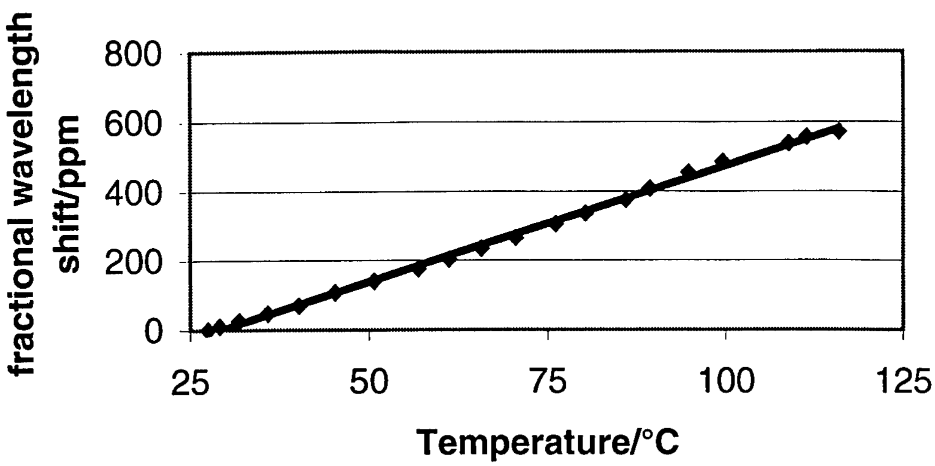


Figure 4b. Response of the same sensor to temperature under constant load.

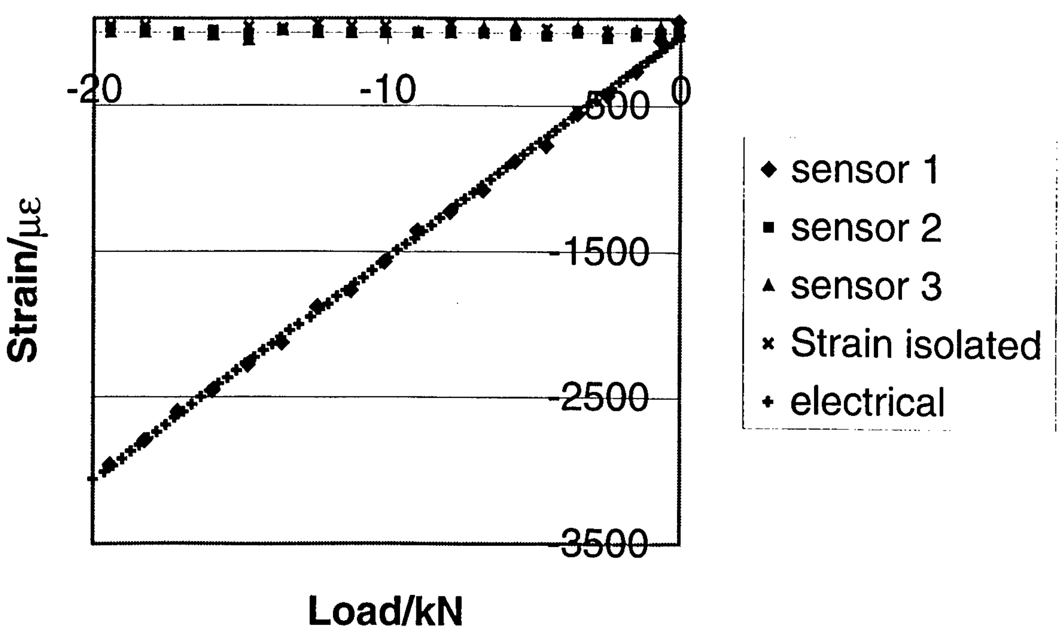


Figure 4c. Combined sensor response for all sensors in the optical rosette strain gauge. The loading direction is parallel to sensor 1 (constant temperature).

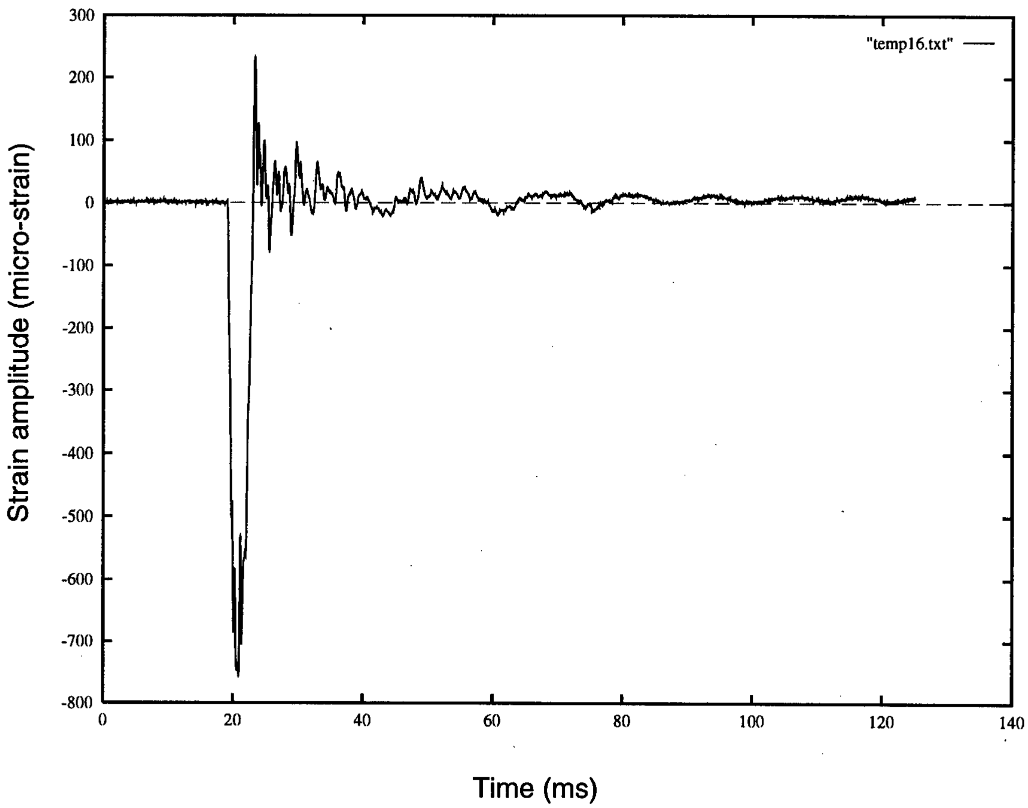


Figure 5. Record of transient strain experienced by an embedded Bragg grating sensor during a 16J impact on the same structure.

Corrosion Modelling and Monitoring:

Neural Networks
&
Neural Network-based Sensor Systems

Prof. Walter Bogaerts
K.U. Leuven - MTM
de Croylaan 2, 3001 Leuven, Belgium
Tel.: +32 / 16 321 222
bogaerts@skynet.be

Neural Networks

Corrosion Modelling

Corrosion-related cracking
(BRITE "*Oracle*" project)

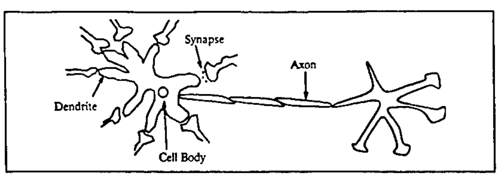
	Austenitic stainless steels	Ni and Ni-based alloys	C-Steels and Low-alloy steels	Titanium
Chloride-induced SCC	X	(X)		
Caustic cracking	X	(X)	X	
IGA and IGSCC by high-temperature water	X	X		
IGA and IGSCC by nitrates			X	
Hydrogen damage	X	X	X	X
and Strain-induced cracking	X	X	X	X

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

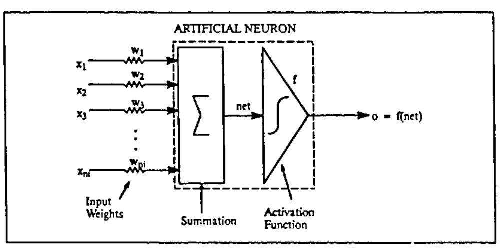
Neural Networks - NN

(Artificial Neural Networks - ANN)

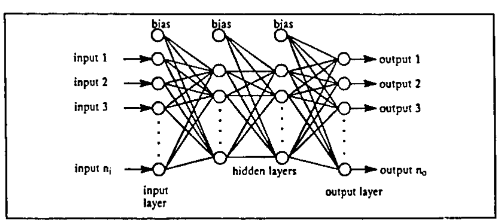
- Biologically inspired
- Mathematical constructs
- Interconnected network of processing elements
- Different types :
  - feedforward / feedback ,
  - supervised / unsupervised learning , ...
  - ➡ ***backpropagation neural network***



Biological neuron



Artficial Neuron

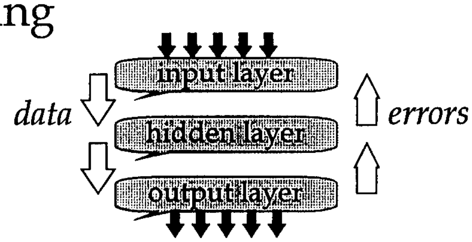


Neural Network

Neural Networks - NN

(Artificial Neural Networks - ANN)

- <u>Different stages</u> :
  - data collection
  - training of the network
  - testing of the network
  - users phase (*e.g.* 'embedding in silicon')
- Learning



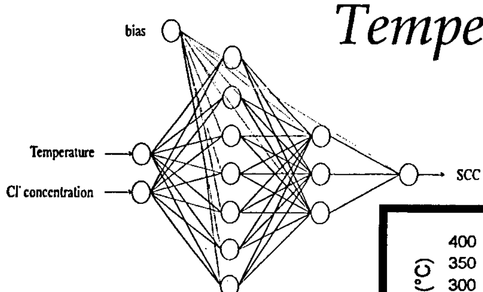
Features

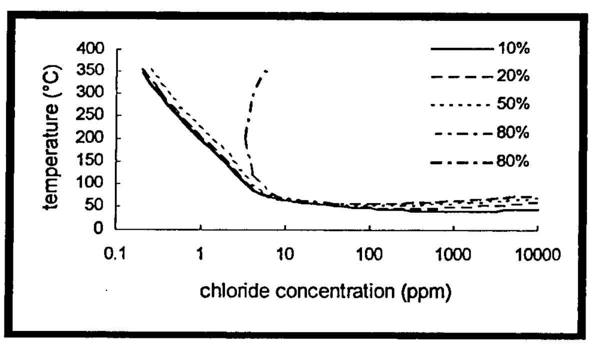
- Advantages
  - learning from examples
  - generalization to new examples
  - abstraction of essential characteristics
  - robustness
- Disadvantages
  - difficulties in training
  - black box

Example :

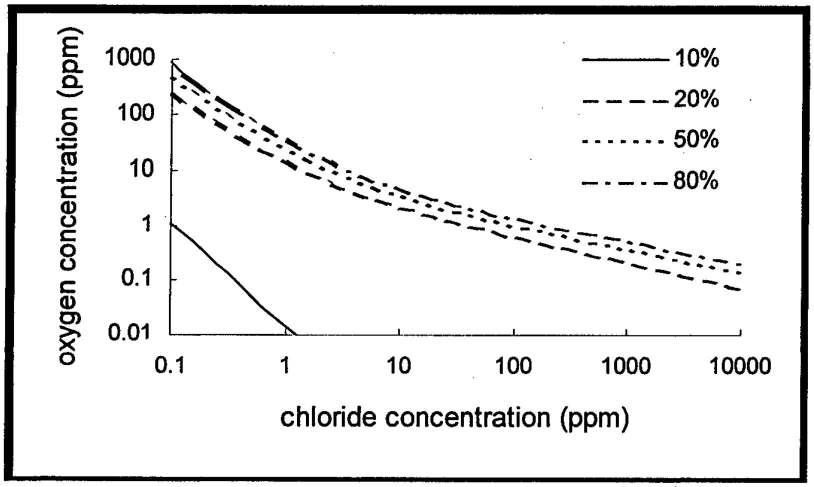
Cl^- cracking *Austenitic stainless steel*

Temperature - chloride

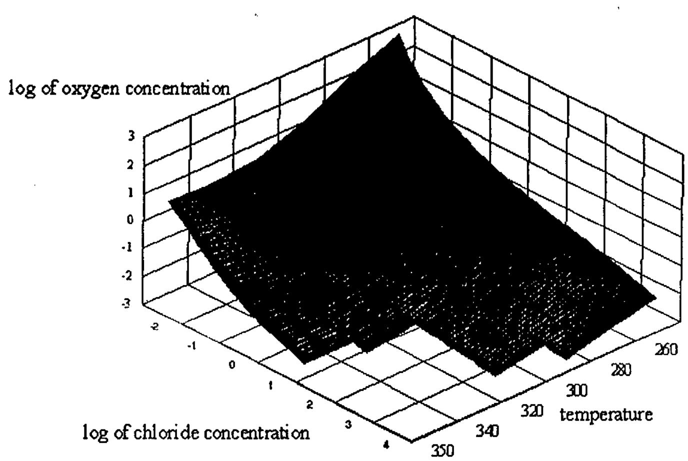




O_2 - Cl^- at elevated temperatures



Temperature, Cl^- and O_2



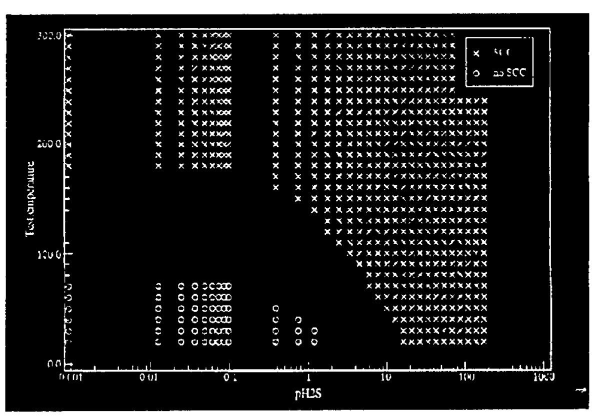
$S^=/Cl^-$ cracking (Hydrogen damage)

Austenitic stainless steels & Ni-alloys

- Material :
  - % Cr, Ni and Mo
  - degree of cold work
  - stress ratio
- Environmental parameters :
  - pH_2S, pCO_2 and NaCl concentration
  - T
  - pH
- Neural network (10 input variables : ***10-dimensional problem***)

Temperature - pH_2S

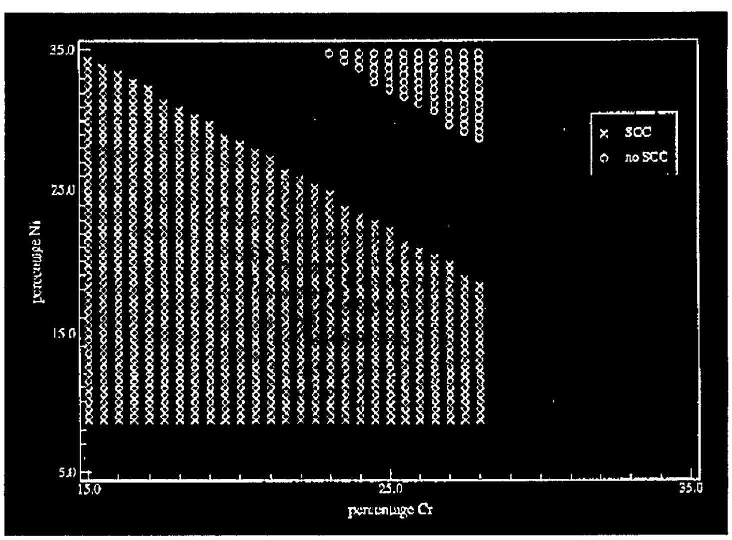
- Cr : 18.4%
- Ni : 9.9%
- Mo : 0%
- degree of cold work : 0%
- Stress ratio : 1.0
- pCO_2 : 10 bar
- NaCl : 3.0%
- pH $\leq$ 3.5



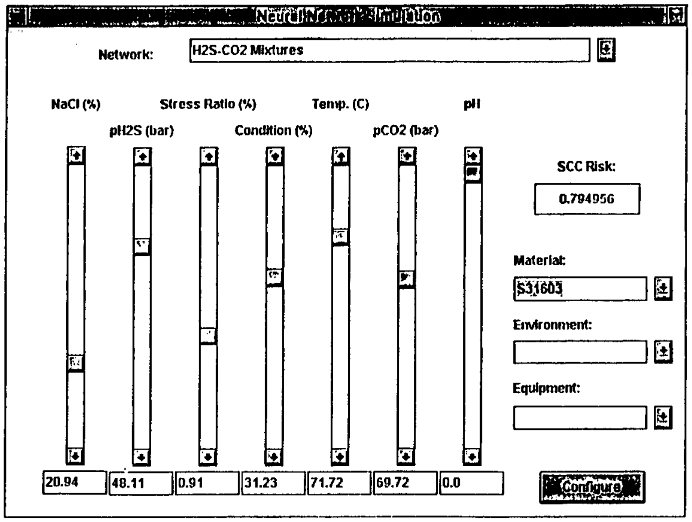
Chromium - nickel

- degree of cold work: 40%
- Mo: 3.0%
- stress ratio: 1.2
- temperature: 200 °C
- pH_2S: 16 bar
- pCO_2: 20 bar
- NaCl: 28.17%
- pH ≤ 3.5

Chromium - nickel



Sample screens...

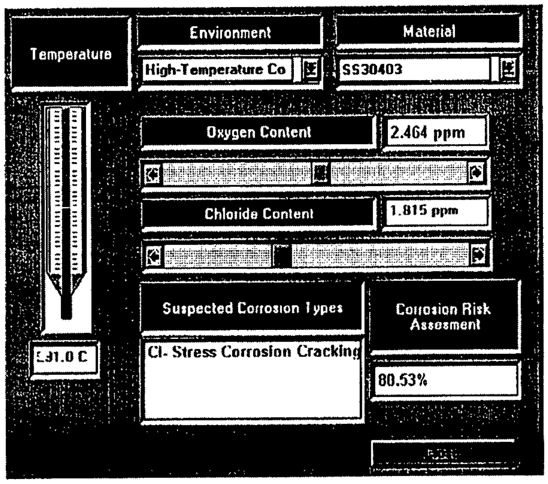


Neural Networks

"Intelligent" SCC-Sensor

- Probe with measurement of
  e.g. T, Oxygen and Chloride
- C-code of neural network
- EPROM
- Result : sensor with *true* "intelligent" data analysis (interpretation)

Corrosion-cracking risk assessment



Corrosion Morphologies

(classification : pits , cracks)

Tasks Neural Networks can Perform

(1a)

- **Classification**
- Self-organization & category formation
- Associative memory
- Modelling & function approximation

TASK :

Sort objects into a simple set of categories;

Classify input patterns into one of a number of possible outputs

Tasks Neural Networks can Perform

(1b)

- **Classification**
- Self-organization & category formation
- Associative memory
- Modelling & function approximation

Examples :

Recognize letters of the alphabet;

Spoken words;

Chemical structure;

Material failure morphology, ...

Tasks Neural Networks can Perform

(1c)

• Classification

Practical Applications :

- Feasibility study ['METANEURAL' development]: Character recognition
- IWT project (with UIA) : Recognition & classification of corrosion morphologies from images --> "intelligent pig"
- Failure analysis - diagnosis [BRITE-Euram "Fractal"]

Tasks Neural Networks can Perform

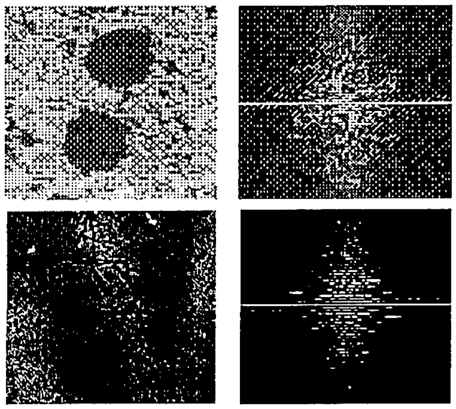
(3b)

Classification
Self-organization & category formation
• Associative memory
Modelling & function approximation

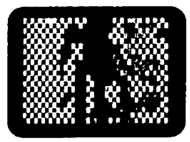
Example :

Retrieve the complete image of an aircraft from a partial trace, ...

Neural Networks — Inspection Failure Analysis



> 96% correct



(e.g. "intelligent pig"/ endoscope images)

Best results

Combination of BP:

different methods to enhance the contrast
different reductions
different preprocessing

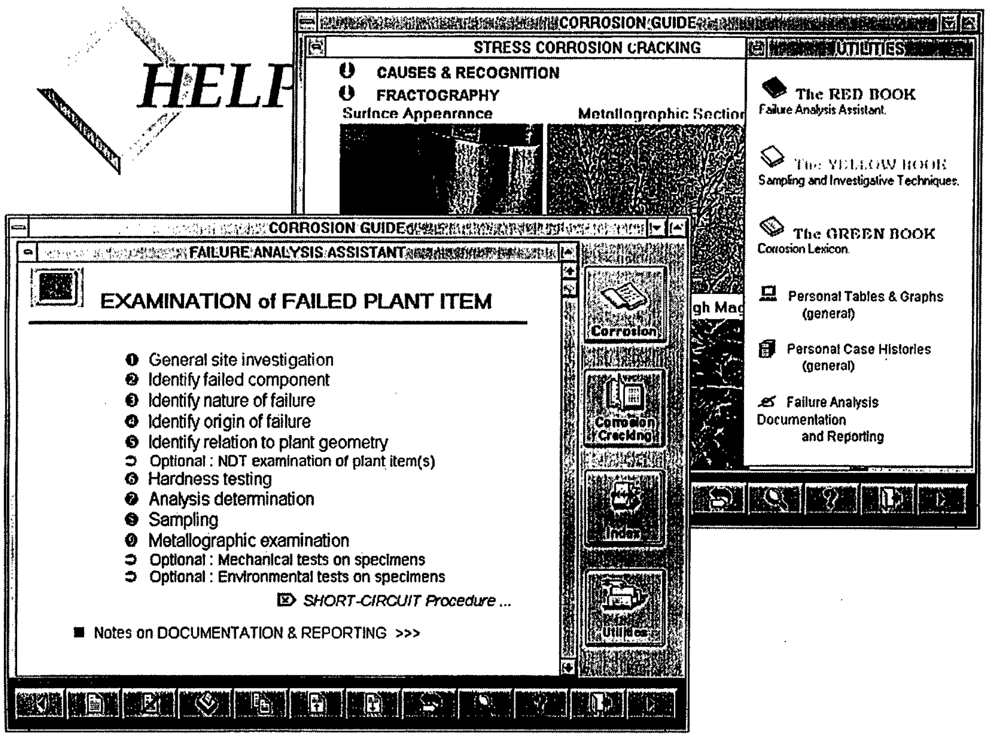
best result by combining 8 BP networks:

training set: 99.28%
test set: 84.78%

Combination of different NNs:

best results by combining 8 NNs:

training set: 98.55%
test set: ~~84.78%~~ ~~85,25~~
~97%



Conclusions

- case histories

  valuable information:

  difficult to extract

- neural networks:

  adept at:

  classification tasks

  data analysis : "superregression"

- case histories + neural net

automated corrosion risk prediction

Conclusions ...

- Neural Networks are suited for a variety of tasks
- Various factors determine whether or not it is opportune to apply NN to a certain application
- NN may be a *viable/suitable* technology if : ...

NN may be a Viable Technology IF ... (1)

- **The problem is not well-understood** :
  models, rules or heuristics describing the problem are not available and are difficult to formulate or expensive to discover
- **The problem area is rich in historical data or examples**
- **The application is data intensive and dependent upon multiple interacting parameters**

STRUCTURAL HEALTH MONITORING OF FULLSCALE COMPONENTS USING ACOUSTIC EMISSION AND FIBER OPTIC SENSORS

Craig B. Van Way, Jayanth N. Kudva
Northrop Grumman Corporation
Military Aircraft Systems Division
Dept. 9B71/63, One Hornet Way
El Segundo, CA - 90245, USA

Capt. V. Scott May, Michael L. Zeigler
Air Force Research Laboratories
Wright Patterson Air Force Base
Area B, Bldg. 45
Dayton, OH, 45433, USA

1.0 ABSTRACT

The principles governing aircraft life cycle monitoring in the United States can be invariably traced back to the Air Force's Aircraft Structural Integrity Program (ASIP) [1]. ASIP, since its implementation in the late fifties, has had unparalleled success at increasing fleet readiness and reducing aircraft down time through the prediction, identification and containment of structural degradation. The ASIP ideal is to field a structure which never develops unanticipated flaws. The reality, of course, is there will always be unexpected problems during the life of a structure. For such eventualities, ASIP has included in its charter a task to provide monitoring of an airframe from its introduction into the fleet until its retirement. And all of this success has been accomplished with, in some cases, relatively rudimentary technologies, including manual flight logs, counting accelerometers, and limited strain gage data.

Recent advances in sensing systems and data processing hardware have made it possible to greatly expand the type and amount of information used to assess the integrity of aircraft structural components. The joint US Air Force/Navy Smart Metallic Structures (SMS) program [2] being conducted at the Northrop Grumman Corporation is investigating new sensors, sensing systems, and hardware architecture technologies to augment current ASIP practices. In particular, the use of acoustic emission (AE) transducers and fiber optic sensors to provide damage detection and strain monitoring in metallic structures have been proven to be viable structural health monitoring techniques. Recent testing under the SMS program has successfully demonstrated each of these systems on increasingly complex structural components from simple coupons up through a full scale F/A-18 wing carry through bulkhead. In addition, a sophisticated distributed hierarchical architecture was created under the program to support the sensor array and provide a method to collect and analyze sensor data from remote processing stations.

This paper first presents an overview of ASIP and its relationship to structural health monitoring (SHM), followed by a brief review of SHM requirements (a more detailed discussion of ASIP and SHM can be found in Reference 3). The bulk of the paper consists of a review of the SMS program, focusing on the health monitoring system architecture (specifically the sensors, processors, and analysis algorithms contained therein), the testing performed under the program, and the lessons learned. Health monitoring implementation payoffs are briefly discussed, and finally, conclusions and a preview of future programs are given.

2.0 ASIP AND STRUCTURAL HEALTH MONITORING

ASIP can be defined as the design and implementation of engineering, test, and logistics tasks to minimize the possibility of catastrophic structural failure resulting from unanticipated or undetected structural or material degradation [4].

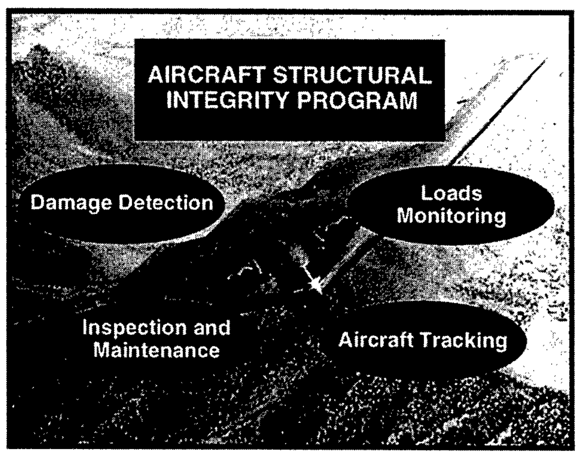


Figure 1. ASIP Functional Tasks

ASIP has fashioned over 20 years of experience acquiring US fleet operational usage data into a comprehensive plan to establish individual aircraft structural integrity. ASIP requires that aircraft structures, both metallic and composite, be capable of withstanding an assumed damage initiation and

Paper presented at the RTO AVT Specialists' Meeting on "Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs", held in Brussels, Belgium, 11-12 May 1998, and published in RTO MP-7.

subsequent growth to an equivalent of two design lifetimes of service usage. Typically, the assumed structural damage is set at the minimum detectable by standard non-destructive inspection (NDI). The inspection interval is determined based on one half of the time it takes for the assumed flaw to become critical. ASIP's success relies heavily on these frequent aircraft inspections to ensure fatigue cracking, or other flaws, get the necessary attention, and appropriate corrective action, before vehicle safety is compromised. To provide fleet managers with critical information on aircraft status, all aircraft flight histories, inspection results, maintenance actions, and damage predictions must be documented in a summary ASIP report.

In recent years there has been a concerted effort from the research and development community [5-21] aimed at further improving the ASIP paradigm by making use of new sensors and data processing techniques to modernize and enhance current tracking methodologies with the ultimate goal of "condition based maintenance". Put simply, aircraft structural inspections and maintenance will be performed only when there is an indication of damage, unlike the current schedule driven system, when costly inspections often find no defects.

Aircraft and other aerospace structures experience various types of flight and operationally induced damage. In metallic structures, the predominant damage modes are fatigue cracking, corrosion, and stress corrosion cracking. Cracks typically occur around fastener holes, cut-outs and sharp bends, and critical crack sizes could be as small as 0.1 inches. A monitoring system will have to be capable of detecting minute cracks in geometrically complex areas where the precise location is not known a priori. While corrosion is a serious problem, especially for naval aircraft, there are no quantitative ASIP requirements for corrosion monitoring, as yet. Difficulties arise in trying to correlate corrosion and performance degradation because the relationship is not quantitatively well understood, or easily predicted by analytical or statistical methods. However, consideration of corrosion is still important for the development of future SHM systems. For composite structures, four important failure/damage modes need to be addressed. These are low-velocity impact damage, delamination, stiffener-skin separation, and failure of bolted and bonded joints. Of these, impact damage caused by dropped tools, runway debris, etc., is the most prevalent and its detection would perhaps have the most significant benefits for an SHMS. Detailed discussions of typical damage modes can be found in References 13-15.

In addition to damage detection, an SHMS must be capable of performing the ASIP functions of loads and environmental spectra survey (L/ESS) and individual aircraft tracking (IAT). The L/ESS program is used to obtain time history records of the parameters necessary for defining the stress state of the airframe, and the IAT program predicts flaw growth in critical areas of the structure. By enhancing current tracking programs, a SHMS would provide increased data validity and return rates, while at the same time improving overall program efficiency.

Structural health monitoring, the concept of instrumenting an aircraft structure with an array of sensors, remote data collection units, high speed communications, and a central processing and analysis computer for the purpose of providing automated damage detection and loads monitoring, has benefited in recent years from significant advances in sensors, data acquisition, electronic miniaturization, and sensor system integration. It is now possible to create a system which can handle both the severe processing demands of modern damage detection techniques, and the huge database management tasks associated with maintaining an entire fleet of aircraft at peak readiness. These developments provide an avenue to apply the wealth of health monitoring technologies that have been fostered by applied research programs over the years to the enhancement of aircraft tracking.

Recent work at the Northrop Grumman Corporation on the joint US Air Force/Navy Smart Metallic Structures (SMS) program [2] has advanced the technology state-of-the-art further, with the successful demonstration of a laboratory prototype structural health monitoring system (SHMS), combining acoustic emission (AE) crack detection, fiber optic strain recording, and aircraft parameter collection with an innovative distributed hierarchical data acquisition and analysis system, to provide a structural integrity assessment capability for an instrumented structure.

3.0 SMART METALLIC STRUCTURES (SMS) PROGRAM

Utilizing advanced monitoring and data processing technologies to design, develop, and test a structural health monitoring system on a large aircraft component is the primary objective of the SMS program. The SHMS was designed to detect fatigue damage in metallic structures using AE transducers and crack gages, and to monitor loads and usage parameters with a combination of low cost fiber optic extrinsic Fabry-Perot interferometers (EFPI) sensors, foil strain gages, and accelerometers (for the remainder of this paper, all discussions of an SHMS will refer specifically to the SMS SHMS). A highly sophisticated data acquisition network was created to support the sensor array by collecting and storing the generated data, and a series of algorithms were incorporated to perform fatigue damage location, crack growth analysis, and data reduction.

3.1 Sensors

AE and fiber optic sensors have been used extensively in health monitoring applications. AE based sensors provide the only established technique capable of remote damage sensing, i.e., they do not have to be located in the immediate vicinity of the damage [11, 12]. Fiber optic sensors exhibit high strain sensitivity, excellent strain resolution, and multiplexing capability. Additionally, fiber optic sensors demonstrate an immunity to electromagnetic interference

(EMI), a common source of electrical noise in an aircraft environment [13]. Foil strain gages and accelerometers are commonly used in existing aircraft tracking programs, and thus are an important consideration in the overall design of an aircraft SHMS.

3.2 System Architecture

Figure 2 shows a schematic of the SHMS architecture. It consists of a distributed network of AE transducers, strain gages, accelerometers, and crack gages controlled by two different remote hosts, the acoustic emission smart sensor (AESS), which monitors the AE sensors, and the utility smart sensor (USS), which measures structural loads and aircraft environment via the accelerometers, strain, and crack gages (Figure 3). The central computer, which consists of a host processor, the input/output (I/O) controller board, and a digital signal processor (DSP), is connected to the sensor controllers via two separate high speed data buses, and performs the functions of data collection, analysis, and storage (Figure 4). The Mil-Std-1553B board interfaces with the analysis and display software contained in the structural health assessment and review program (SHARP) [14], and would, in an aircraft implementation, provide communications with the flight computer and the standard flight data recorder (SFDR). The system architecture is designed to be modular and adaptable to different structural and system configurations. Raw sensor data is collected from each of the monitored zones and passed to the local processors (AESS, USS), and the central processor interrogates each of the local processors in turn to extract the sensor data.

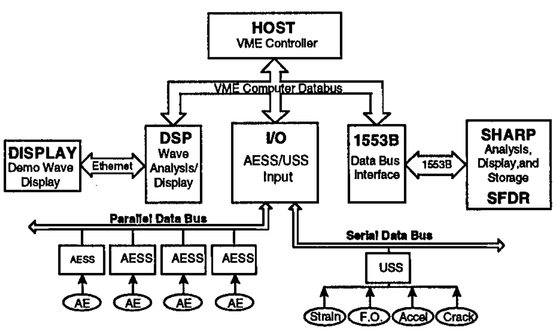
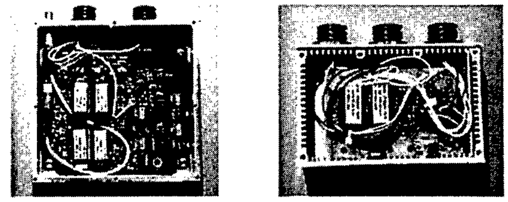


Figure 2. SHMS System Architecture



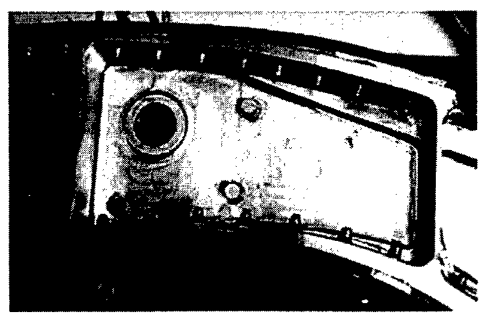
3.a AESS **(b) USS**

Figure 3. Remote Hosts

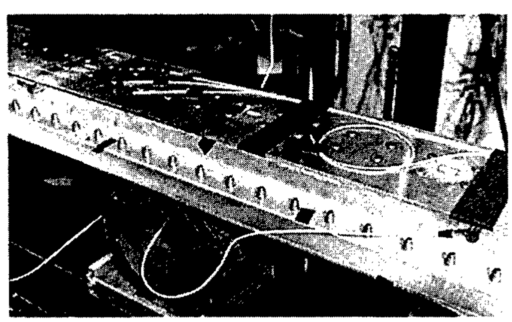
Figure 4. Central Processor

3.3 Sub-element Testing

The system was tested on a series of complex sub-elements, such as those shown in Figures 5 (a) and (b), which verified both the ability of the AE system to identify flaws and of the fiber optic system to monitor strain in complex structures.



(a) Multiple Bay Specimen



(b) Wing Spar Specimen

FIGURE 5. SUB-ELEMENT TEST ARTICLES

However, while the system was successful in finding cracks, the tests identified the difficulties in detecting a propagating acoustic signal. Namely, the signals can be easily detected at locations as far away as 18 inches in a simple geometry like a wing spar web, but in more complex geometric configurations, such as the stiffeners between the bays in a machined bulkhead, or in the rows of fasteners of a built-up wing spar, the crack signals are much more difficult to pick up. The AE sensors had to be located within a few inches of the flaw to obtain positive identification in these cases. The fiber optic sensors performed flawlessly in all the tests, demonstrating excellent fidelity to the point of measuring small strain variations resulting from out of plane bending in the wing spar which were not detectable with standard strain gages. These tests are discussed in detail in References 20 and 21.

3.4 Element Testing

The next program milestone was used to evaluate the operation of the SHMS hardware and software components, including the USS, AESS, central processor, storage units, and data communication, and their ability to collect the critical load and damage data. This test had two goals: 1) to integrate the independently developed damage and loads monitoring modules, and 2) utilize the system to monitor a complex structure. The test article, representative of the F/A-18 F.S. 488 bulkhead, was a machined specimen consisting of four bays separated by stiffeners, with access holes in two of the bays (Figure 6). Crack initiation notches were placed in the specimen at the 9 o'clock position of both access holes.

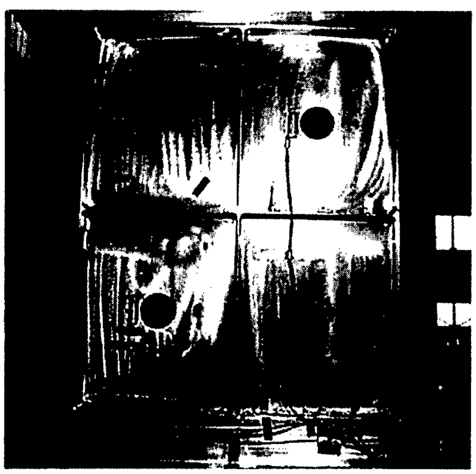
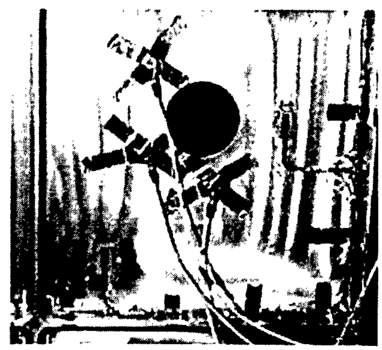


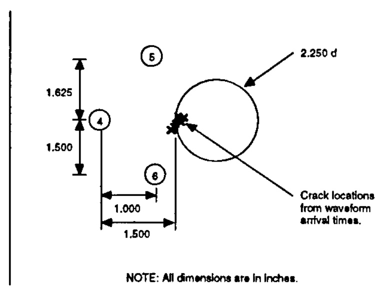
FIGURE 6. ELEMENT TEST ARTICLE

The AE sensors were configured in an array around the access holes in the upper right and lower left bays (sensors not visible in figure). Figure 7 shows a typical configuration of the sensors around one of the access holes, and the results of the post test crack location analysis. The figure shows the high accuracy of the AE source location method used during this test.

Figure 8 shows a typical crack waveform from the lower left access hole and its corresponding frequency content. It can be seen in the figure that the signal has a well defined time of arrival at each of the sensors (at roughly 40 ms) and contains significant frequency content from 200 kHz up to 1.5 MHz, both strong indicators of a crack signal. Additionally, a waveform generated by a fatigue crack typically has a low pre-trigger energy content, defined as the area under the curve in the first 25% of the signal. As can be seen in the figure, this particular event has a very low amplitude prior to the arrival of the signal. (In the next section a direct comparison of a crack signal and a structural noise event is made showing these distinguishing characteristics even more clearly.)

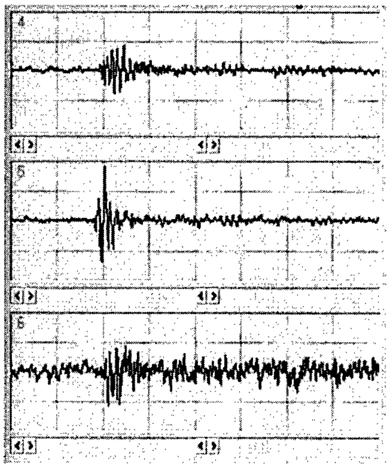


(a) Typical Sensor Locations



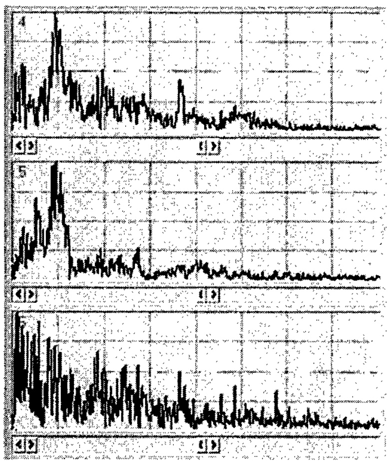
(b) Typical Source Location Results

FIGURE 7. ELEMENT TEST SENSOR LOCATIONS



(a) TIME VS. AMPLITUDE

(TIME/DIV= 20.48 ms)



(b) FREQUENCY VS. AMPLITUDE

(FREQ./DIV= 390.6 kHz)

FIGURE 8. TASK 3 TEST CRACK WAVEFORM

3.5 Final Demonstration Test

The final test of the SMS program was conducted at the Wright Patterson Air Force Base Structural Test Facility, Dayton, Ohio, USA, from May 15 to June 15, 1997. The final demonstration article, the F/A-18 C/D F.S. 488 wing attach bulkhead (Figure 9), was chosen based on its applicability to both Air Force and Navy aircraft structures and because of existing detailed analysis results and ready availability of hardware (the bulkhead is manufactured by Northrop Grumman). The bulkhead was instrumented with the SHMS system and subjected to static and fatigue loading. During the test, approximately 9000 equivalent spectrum flight hours were applied to the bulkhead, leading to a catastrophic fatigue failure just above the wheel well cutout area on the bulkhead (Figure 10).

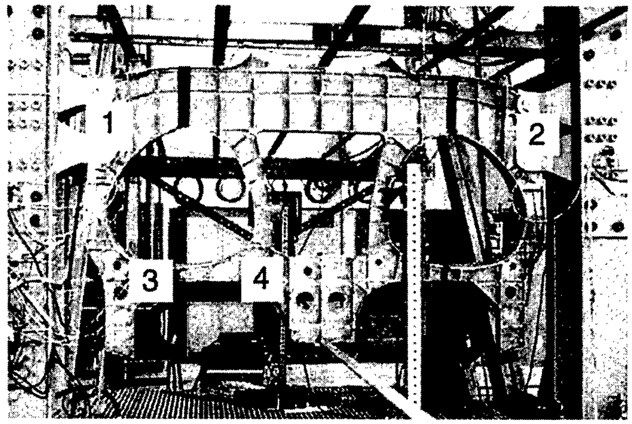


FIGURE 9. F/A-18 WING ATTACH BULKHEAD MOUNTED IN FATIGUE TEST FRAME.

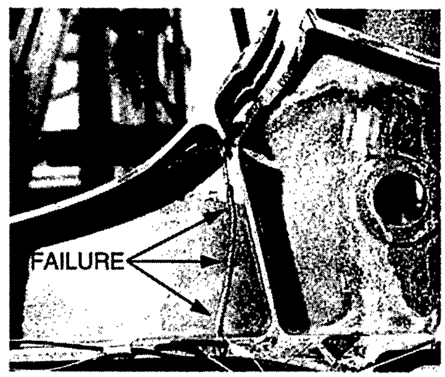


FIGURE 10. CATASTROPHIC FATIGUE FAILURE OF BULKHEAD

Eighteen AE sensors were used to monitor four locations on the bulkhead: zones 1 and 2 are the wing attach lug areas on both sides of the bulkhead, zone 3 is the two access holes in the lower left hand region of the bulkhead, and zone 4 is the inboard wheel well radius on the left hand side. The anticipated failure site was either of the wing attach lug areas based on historical data, and thus these areas received the highest sensor coverage. The failure, however, occurred in the wheel well radius (zone 4, Figure 8), which had been identified as a high stress area in the finite element model, and for this reason was selected as a monitoring zone. No crack indications were found during a scheduled eddy current inspection of the area after 6000 flight hours. The failure came just prior to the 9000 hour inspection, meaning the cracks initiated and grew to failure during this 3000 hour interval.

Crack signals were first detected by the SHMS at approximately 7000 spectrum flight hours. The AE analysis system was able to discriminate signals due to crack growth from background mechanical and structural noise. Figure 10 shows the sensor arrangement in the wheel well zone, and Figures 11 and 12 show time versus amplitude and frequency versus amplitude plots for a structural noise event (Figure 11) and a crack event (Figure 12). Inspection of the noise signal shown in Figure 11 reveals no distinct event arrival in the time domain portion, and the corresponding frequency plot indicates most of the event energy is below 200 kHz. As stated earlier, both of these characteristics indicate that the recorded event is not related to crack growth. Furthermore, the pre-trigger energy content of this waveform is quite high, another strong indicator that it is not a crack event.

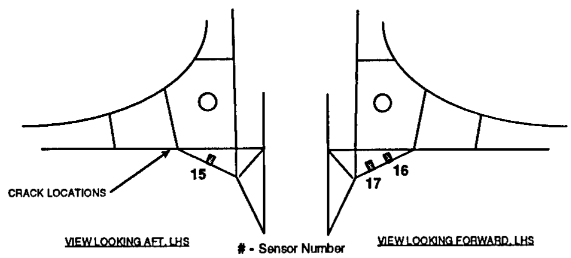
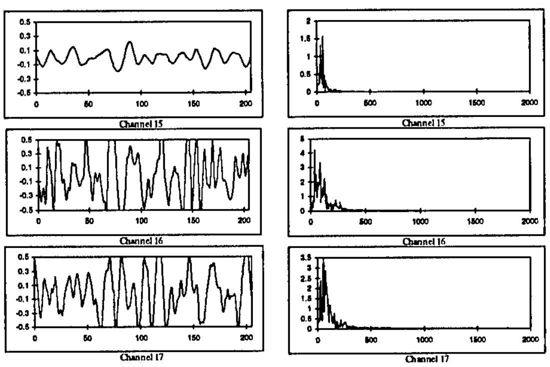
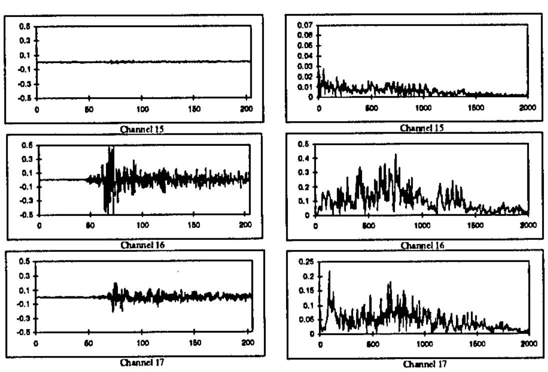


FIGURE 10. WHEEL WELL RADIUS SENSOR LOCATIONS.



TIME VS. AMPL. FREQ. VS. AMPL.

FIGURE 11. STRUCTURAL NOISE EVENT.



TIME VS. AMPL. FREQ. VS. AMPL.

FIGURE 12. CRACK GROWTH EVENT.

However, the crack growth event shown in Figure 12 displays all of the characteristic features. In the time domain, the signal has a distinct time of arrival at channels 16 and 17 (sensor 15 had come loose during this portion of the test). Additionally, in the frequency space, it can be seen that this event contains significant energy in the higher frequencies, indicating a crack event. Numerous similar events were evident in the collected data in the hours leading up to the eventual failure.

Continuous strain data was recorded during the test with both foil and fiber optic strain gages. The five foil strain sensors being monitored by the USS controller units were collocated with the AE sensor groups to record continuous strain traces as well as instantaneous strain levels during acoustic events for use in the AE analysis. An additional eleven fiber optic strain gages were distributed over the bulkhead to monitor and record strain levels in critical regions. One of the fiber optic gages used during the test is shown in Figure 13. The gage is in the center of the picture and is installed on the bulkhead with a kapton substrate (seen as a dark rectangle in the figure). A significant result of the test was that the foil gages began to fail in fatigue as early as 2,000 spectrum flight hours, whereas the fiber optic gages survived the entire test with no signal degradation.

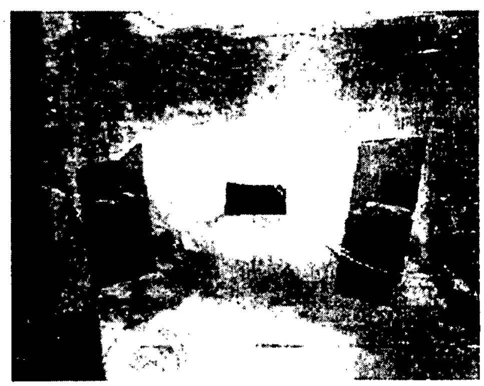


FIGURE 13. SHMS FIBER OPTIC STRAIN GAGE.

Overall, the test can be seen to be a successful demonstration of the prototype SHMS developed under the SMS program. Each of the program objectives were met or exceeded: 1) metallic fatigue events were detected and located using AE, 2) strain was monitored continuously using fiber optic sensors, 3) data from multiple sensor types was collected and stored, and 4) a full scale aircraft structural component was monitored under fatigue cycling. The only drawback of this test, which is currently being addressed, was the fact that the damage detection algorithms were run post-test. That is, the source location algorithms based on the AE data had to be run off-line. Thus, there was no way to predict when the bulkhead would fail, only that the crack signals were present just prior to failure. Automation of the crack location algorithms such that damage sources can be identified in real time is the single most important future development for the metallic damage detection system.

4.0 SYSTEM BENEFITS AND PAYOFFS

The long-term goal of health monitoring is to create an aircraft with a 'condition based maintenance' system, i.e., replace analytical flaw tracking with automated structural damage detection and evaluation. While ideal performance capabilities have yet to be attained, health monitoring applications could enhance current methods of analytic flaw tracking. Payoffs include fewer special inspection requirements at repaired or known critical locations as well as improved mission readiness. In addition, initial steps to automate the data collection and transmission procedures can improve data validity and reduce turnaround times.

5.0 CONCLUDING REMARKS AND FUTURE PROGRAMS

The next critical step in the development of an aircraft health monitoring system will be the creation of a flight qualified system. Significant hardware and software development will be required before a system can be installed and flown on an applicable test bed. And while the sensor technologies for monitoring local areas (i.e. hot spots) are fairly mature, a great deal of work will be demanded to automate the process of collecting and analyzing the sensor data, particularly in the acoustic emission damage detection arena. Additionally, one of the largest gaps in current research efforts is the consideration of damage detection and loads monitoring schemes for composite materials. New programs currently starting at NGC and elsewhere are focused on developing the required technologies in both of these areas. There is also a rising demand for health monitoring technologies in space applications, particularly with the current emphasis on developing a reusable launch vehicle capable of launch turnaround in a matter of days. These and other ongoing efforts collectively will pave the way towards the development of a completely automated SHMS providing total aircraft coverage and resulting in unprecedented levels of fleet readiness, flight safety, and life cycle cost reduction.

6.0 ACKNOWLEDGMENTS

The work documented in this paper was performed under a government contract to a team lead by Northrop Grumman Corporation entitled "Smart Metallic Structures (SMS)," F33615-92-C-3203. The contract monitors are Capt. V. Scott May and Mr. Michael Ziegler of the Air Force Wright Laboratories and Dr. James Alper of the Naval Air Warfare Center.

The members of the multidisciplinary SMS team include Honeywell Corporation (system hardware development), Digital Wave Corporation (acoustic emission monitoring), Mission Research Corporation (system software), and Fiber & Sensors Technologies (fiber optic sensing system) and the support provided by these organizations is greatly appreciated.

7.0 REFERENCES

1. MIL-STD-1530A, Military Standard, Aircraft Structural Integrity Program, Airplane Requirements, December 1975.
2. Van Way, C. B., et. al., "Development Of An Automated Aircraft Structural Integrity Monitoring System–Overview Of The Air Force/Navy Smart Metallic Structures Program," 37th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Salt Lake City, Utah, April 15-17, 1996.
3. Kudva, J.N., et. al., "Structural Health Monitoring of Aircraft Components," Paper No. 9, AGARD-LS-205, Presented at the 79th Meeting of the AGARD Structures and Materials Panel, Paris, France, Nov., 1996.
4. Handbook of Force Management Methods, AFWAL-TR-81-3079, April 1981.
5. Schmidt, W., and Boller, C., "Smart Structures - A Technology for the Next Generation Aircraft", Paper No. 1 AGARD-CP-531, Presented at the 75th Meeting of the AGARD Structures and Materials Panel, Lindau, Germany, Oct. 1992.
6. Boller, C., Dilger, R., "In-flight Aircraft Structures Health Monitoring Based on Smart Structures Technology", Paper No. 17, AGARD-CP-531, Presented at the 75th Meeting of the AGARD Structures and Materials Panel, Lindau, Germany, Oct. 1992.
7. Hickman, G. A., Gerardi, A., "Application of Smart Structures to Aircraft Health Monitoring", Journal of Intelligent Material Systems and Structures, Vol. 1, pp. 411 - 430, July 1991.
8. Gentry, J. D., et al., "The Impact of Smart Structures on Aircraft Structural Integrity Programs", 1992 USAF ASIP Conference, San Antonio, TX, December 1992.
9. Appels, A. J., Capt., "The cost effectiveness of aircraft structural integrity programs", 1993 USAF ASIP Conference, San Antonio, TX, December 1993.
10. Van Way, C. B., et al., "Integration of smart structures concepts for improved structural integrity monitoring of the T-38 aircraft," 1993 USAF ASIP Conference, San Antonio, TX, December 1993.
11. McBride, S. L., et al., "Acoustic Emission Determination of Crack Presence and Crack Advance During Flight", Journal of Acoustic Emission, Nov. 8, pp. 4 - 7, 1989.
12. Gorman, M. R., "Acoustic emission for the 1990s," Proceedings of the IEEE: Ultrasonics, Ferroelectrics and Frequency Control, 1992.
13. Murphy, K. A., et al., "Elliptical-core, two-mode optical fiber sensor implementation methods," Journal of Lightwave Technologies, November 1992.

14. Gentry, J. D., et al., "Structural health assessment and review program (SHARP) - prototype of an on-board structural health monitoring system," 1993 SPIE Smart Structures Conference, Albuquerque, NM, February 1993.

15. Kudva, J. N., et al., Smart Structures Concepts Requirements Definition (SSCORE) Phase I Interim Report, WL-TR-93-3041, March 1993.

16. Van Way, C. B., et al., Smart Metallic Structures (SMS) Task 1 Report, Northrop Corp., Contract No. F33615-92-C-3203, Feb., 1993.

17. Kudva, J. N., et al., "Smart structures concepts for aircraft structural health monitoring", 1993 SPIE Smart Structures Conference, Albuquerque, NM, February 1993.

18. Marantidis, C., et al., "Acoustic emission in an on-board structural health monitoring system for military aircraft," 1994 SPIE Smart Structures Conference, Orlando, FL, February 1994.

19. Marantidis, C., et al., "Sensor and sensing technologies for structural health monitoring of aircraft," 1993 SPIE Smart Structures Conference, Albuquerque, NM, February 1993.

20. Van Way, C. B., et al., "Aircraft structural health monitoring system development – overview of the Air Force/Navy Smart Metallic Structures program," 1995 SPIE Smart Structures and Materials Conference, San Diego, CA, February, 1995, Paper No. 2443-30.

21. Van Way, C. B., et. al., "Development of an Automated Aircraft Structural Integrity Monitoring System-Overview of the Air Force/Navy Smart Metallic Structures Program," AIAA-96-1615-CP, paper presented at the 37th AIAA SDM Conference, Salt Lake City, UT, April 1996.

REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's References	3. Further Reference	4. Security Classification of Document
	RTO-MP-7 AC/323(AVT)TP/4	ISBN 92-837-1005-3	UNCLASSIFIED/ UNLIMITED

5. Originator Research and Technology Organization
North Atlantic Treaty Organization
BP 25, 7 rue Ancelle, F-92201 Neuilly-sur-Seine Cedex, France

6. Title

Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs

7. Presented at/sponsored by

The Specialists' Meeting of the RTO Applied Vehicle Technology Panel (organised by the former AGARD Structures and Materials Panel) held in Brussels, Belgium, 11-12 May 1998.

8. Author(s)/Editor(s)	9. Date
Multiple	November 1998

10. Author's/Editor's Address	11. Pages
Multiple	252

12. Distribution Statement There are no restrictions on the distribution of this document. Information about the availability of this and other RTO unclassified publications is given on the back cover.

13. Keywords/Descriptors

Military aircraft
Loads (forces)
Equipment health monitoring
Life cycle costs
Cost engineering
Fatigue (materials)
Service life
Fatigue life
Utilization
Helicopters
Machinery health monitoring

14. Abstract

Contains the papers presented at a Specialists' Meeting on Exploitation of Structural Loads/Health Data for Reduced Life Cycle Costs, organised by the Applied Vehicle Technology Panel (AVT) of RTO, in Brussels, Belgium, 11-12 May 1998.

The papers highlight the potential benefits from the exploitation of the information derived from modern and future monitoring systems in terms of improved airworthiness and preventative maintenance. The meeting concentrated on the collection, analysis and use of loads/health data by the military for fleet maintenance and logistics planning. Systems and techniques for data gathering and automated analysis were described by authors from a number of NATO nations to provide a valuable insight into how such systems contribute to reducing life cycle costs of military hardware.

The papers are presented under the following headings:

- Current and future structural monitoring
- Engine/helicopter hums
- Current military/civil experience
- Monitoring systems and analysis

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION
BP 25 • 7 RUE ANCELLE
F-92201 NEUILLY-SUR-SEINE CEDEX • FRANCE
Télécopie 0(1)55.61.22.99 • Télex 610 176

DIFFUSION DES PUBLICATIONS RTO NON CLASSIFIEES

L'Organisation pour la recherche et la technologie de l'OTAN (RTO), détient un stock limité de certaines de ses publications récentes, ainsi que de celles de l'ancien AGARD (Groupe consultatif pour la recherche et les réalisations aérospatiales de l'OTAN). Celles-ci pourront éventuellement être obtenues sous forme de copie papier. Pour de plus amples renseignements concernant l'achat de ces ouvrages, adressez-vous par lettre ou par télécopie à l'adresse indiquée ci-dessus. Veuillez ne pas téléphoner.

Des exemplaires supplémentaires peuvent parfois être obtenus auprès des centres nationaux de distribution indiqués ci-dessous. Si vous souhaitez recevoir toutes les publications de la RTO, ou simplement celles qui concernent certains Panels, vous pouvez demander d'être inclus sur la liste d'envoi de l'un de ces centres.

Les publications de la RTO et de l'AGARD sont en vente auprès des agences de vente indiquées ci-dessous, sous forme de photocopie ou de microfiche. Certains originaux peuvent également être obtenus auprès de CASI.

CENTRES DE DIFFUSION NATIONAUX

ALLEMAGNE
Fachinformationszentrum Karlsruhe
D-76344 Eggenstein-Leopoldshafen 2

BELGIQUE
Coordinateur RTO - VSL/RTO
Etat-Major de la Force Aérienne
Quartier Reine Elisabeth
Rue d'Evere, B-1140 Bruxelles

CANADA
Directeur - Gestion de l'information
(Recherche et développement) - DRDGI 3
Ministère de la Défense nationale
Ottawa, Ontario K1A 0K2

DANEMARK
Danish Defence Research Establishment
Ryvangs Allé 1
P.O. Box 2715
DK-2100 Copenhagen Ø

ESPAGNE
INTA (RTO/AGARD Publications)
Carretera de Torrejón a Ajalvir, Pk.4
28850 Torrejón de Ardoz - Madrid

ETATS-UNIS
NASA Center for AeroSpace Information (CASI)
Parkway Center, 7121 Standard Drive
Hanover, MD 21076

FRANCE
O.N.E.R.A. (Direction)
29, Avenue de la Division Leclerc
92322 Châtillon Cedex

GRECE
Hellenic Air Force
Air War College
Scientific and Technical Library
Dekelia Air Force Base
Dekelia, Athens TGA 1010

ISLANDE
Director of Aviation
c/o Flugrad
Reykjavik

ITALIE
Aeronautica Militare
Ufficio Stralcio RTO/AGARD
Aeroporto Pratica di Mare
00040 Pomezia (Roma)

LUXEMBOURG
Voir Belgique

NORVEGE
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25
N-2007 Kjeller

PAYS-BAS
RTO Coordination Office
National Aerospace Laboratory NLR
P.O. Box 90502
1006 BM Amsterdam

PORTUGAL
Estado Maior da Força Aérea
SDFA - Centro de Documentação
Alfragide
P-2720 Amadora

ROYAUME-UNI
Defence Research Information Centre
Kentigern House
65 Brown Street
Glasgow G2 8EX

TURQUIE
Millî Savunma Başkanliği (MSB)
ARGE Dairesi Başkanliği (MSB)
06650 Bakanliklar - Ankara

AGENCES DE VENTE

NASA Center for AeroSpace Information (CASI)
Parkway Center
7121 Standard Drive
Hanover, MD 21076
Etats-Unis

The British Library Document Supply Centre
Boston Spa, Wetherby
West Yorkshire LS23 7BQ
Royaume-Uni

Canada Institute for Scientific and Technical Information (CISTI)
National Research Council
Document Delivery,
Montreal Road, Building M-55
Ottawa K1A 0S2
Canada

Les demandes de documents RTO ou AGARD doivent comporter la dénomination "RTO" ou "AGARD" selon le cas, suivie du numéro de série (par exemple AGARD-AG-315). Des informations analogues, telles que le titre et la date de publication sont souhaitables. Des références bibliographiques complètes ainsi que des résumés des publications RTO et AGARD figurent dans les journaux suivants:

Scientific and Technical Aerospace Reports (STAR)
STAR peut être consulté en ligne au localisateur de ressources uniformes (URL) suivant:
http://www.sti.nasa.gov/Pubs/star/Star.html
STAR est édité par CASI dans le cadre du programme NASA d'information scientifique et technique (STI)
STI Program Office, MS 157A
NASA Langley Research Center
Hampton, Virginia 23681-0001
Etats-Unis

Government Reports Announcements & Index (GRA&I)
publié par le National Technical Information Service
Springfield
Virginia 2216
Etats-Unis
(accessible également en mode interactif dans la base de données bibliographiques en ligne du NTIS, et sur CD-ROM)

Imprimé par le Groupe Communication Canada Inc.
(membre de la Corporation St-Joseph)
45, boul. Sacré-Cœur, Hull (Québec), Canada K1A 0S7